YET04001

H-IIA
User's Manual

Ver. 4.0

February 2015

MHI LAUNCH SERVICES

PREFACE

This H-IIA User's Manual presents information regarding the H-IIA launch vehicle and its related systems and launch services.

This document contains information for launch services including mission performance capability, environmental conditions, spacecraft and launch vehicle interface conditions, launch operations and interface management. A brief description of the H-IIA launch vehicles and the launch facilities of Tanegashima Space Center is also included. As the H-IIA program is progressing, this document is subject to change and will be revised periodically.

Requests for further information or inquiries related to this manual or interfaces between spacecraft and the H-IIA launch system should be addressed to :

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ABBREVIATIONS

AB	Administration Building
AH	Ampere - Hour
AMP	Amplifier
ASTM	American Society for Testing and Materials
B/H	Blockhouse
CAM	Collision Avoidance Maneuver
CCAM	Contamination and Collision Avoidance Maneuver
CCW	Counter Clock Wise
CDR	Command Destruct Receiver
CDR	Critical Design Review
CFRP	Carbon Fiber Reinforced Plastic
CG	Center of Gravity
CHe	Cryogenic Gaseous Helium
CLA	Coupled Loads Analysis
CM4	Four-point Continuous Monitor
CVCF	Constant Voltage Constant Frequency
CVCM	Collected Volatile Condensable Materials
CW	Clock Wise
C_3	Orbital energy parameter
DLF	Design Load Factor
EGSE	Electrical Ground Support Equipment
EMC	Electromagnetic Compatibility
EMI	Electromagnetic Interference
ESA/ESCC	European Space Agency / European Space Components Coordination
FHF	Fuel Handling Facility
FM	Flight Model
FMAR	Final Mission Analysis Review
GCC	Guidance Control Computer
GCC1	Guidance Control Computer for 1st stage
GCC2	Guidance Control Computer for 2nd stage
GEO	Geostationary Earth Orbit
GFRP	Glass Fiber Reinforced Plastic
GHe	Gaseous Helium
GN_2	Gaseous Nitrogen
GOX	Gaseous Oxygen
GPS	Global Positioning System
GSE	Ground Support Equipment
GTO	Geostationary Transfer Orbit
h	Altitude
h_a	Apogee Altitude
HEPA	High Efficiency Particulate Air
HGS	High-pressure Gas Storage

h_p	Perigee Altitude
HTPB	Hydroxyl Terminated PolyButadiene
ICD	Interface Control Document
i	Inclination
IATA	International Air Transport Association
ICD	Interface Control Document
IMDG	International Maritime Dangerous Goods
INMARSAT	International Maritime Satellite Organization
IPA	IsoPropyl Alcohol
IRD	Interface Requirement Document
IRIG	Inter Range Instrumentation Group
ISAS	Institute of Space and Astronautical Science
ISDN	Integrated Services Digital Network
ITV	Industrial Television
JAXA	Japan Aerospace Exploration Agency
JOP	Joint Operations Plan
LAN	Local Area Network
LB	Launch Support Building
LCDR	Launch Conductor
LE	Liquid Rocket Engine
LEO	Low Earth Orbit
LHS	Liquid Hydrogen Storage
LH_2	Liquid Hydrogen
LOS	Liquid Oxygen Storage
LOSP	Launch Operations Support Plan
LOX	Liquid Oxygen
LP	Launch Pad
LPLF	Lower Payload Fairing
LRR	Launch Readiness Review
LSDM	Launch Site Daily Meeting
LSOM	Launch Site Operations Manager
LSP	Launch Services Provider
LSRM	Launch Site Readiness Meeting
L/V,LV	Launch Vehicle
LVRR	Launch Vehicle Readiness Review
MAR	Mission Analysis Review
MDR	Mission Modification Design Review
MECO	Main Engine Cutoff
MECOM	Main Engine Cutoff Command
MHI	Mitsubishi Heavy Industries, Ltd.
MIS	Mission Integration Schedule
MILSET	Mitsubishi Launch Site Service Team
ML	Movable Launcher
MTCS	Masuda Tracking and Communication Station
N/A	Not Applicable
NASA	National Aeronautics and Space Administration
NASDA	National Space Development Agency of Japan
NDTF	Nondestructive Test Facility
NTO	Nitrogen Tetra Oxide
NVR	Non Volatile Residue
OA	Overall

OBS	On Board Software
OHF	Oxidizer Handling Facility
OIS	Operational Intercommunication Telephone System
OSO	Osaki Staff Office
PAF	Payload Adapter Fitting
PCD	Pitch Circle Diameter
PCS	Probability Command Shutdown
PDB2	Power Distribution Box 2nd Stage
PDP	Power Distribution Panel
PFM	Proto Flight Model
PFM	Post Flight Meeting
PIF	Poly Iso-cyanurate Form
pl	Place
PLA	Payload Adapter
PLF	Payload Fairing
PMAR	Preliminary Mission Analysis Review
PMM	Program Management Meeting
PRD	Program Requirement Document
PRR	Program Requirement Review
PSR	Pre-Shipment Review
PSS	Payload Support Structure
PTFE	Polytetrafluoroethylene
Q	Dynamic Pressure
QD	Quick Disconnect
R	Radius
RCC	Range Control Center
RCS	Reaction Control (gas jet) System
REF	Reference
RF	Radio Frequency
RH	Relative Humidity
RNR	Radio Navigation Receiver
RSRR	Range Safety Readiness Review
SBB	Solid Booster Test Building
S/C,SC	Spacecraft
SCAPE	Self-Contained Atmospheric Protective Ensemble
SCO	Spacecraft Organization (both customer and manufacturer)
SCRR	Spacecraft Readiness Review
SDB2	Sequence Distribution Box 2nd Stage
SECO	Second Engine Cutoff
SECOM	Second Engine Cutoff Command
SECT	Section
SEIG	Second Engine Ignition
SEP	Separation
SFA	Spacecraft and Fairing Assembly Building
SFA2	No.2 Spacecraft and Fairing Assembly Building
SLB	Supporting Launch Building
Sm^3	Standard Cubic Meter
SOW	Statement of Work
SPL	Sound Pressure Level
SPLB	Small Satellite Propellant Loading Building
SPM	Solid Propellant Motor

SPM	Single Point Monitor
SR	Safety Review
SRB	Solid Rocket Booster
SSO	Sun Synchronous Orbit
STA1	No.1 Spacecraft Test and Assembly Building
STA2	No.2 Spacecraft Test and Assembly Building
STA	STAtion (station from reference plane)
STM	Structural Test Model
T.B.C.	To Be Confirmed
T.B.D.	To Be Determined
T.B.R.	To Be Revised
TDRSS	Tracking and Data Relay Satellite System
TIM	Technical Interchange Meeting
TML	Total Mass Loss
TNSC	Tanegashima Space Center
TVC	Thrust Vector Control
TYP	Typical
UHF	Ultra High Frequency
UM	Umbilical Mast
UPLF	Upper Payload Fairing
UPS	Uninterruptible Power System
UTC	Coordinated Universal Time
VAB	Vehicle Assembly Building
VC+UV	Visibly Clean Plus Ultra-Violet
VDC	Voltage Direct Current
VOS	Vehicle On Stand
XLR	X Latching Resilient Rubber Compound
α	Angle of Attack
ϕ	Diameter
ω	Argument of Perigee
Ω	Ascending Node

Chapter 1 INTRODUCTION

1.1 Purpose

This user's manual provides information on MHI Launch Services, H-IIA launch system, and following associated information for potential customers to assess the suitability and compatibility with the spacecraft.

(1) General description of H-IIA launch vehicle and launch site (Tanegashima Space Center, TNSC)
(2) Mission performance
(3) Prelaunch and flight environmental conditions
(4) Interface with the spacecraft and H-IIA launch vehicle
(5) Launch operations
(6) Interface management process

1.2 MHI Launch services

1.2.1 Description

Japan Aerospace Exploration Agency (JAXA) developed the H-IIA launch system including the launch vehicle and launch facilities to upgrade former H-II launch system. Mitsubishi Heavy Industries (MHI) has acted as a launch services provider through technical transfer from JAXA since April, 2003 and has started launch services business since 2007.

All services associated with spacecraft including interface coordination are provided by MHI as well as launch vehicle manufacturing and launch operations at TNSC. Responsibility associated with flight safety, range safety and launch site facility is taken by JAXA. For each mission, MHI assigns a mission manager as a single point of contact to work with the customer and coordinate contractual and technical interface activities for the whole mission. Figure 1.2-1 shows the organizational relationship of customer, JAXA, and MHI and major responsibilities of the Mission Manager.

1.2.2 Definition

(1) Spacecraft Organization(SCO)

SCO is defined as spacecraft owner and spacecraft manufacturer.

(2) Launch Services Provider(LSP)

The H-IIA launch vehicle services provider is Mitsubishi Heavy Industries, who will execute spacecraft launch services with JAXA Tanegashima Space Center and other partner companies.

(3) Mitsubishi Launch Site Service Team (MILSET)

MILSET is a team organized to launch the H-IIA launch vehicle by LSP.

(4) Launch operations

Generic term referring to activities preformed in the launch site, such as assembly, preparation, testing and joint operations of the spacecraft and launch vehicle.

(5) Spacecraft Program Manager

Spacecraft Program Manager is a SCO person who is responsible for all the interface activity with LSP.

(6) Spacecraft Mission Director

Spacecraft Mission Director is a SCO person who is responsible for the launch activity coordination.

(7) Mission Manager

Mission Manager is an LSP person who is responsible for contracts, interface coordination of the launch, and technical matters related to launch operations (from spacecraft arrival at TNSC through completion of the post liftoff operations).

(8) Spacecraft system

Spacecraft system is a system including the spacecraft, associated facilities, and Ground Support Equipment (GSE).

(9) H-IIA launch vehicle system

H-IIA launch vehicle system is a system including the launch vehicle, associated facilities, and GSE.

(10) “L”

“L” is defined as launch day. For example, “L- (number)” represents month(s) or day(s) up to the launch day.

(11) “X”

“X” is defined as launch time. For example, “X- (number)” represents hour(s), minute(s), or second(s) up to lift off.

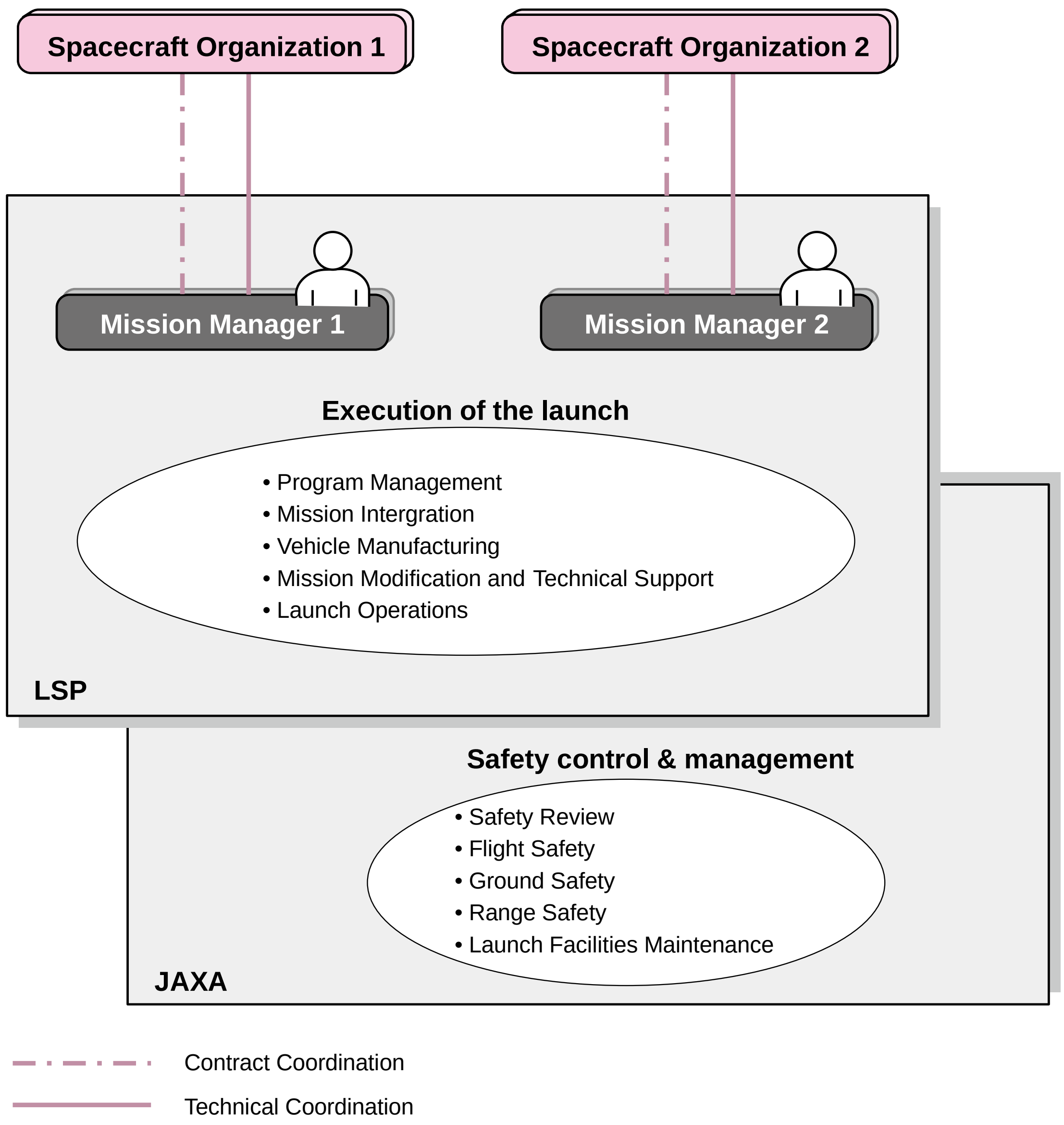


Figure 1.2-1 SCO / LSP / JAXA Relationship

1.3 H-IIA Launch System

1.3.1 H-IIA Launch Vehicle Description

(1) General

Figure 1.3-1 shows H-IIA launch vehicle family, and Table 1.3-1 shows a summary of H-IIA subsystems and characteristics.

The "H2A202" consists of the first stage with two SRB-As, the second stage, and the payload fairing as shown in Figure 1.3-2. The "H2A204" consists of the first stage with four SRB-As, the second stage, and the payload fairing as shown in Figure 1.3-3.

An optional long-coast GTO (KOHDO-KA) is available for customers to effectively minimize the spacecraft onboard propellant needed to reach GEO from GTO and provide an additional lifetime. Overview of the long-coast GTO option is described later in this section 1.3.1 (7).

(2) First Stage

The first stage consists of the high performance cryogenic LE-7A engine, engine section, LH_2 tank, center body section, LOX tank and interstage section. The LE-7A engine produces 1100kN of thrust using liquid hydrogen (LH_2) and liquid oxygen (LOX) as propellant. Two or four solid rocket boosters (SRB-A) are attached to the first stage in order to augment the total thrust during ascent phase.

(3) SRB-A (Solid Rocket Booster)

SRB-As augment total thrust by firing for approximately 100 seconds from liftoff for the H2A202, and 120 seconds for the H2A204. The motor case of the SRB-A is monolithic and made of filament winding composite material (CFRP). Additionally, electromechanical thrust vector control (TVC) system for gimbaling the nozzle is applied.

(4) Second Stage

The second stage consists of the highly reliable cryogenic LE-5B engine, LH_2 tank, LOX tank and avionics system installed on the equipment panel. The LE-5B engine, which produces 137kN of thrust using liquid hydrogen (LH_2) and liquid oxygen (LOX) as propellant, has a multi-restart capability to meet various mission requirements. Attitude control is performed using reliable electromechanical actuators of the LE-5B engine and the hydrazine gas-jet reaction control system (RCS).

The RCS is used for attitude control and propellant settling of the second stage before and after the spacecraft separation. The RCS is mounted under the avionics equipment panel.

Most components of the avionics system are mounted on the avionics equipment panel.

(5) Payload fairings

The payload fairing protects the spacecraft from external environment to which the spacecraft is exposed, from the time of encapsulation through the atmospheric ascent phase.

3 types of payload fairings shown in Figure 1.3-4 and Figure 1.3-5 are available. Each type is compatible with H-IIA launch vehicle family.

Allowable envelop of the spacecraft in the fairing is shown in Figure 1.3-4 for the Model 4S and 5S used for a dedicated launch, and Figure 1.3-5 for the Model 4/4D-LC used for dual launch.

Further description of payload fairings is provided in Chap. 4 and Appendix 4.

(6) Payload adapters / separation system

The spacecraft is mounted on the launch vehicle using a payload adapter shown in Figure 1.3-6.

Several types of payload adapters are available. Each type is compatible with H-IIA launch vehicle family, including separation systems.

Figure 1.3-6 shows the payload adapters with dimensions available to the customer. Detail information on the adapters and separation systems including mechanical interfaces are presented in Chap. 4 and Appendix 4.

(7) Option for long-coast GTO mission

For geostationary orbit mission, long-coast second stage option is available to reduce the delta-V, which the spacecraft is injected into geostationary orbit from the geostationary transfer orbit (GTO), by boosting the second stage near the apogee.

Following modifications applied to the second stage make it possible to enhance the mission time, and suppress the cryogenic propellant loss during long-coast.

The standard GTO mission time is approximately 0.5h, while the long-coast GTO mission is approximately 4.5h.

(a) White painted LH_2 tank
(b) Additional LOX trickling chill down subsystem
(c) Additional GH_2 vent subsystem
(d) LE-5B 60% throttling capability
(e) Additional battery
(f) Miscellaneous improvement for long-time coast mission

Associated information is presented in the following sections.

2.1.1 Mission profile
2.3 Geostationary Transfer Orbit (GTO) Mission

1.3.2 Launch Facilities

The H-IIA launch vehicle is finally integrated and launched from the TNSC in Tanegashima Island, located at the southwest of Japan.

Launch operations are carried out in Osaki Range of the TNSC. Figure 1.3-7 and Figure 1.3-8 shows the location of the TNSC and its major facilities, and the overview of the launch complex, respectively.

Figure 1.3-9 shows the typical launch site operations process. Customer is offered to use the buildings for spacecraft processing as either of the following cases.

Case 1: (Normal)

-For spacecraft checkout, No.2 Spacecraft Test and Assembly Building (STA2)

-For propellant loading and encapsulation, Spacecraft and Fairing Assembly Building (SFA)

Case 2: (Option)

-For spacecraft checkout propellant loading and encapsulation, No.2 Spacecraft and Fairing Assembly Building (SFA2)

Case3: (Backup)

-For spacecraft checkout, No.1 Spacecraft Test and Assembly Building (STA1)

-For propellant loading, Small Satellite Propellant Loading Building (SPLB)

-For encapsulation, Spacecraft and Fairing Assembly Building (SFA)

Table 1.3-1 Summary of H-IIA subsystems and characteristics

Item		H2A202	H2A204	NOTE
Overall length (m)		53	53	
Diameter (m)	: Core	4.0	4.0	
	: Fairing	4.07 / 5.1	4.07 / 5.1	4S, 4/4D type / 5S type
Total weight (ton)	: Without Payload	289	443	include payload adapter
Payload fairing	: Honeycomb sandwich			
4S,5S type	: Clamshell	✓	✓	
4/4D-LC type	: Upper Clamshell, Lower Clamshell	✓	✓	
Payload adapter				
1194M, 1666MA	: Aluminum, V band	✓	✓	
937LS-H,1194LS-H, 1666LS-H	: Low shock separation	✓	✓	Contact LSP for details.
First stage		1	1	
Propellant	: LH_2 / LOX	100 ton	100 ton	usable mass
Tank	: LH_2 aluminum isogrid : LOX aluminum isogrid			
Propulsion	: LE-7A engine x 1 Thrust 1098 kN Isp 440 sec Burning time 390 sec			in vacuum in vacuum Auxiliary engine for roll control
Avionics	: Guidance Control Computer, Flight Termination, Rate Gyro Package, Lateral Acceleration Unit, VHF Telemetry, Electrical Power			use data bus system (MIL-STD-1553B)
SRB-A (for H2A202)		2	N/A	
Propellant (per each)	: HTPB composite Propellant weight 65 ton Thrust (max) 2520 kN Isp 283 sec Burning time 100 sec			in vacuum in vacuum
Motor case	: Monolithic CFRP Diameter 2.5 m			
SRB-A (for H2A204)		(2)	4	Can be also used for H2A202(option)
Propellant (per each)	: HTPB composite Propellant weight 66 ton Thrust (max) 2300 kN Isp 283 sec Burning time 120 sec			in vacuum in vacuum
Motor case	: Monolithic CFRP Diameter 2.5 m			
Second stage		1	1	
Propellant	: LH_2 / LOX	16.6 ton	16.6 ton	usable mass
Tank	: LH_2 aluminum isogrid : LOX aluminum			
Propulsion	: LE-5B engine x 1 Thrust 137 kN Isp 448 sec MR 5.0 Idle mode function Multi-restart function : Reaction control system			in vacuum. 82 kN for optional third burn about 3 % for rated thrust for attitude control
Avionics	: Guidance Control Computer, Inertial Measurement Unit, Flight Termination, UHF Telemetry, C-Band Tracking, Range Safety Command, Electrical Power			Guidance, Navigation, Control and Vehicle Sequencing use data bus system (MIL-STD-1553B)

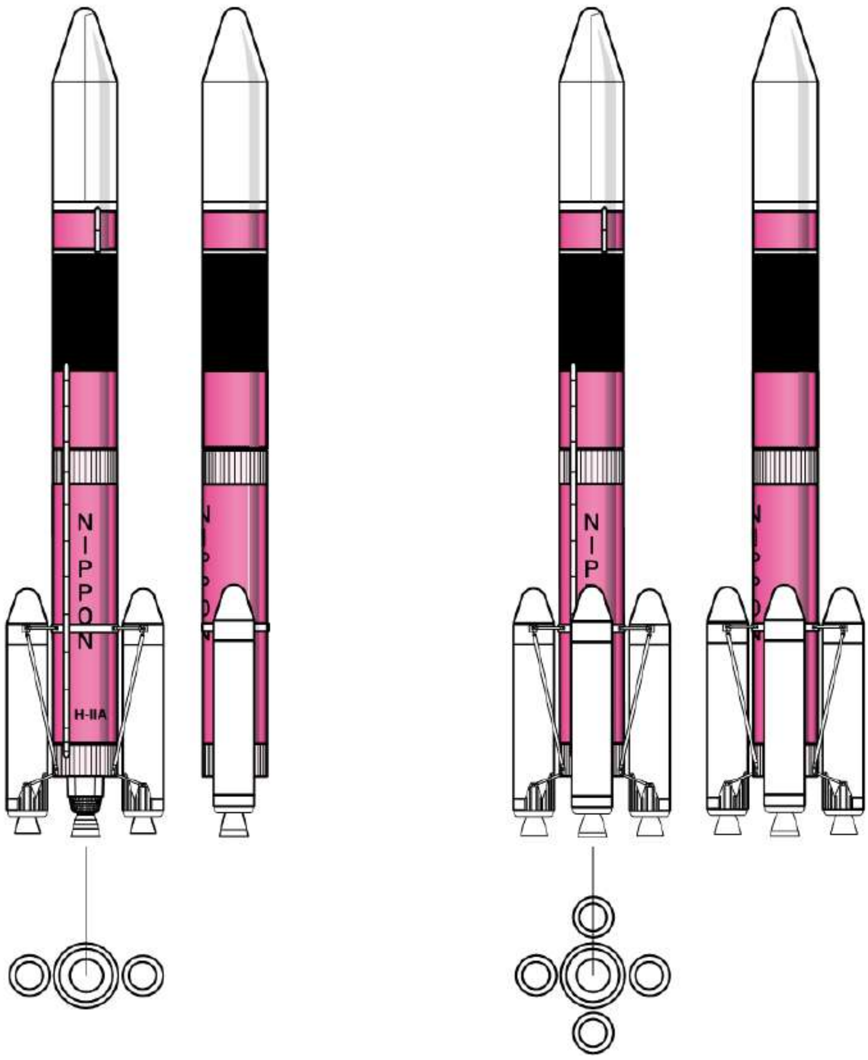
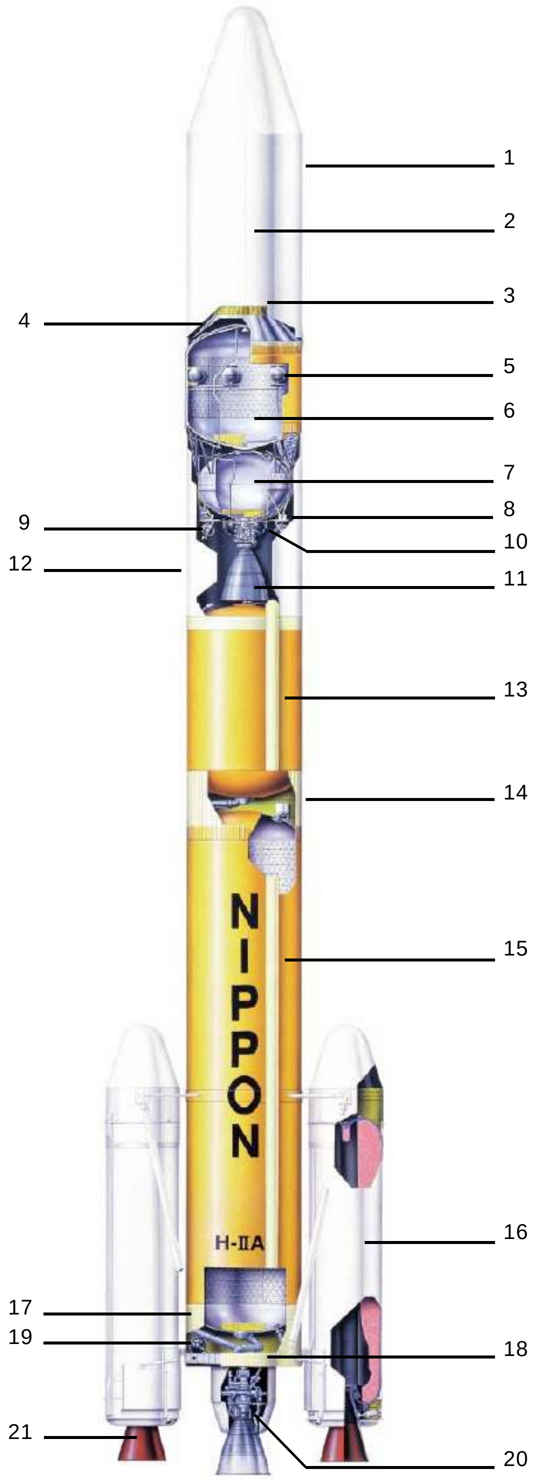
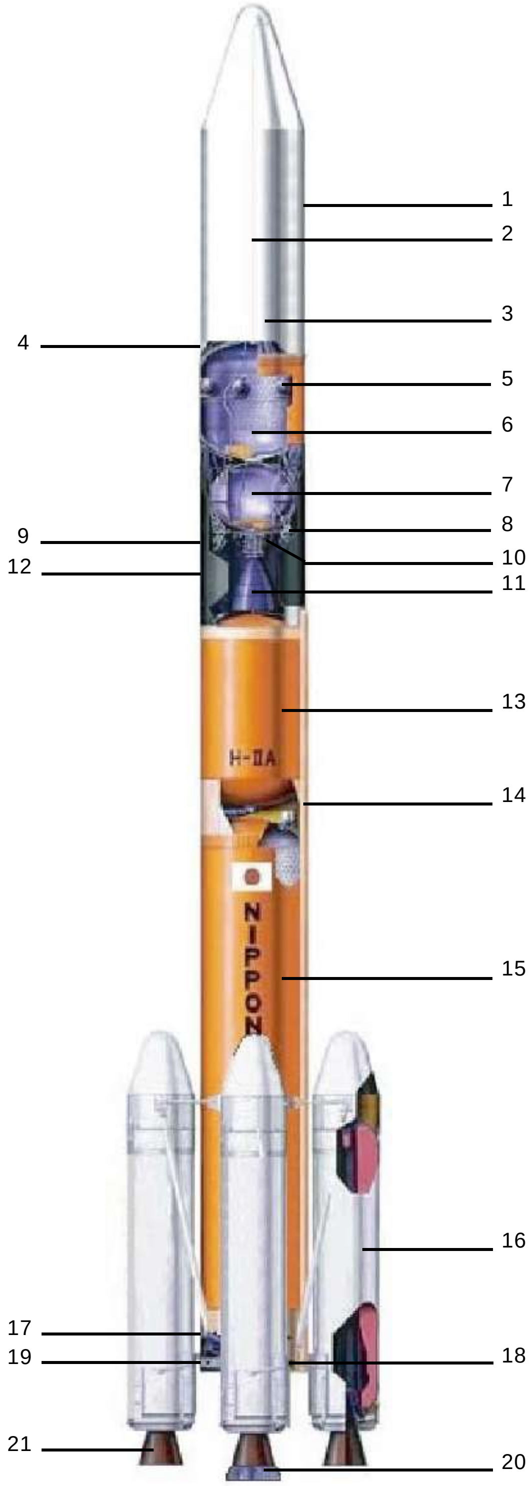


Figure 1.3-1 H-IIA Launch Vehicle Family



1. Payload fairing
2. Spacecraft
3. Payload adapter
4. Payload support structure
5. Cryogenic He bottles
6. Second stage LH_2 tank
7. Second stage LOX tank
8. Avionics Equipment panel
9. Reaction control system
10. Ambient He bottles
11. Second stage engine (LE-5B engine)
12. Interstage section
13. First stage LOX tank
14. Center body section
15. First stage LH_2 tank
16. Solid rocket booster (SRB-A)
17. First stage engine section
18. Auxiliary engine
19. Ambient He bottles
20. First stage engine (LE-7A engine)
21. SRB-A movable nozzle

Figure 1.3-2 H-IIA (H2A202) Launch Vehicle configuration



1. Payload fairing
2. Spacecraft
3. Payload adapter
4. Payload support structure
5. Cryogenic He bottles
6. Second stage LH_2 tank
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8. Avionics Equipment panel
9. Reaction control system
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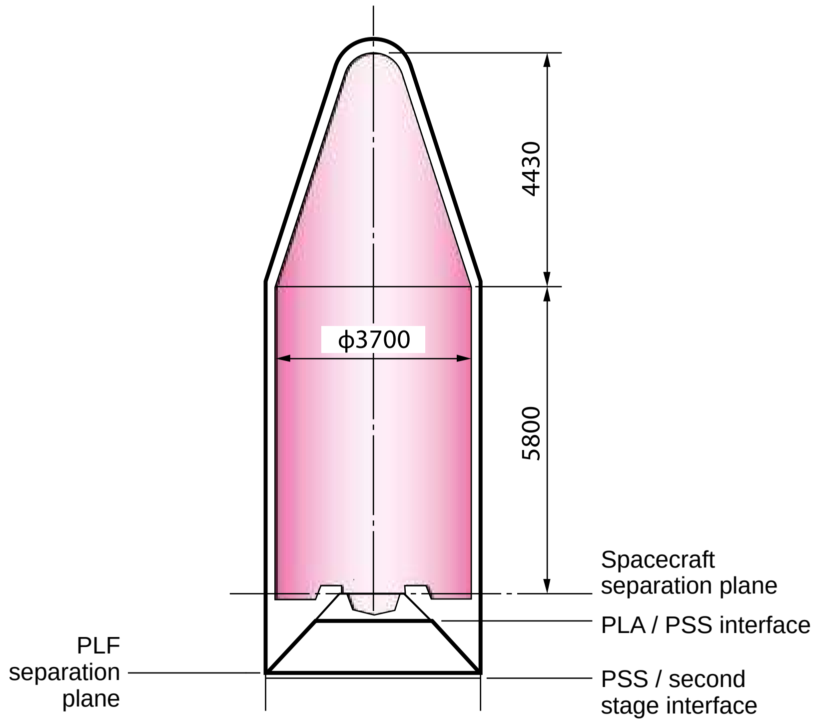
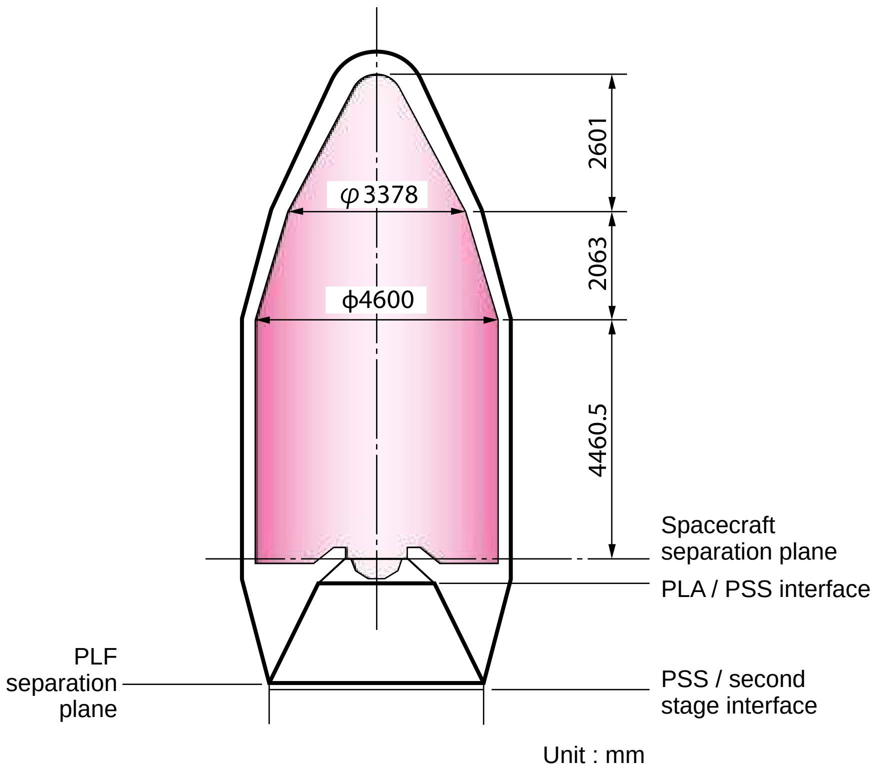
11. Second stage engine (LE-5B engine)
12. Interstage section
13. First stage LOX tank
14. Center body section
15. First stage LH_2 tank
16. Solid rocket booster (SRB-A)
17. First stage engine section
18. Auxiliary engine
19. Ambient He bottles
20. First stage engine (LE-7A engine)
21. SRB-A movable nozzle

Figure 1.3-3 H-IIA (H2A204) Launch Vehicle configuration



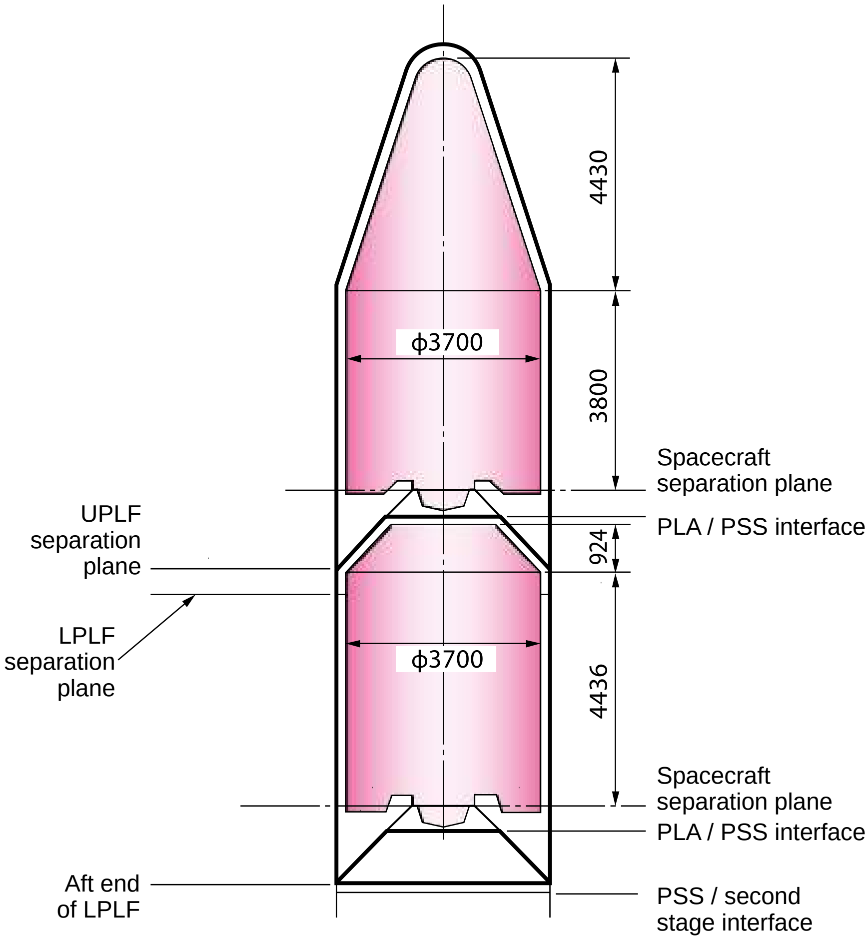
Note) Usable volume diameter is ϕ3700mm~ϕ3750mm (See section 4.5.)

Model 4S



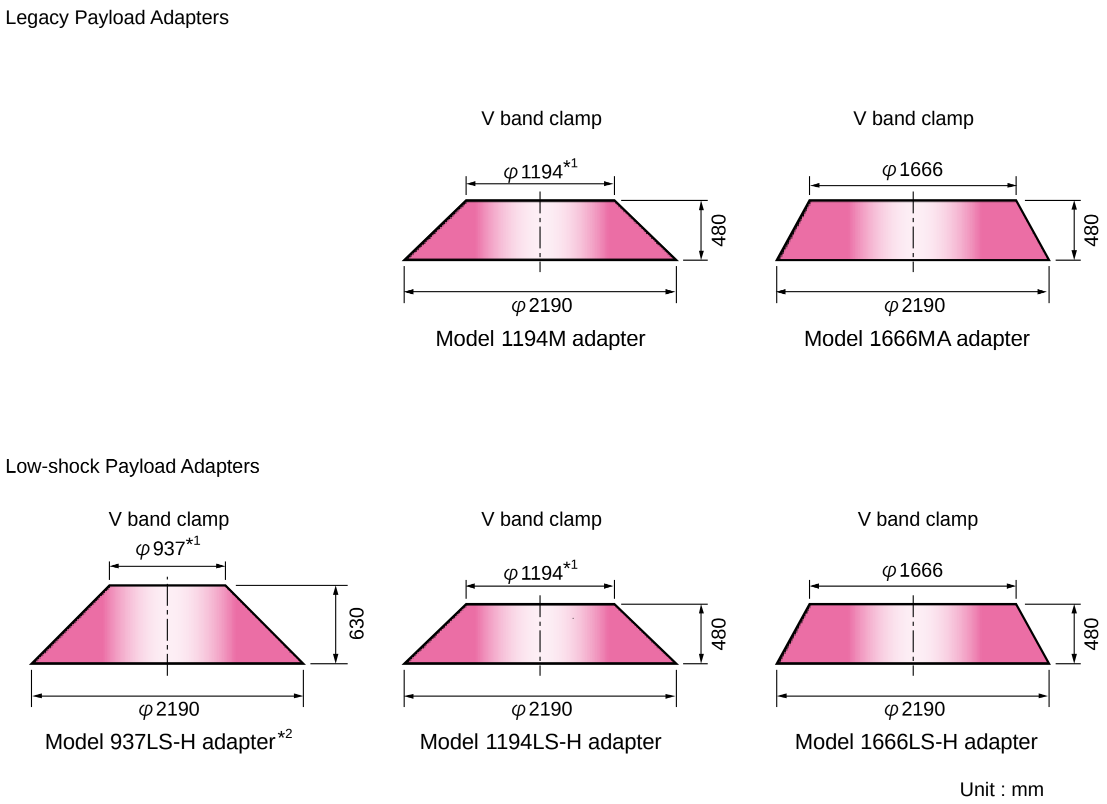
Model 5S

Figure 1.3-4　Payload fairings for dedicated launch



Model 4/4D-LC

Figure 1.3-5 Payload fairings for dual launch



<Note>
*1 : This value shows a nominal interface diameter. (See Appendix 4)
*2 : Contact LSP for details.

LSP can procure other payload adapter upon SCO’s request. Contact LSP for details.

Figure 1.3-6 Payload adapters

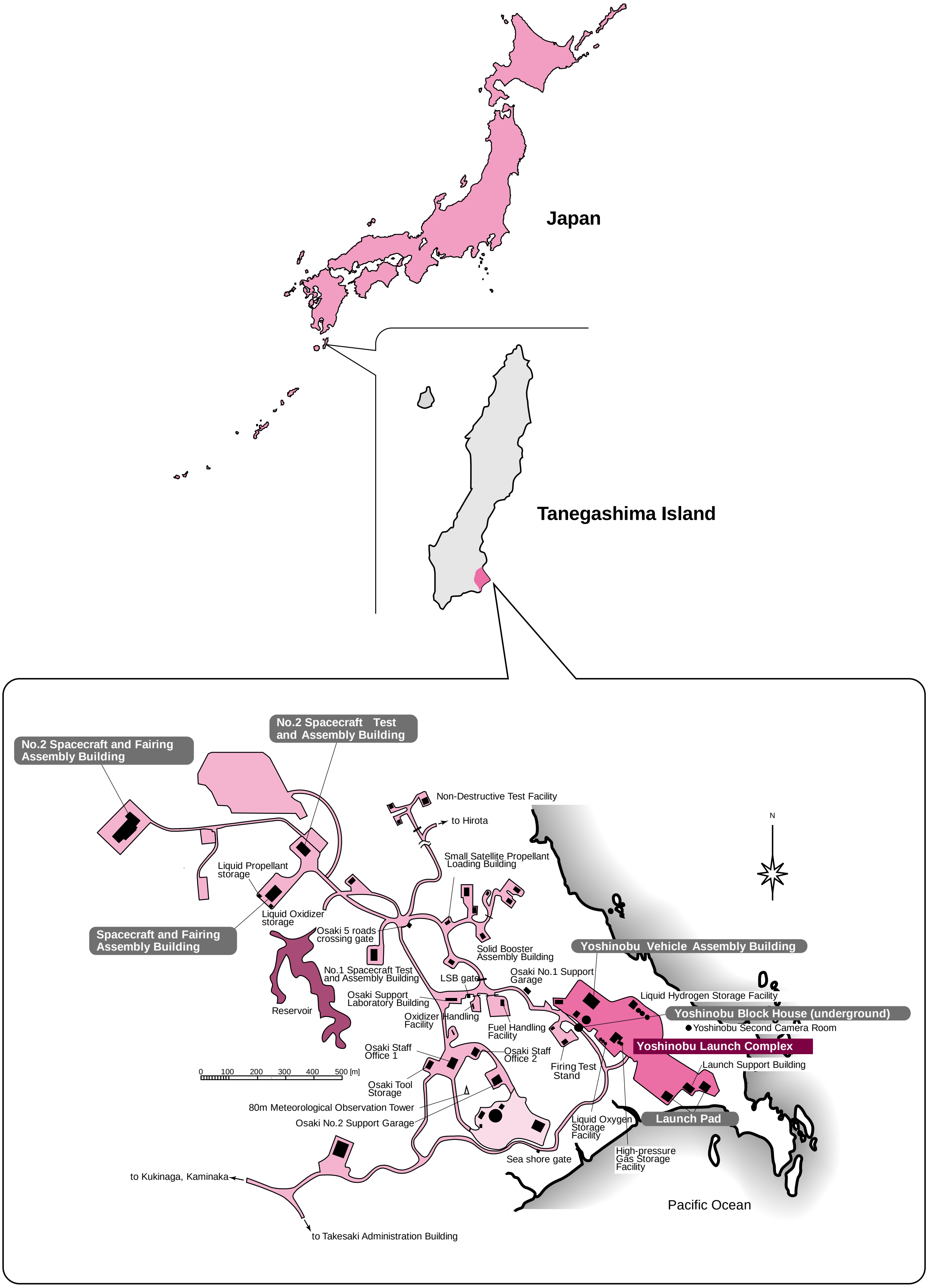
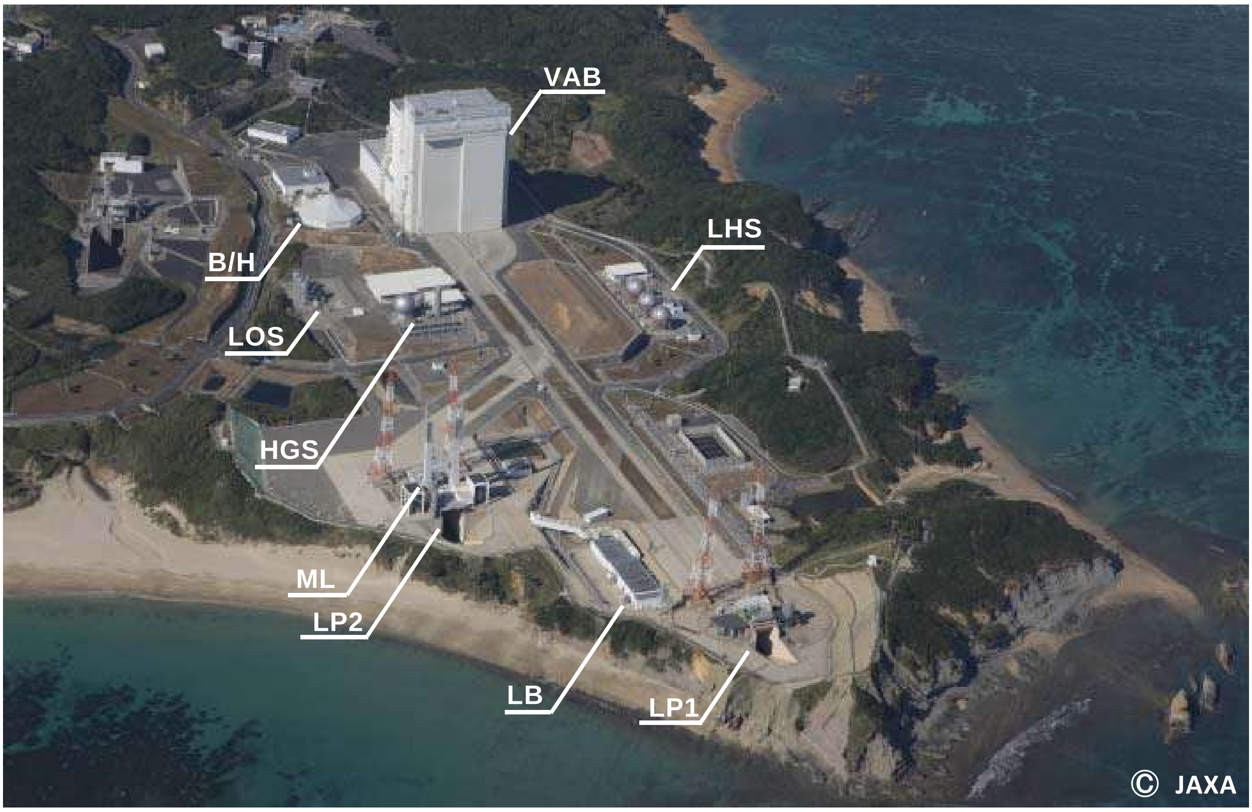


Figure 1.3-7 Location of major facilities in Osaki Range



B/H : Block House
VAB : Vehicle Assembly Building
LP1 : Launch Pad No.1
LP2 : Launch Pad No.2
ML : Movable Launcher
LOS : Liquid Oxygen Storage Facility
LHS : Liquid Hydrogen Storage Facility
HGS : High-pressure Gas Storage Facility
LB : Launch Support Building

Figure 1.3-8 Overview of Yoshinobu Launch Complex

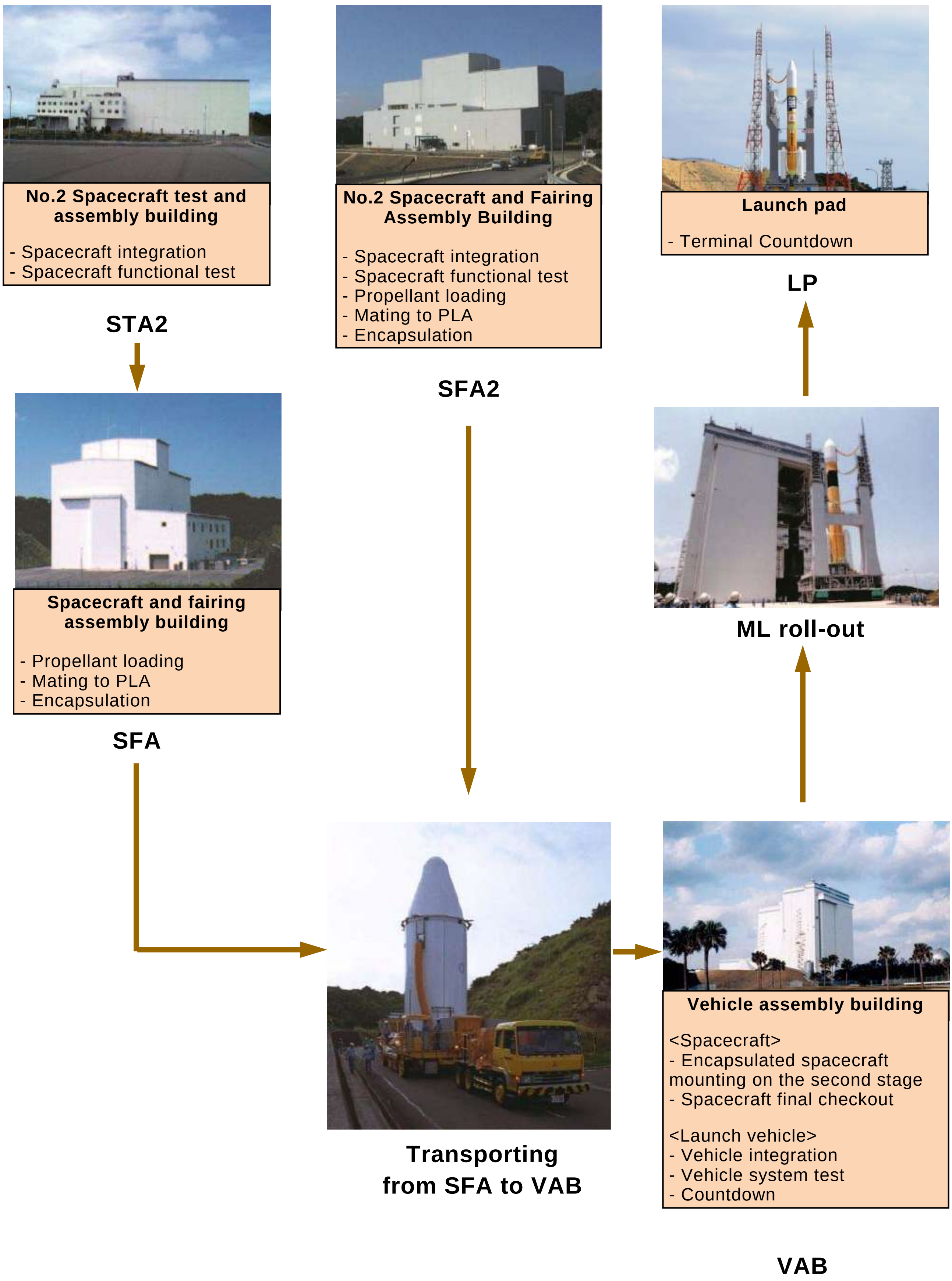


Figure 1.3-9 H-IIA launch operations process

1.4 H-IIA Launch System Related Document

In addition to this user's manual, following two documents are available for the customer.

(1) Facility Guide for H-IIA Payload Launch Campaign
(2) Launch Vehicle Payload Safety Standard

These documents provide the customer with detailed information on launch facilities and range safety requirements as a reference for the preliminary planning phase.

1.4.1 Facility Guide for H-IIA Payload Launch Campaign

Buildings and associated facilities in TNSC and a description of the spacecraft launch site operations are presented in this facility guide.

1.4.2 Launch Vehicle Payload Safety Standard

The requirements for safety control, spacecraft safety design, and launch site operations at TNSC are described. In this user's manual, outline of safety requirements and description of safety reviews is presented in section 6.4.

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Chapter 2 PERFORMANCE

2.1 General

This chapter describes the performance of the H-IIA launch vehicle. The H-IIA launch vehicle accommodates a various mission using 2nd stage restart capabilities to meet SCO's requirements.

In this manual, following mission profiles are presented for preliminary planning purpose:

(1) Geostationary transfer orbit (GTO) mission
(2) Sun synchronous orbit (SSO) mission
(3) Low earth orbit (LEO) mission
(4) Earth escape mission

Typical launch capabilities of the H-IIA launch vehicle family is shown in Table 2.1-1. The performance data in this chapter are based on standard mission modifications of payload fairing described in Section 4.5. Depending on the final configurations and mission requirements of SCO, actual launch capability will vary. Regarding detailed launch capabilities, please contact LSP.

2.1.1 Mission profile

This section describes typical mission profile and sequences of GTO mission for example. Typical GTO mission profiles are shown in Figure 2.1-1 and Figure 2.1-2. Typical sequence of events for standard GTO mission and long-coast GTO mission are shown in Table 2.1-2. For other missions such as SSO and LEO, the time for each sequence of events and flight trajectories will vary.

In the following paragraphs, time for sequence of events for H2A202 is representatively used corresponding to Figure2.1-1.

2.1.1.1 Boost and first stage phase

The main engine LE-7A is ignited at approximately 4.7 seconds before liftoff (X-0). After detecting the rise of the LE-7A combustion pressure, two SRB-As are ignited (at approximately 0.5 second before X-0), subsequently the H-IIA launch vehicle lifts off from the Movable Launcher (ML) , that is X-0. The SRB-As burn out at approximately 99 seconds after liftoff (hereafter called X+99), and then separated from the core stage.

The payload fairing (PLF) is separated after a free molecular heat flux becomes less than 1135 W/m^2, resulting the separation timing of the PLF being different in each mission. At approximately X+396, the first stage engine (LE-7A) shutdown occurs. After LE-7A thrust has tailed off, the first stage separation is executed.

The attitude control is conducted by the first stage engine gimbaling, the SRB-A nozzles gimbaling, and auxiliary thrusters. The gimbaling of the first stage engine (LE-7A) contributes to the pitch / yaw control throughout the booster and the first stage phase. The SRB-A engine nozzle is gimbaled for the pitch / yaw / roll control. Roll axis control is executed by two auxiliary thrusters after the SRB-As burn out.

2.1.1.2 Second stage phase

After the first stage is separated, the second stage engine (LE-5B) is ignited (SEIG1). The LE-5B engine burns for approximately 340 seconds. Just after the second stage (including the payload) is injected to the parking orbit, the engine shuts down once (SECO1). During engine burning, the pitch/yaw control is conducted by the gimbaling of the LE-5B and the roll control is conducted by the reaction control system (RCS). After the engine cutoff, the vehicle starts the coasting flight.

During the coast phase, preparations for the second stage engine restart take place. These are propellant settling, pressure control of LOX / LH_2 tanks, engine chilling down, attitude maneuver and so on. In this phase, the attitude control of the pitch/yaw/roll is performed only by the RCS.

Approximately X+1460 seconds, the LE-5B is restarted (SEIG2) and when the second stage reaches the planned transfer orbit, the engine cutoff occurs again (SECO2). The second burning duration is approximately 200 seconds in a standard GTO mission. Mission time of standard GTO is approximately 0.5 hours.

After spacecraft / second stage separation, the contamination and collision avoidance maneuver (CCAM) is conducted using the RCS and the residual propellant ejection.

In case of long-coast GTO mission, the second stage conducts over 10,000 sec coast after SECO2, and then the LE-5B makes the third burn in order to increase the perigee altitude, and to reduce the inclination, as shown in Table 2.1-2. Mission time of long-coast GTO is approximately 4.5 hours.

Table 2.1-1 Summary of H-IIA launch capability

Mission (Orbit)	Payload Mass		Note
	H2A202	H2A204	
Standard GTO ha = 35,976 km(*1) hp = 250 km i = 28.5 deg ω = 179.0 deg	4,000 kg (8,820 lb) (Model 4S fairing)	5,950 kg (13,100 lb) (Model 4S fairing)	Osculating orbit at Spacecraft separation *1 : Including Kepler bias (190km) delta-V=1,830m/s case
Long-coast GTO ha = 35,586 km hp = 2,700 km i = 20.1 deg ω = 179.0 deg	2,970 kg (6,550 lb) (Model 4S fairing)	4,820 kg (10,600 lb) (Model 4S fairing)	Osculating orbit at Spacecraft separation delta-V=1,500m/s case
SSO h = 800 km i = 98.6 deg	3,300 kg (7,280 lb) (Model 4S fairing)	—	
SSO h = 500 km i = 97.4 deg	5,100 kg (11,200 lb) (Model 4S fairing)	—	

Launch Capability shows with the condition of PCS=99%.
Regarding detailed launch capabilities, please contact LSP.

<Abbreviation>

GTO : Geostationary Transfer Orbit
SSO : Sun Synchronous Orbit
ha : apogee altitude
hp : perigee altitude
h : altitude
i : inclination
ω : argument of perigee

<Remarks>

The radius of the Earth is assumed to be 6378 km.

Table 2.1-2 Typical sequence of events for GTO mission

(1) Standard GTO mission (two burns of second stage)

Events	H2A202 [s]	H2A204 [s]	Remarks
Guidance flight mode on	-18	-18	
LE-7A ignition	-4.7	-4.7	
SRB-A (2 for 202, 4 for 204) ignition	-0.6	-0.6	
Liftoff	0.0	0.0	
SRB-A burn out	99	117	
SRB-A separation	106	first 128 / second 132	First means first pair of SRB-As separation time.
Fairing separation	245	245	
Main engine cutoff (MECO)	396	396	
First stage separation	404	404	
Second stage ignition 1 (SEIG1)	410	410	
Second stage cutoff 1 (SECO1)	753	705	
Second stage ignition 2 (SEIG2)	1463	1361	
Second stage cutoff 2 (SECO2)	1659	1604	
Spacecraft separation	1706	1650	

(2) Long-coast GTO mission (three burns of second stage)

Events	H2A202 [s]	H2A204 [s]	Remarks
Guidance flight mode on	-18	-18	
LE-7A ignition	-4.7	-4.7	
SRB-A (2 for 202, 4 for 204) ignition	-0.6	-0.6	
Liftoff	0.0	0.0	
SRB-A burn out	99	117	
SRB-A separation	106	first 128 / second 132	First means first pair of SRB-As separation time.
Fairing separation	205	205	
Main engine cutoff (MECO)	396	396	
First stage separation	404	404	
Second stage ignition 1 (SEIG1)	410	410	
Second stage cutoff 1 (SECO1)	711	657	
Second stage ignition 2 (SEIG2)	1470	1362	
Second stage cutoff 2 (SECO2)	1652	1593	
Second stage ignition 3 (SEIG3)	15915	15765	
Second stage cutoff 3 (SECO3)	15946	15805	
Spacecraft separation	15997	15855	

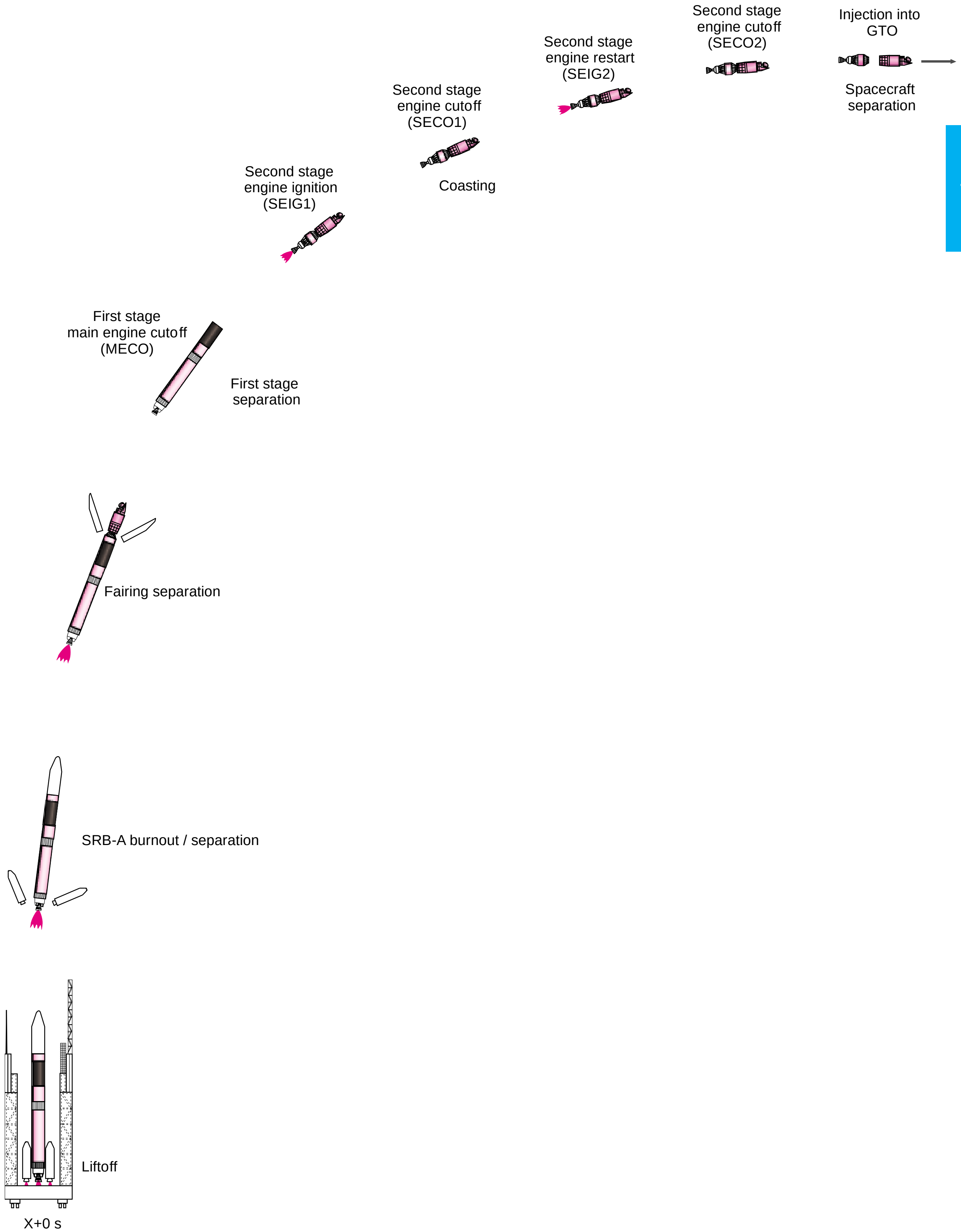


Figure 2.1-1 Typical standard GTO flight sequence for H2A202

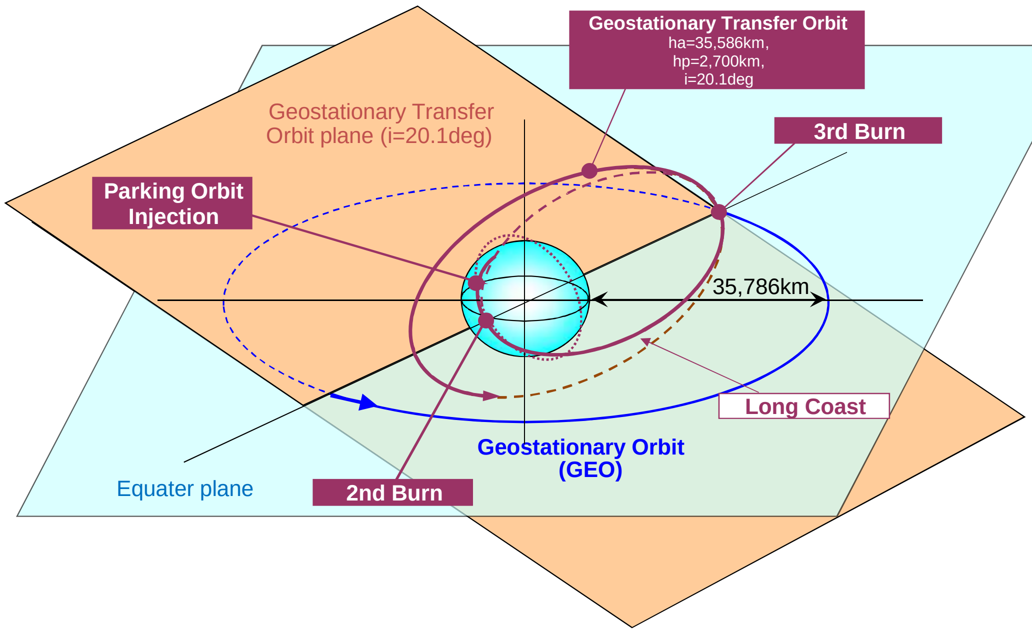


Figure 2.1-2 Typical long-coast GTO flight sequence

2.2 Launch performance definition

2.2.1 Payload mass

Payload mass is referred to as launch capability in this manual. Payload mass is defined as the total mass of the spacecraft injected into the target orbit. PLA is a part of H-IIA, therefore “Payload Mass” does not include PLA mass.

2.2.2 Launch vehicle configurations

Unless otherwise noted, model 4S fairing is used for H-IIA performance presented in this manual. Other types of fairings are also available, such as model 5S or 4/4D-LC for dual-launch or greater volume than model 4S fairing. However, performance will degrade due to the difference of fairing mass and aerodynamic drag as follows:

< Model 5S fairing >

Approximately 0.3 metric-tons less than the performance with model 4S fairing.

< Model 4/4D-LC fairing >

The sum of two satellites should be approximately 1.1 metric-tons less than the performance with model 4S fairing. For example, if the performance with model 4S fairing is 5.1 metric-ton, two satellites weighting 2.0 metric-tons each can be injected into the same orbit.

2.2.3 Launch vehicle performance

The H-IIA launch vehicle performance is presented with 99% probability of second stage command shut down.

2.3 Geostationary Transfer Orbit (GTO) Mission

2.3.1 Launch capability for dedicated launch

(1) Standard GTO mission

Payload mass for standard GTO dedicated-launch mission based on typical orbital parameters in Section 2.3.3(1) are as follows.

For H2A202 (delta-V to GEO =1830m/s),
- -Model 4S fairing: 4,000 kg
- -Model 5S fairing: 3,700 kg

For H2A204 (delta-V to GEO =1830m/s),
- -Model 4S fairing: 5,950 kg
- -Model 5S fairing: 5,700 kg

Payload mass based on other orbit parameters are shown as below.
- -H2A202 and H2A204 with model 4S fairing: Figure 2.3-1
- -H2A202 and H2A204 with model 5S fairing: Figure 2.3-2

(2) Long-coast GTO mission

Payload mass for long-coast GTO dedicated-launch mission based on typical orbital parameters in Section 2.3.3(2) are as follows.

For H2A202 (delta-V to GEO =1500m/s),
- -Model 4S fairing: 2,970 kg
- -Model 5S fairing: 2,670 kg

For H2A204 (delta-V to GEO =1500m/s),
- -Model 4S fairing: 4,820 kg
- -Model 5S fairing: 4,520 kg

Payload mass based on other orbit parameters are shown as below.
- -H2A202 with model 4S fairing: Figure 2.3-3
- -H2A202 with model 5S fairing: Figure 2.3-4
- -H2A204 with model 4S fairing: Figure 2.3-5
- -H2A204 with model 5S fairing: Figure 2.3-6

2.3.2 Launch capability for dual (GTO and GTO) launch

The combined mass of two spacecrafts with model 4/4D-LC fairing for GTO dual-launch mission is 1,100kg*1 less than the payload mass with 4S fairing in Section 2.3.1.

*1: Lower fairing mass which is separated after the upper spacecraft separation

2.3.3 Typical orbital parameters

(1) Standard GTO mission

Typical orbital parameters for standard GTO mission are as follows :

Apogee altitude	ha	=	35,976 km	*1
Perigee altitude	hp	=	250 km	*2
Inclination	i	=	28.5 deg	
Argument of perigee	ω	=	179 deg	*2

Assuming the Earth's equator radius is 6,378 km and these values are osculating orbit parameters at spacecraft separation.

*1 : This value includes Kepler bias (190 km).

*2 : Perigee altitude from 250 km to 400 km is designed depending on SC inclination requirement to ensure the RF link at SC separation between ground station and LV. Argument of perigee is also similar condition to perigee altitude. These values are coordinated with SC mission requirement and mission trajectory design at technical interchange meeting. For more detail information, please contact LSP.

(2) Long-coast GTO mission

Typical orbital parameters for long-coast GTO mission are as follows :

Apogee altitude	ha	=	35,586 km
Perigee altitude	hp	=	2,700 km
Inclination	i	=	20.1 deg
Argument of perigee	ω	=	179 deg

2.3.4 Injection accuracies

(1) Standard GTO mission

Typical injection accuracies for GTO standard mission are as follows based on the orbital parameters in Section 2.3.3(1).

Apogee altitude	ha	=	± 180 km
Perigee altitude	hp	=	± 4 km
Inclination	i	=	± 0.02 deg
Argument of perigee	ω	=	± 0.4 deg
Longitude of ascending node	Ω	=	± 0.4 deg

(These values are 3-sigma level.)

(2) Long-coast GTO mission

Typical injection accuracies for long-coast GTO mission are as follows based on the orbital parameters in Section 2.3.3(2).

Apogee altitude	ha	=	± 180 km
Perigee altitude	hp	=	± 60 km
Inclination	i	=	± 0.2 deg
Argument of perigee	ω	=	± 0.6 deg
Longitude of ascending node	Ω	=	± 0.6 deg

2.3.5 Typical sequence of events

Typical sequence of events of H2A202 and H2A204 for standard GTO mission and long-coast GTO mission is shown in Table 2.1-2.

2.3.6 Typical trajectory

Typical flight trajectory and parameters for GTO mission are shown below.

- -Standard GTO mission (H2A202) : Figure 2.3-7～2.3-10
- -Standard GTO mission (H2A204) : Figure 2.3-11～2.3-14
- -Long-coast GTO mission (H2A202) : Figure 2.3-15～2.3-20
- -Long-coast GTO mission (H2A204) : Figure 2.3-21～2.3-26

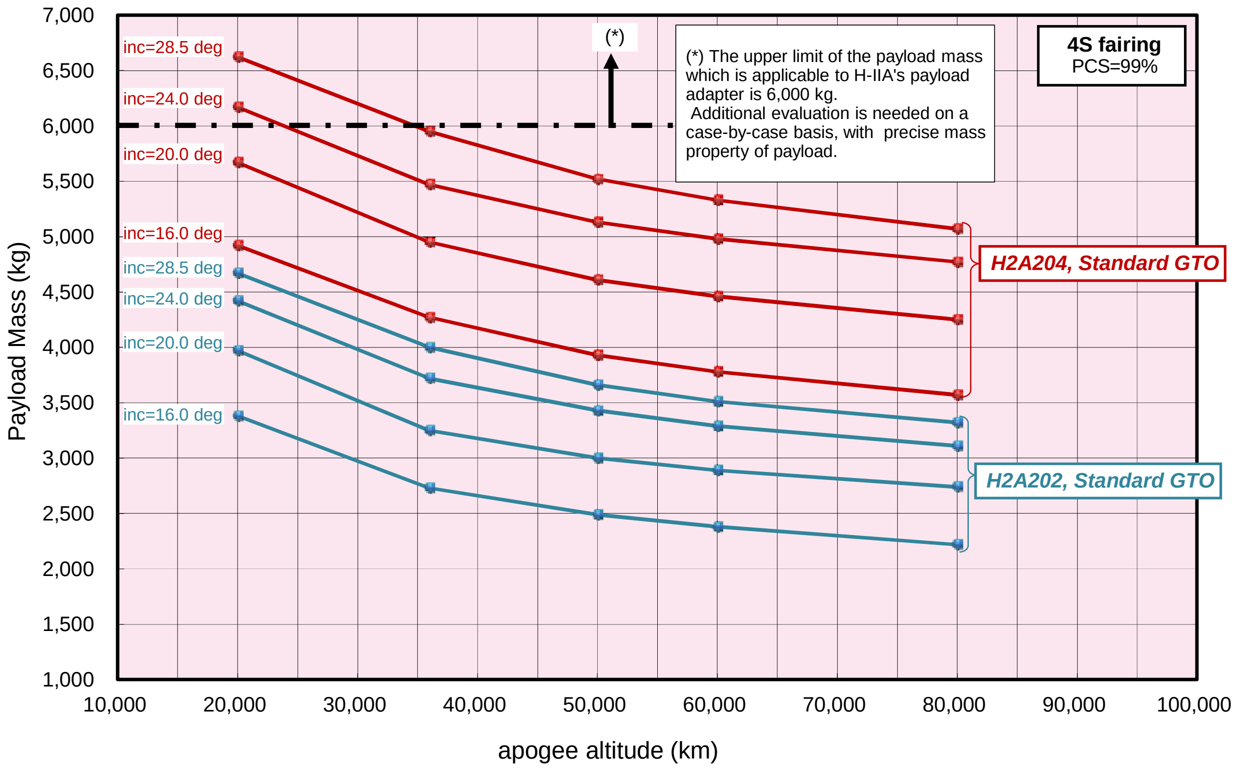


Figure 2.3-1 Standard GTO mission Launch Capabilities (H2A202 and H2A204 with model 4S Fairing)

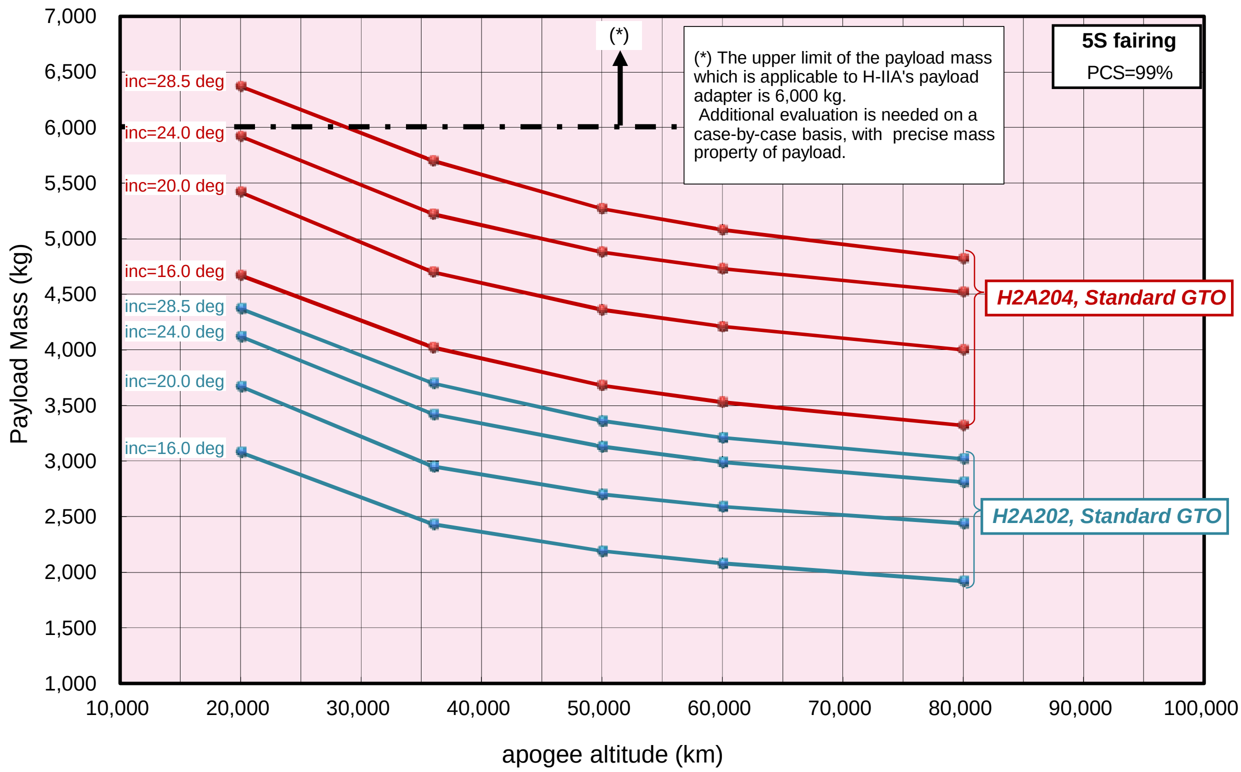


Figure 2.3-2 Standard GTO mission Launch Capabilities (H2A202 and H2A204 with model 5S Fairing)

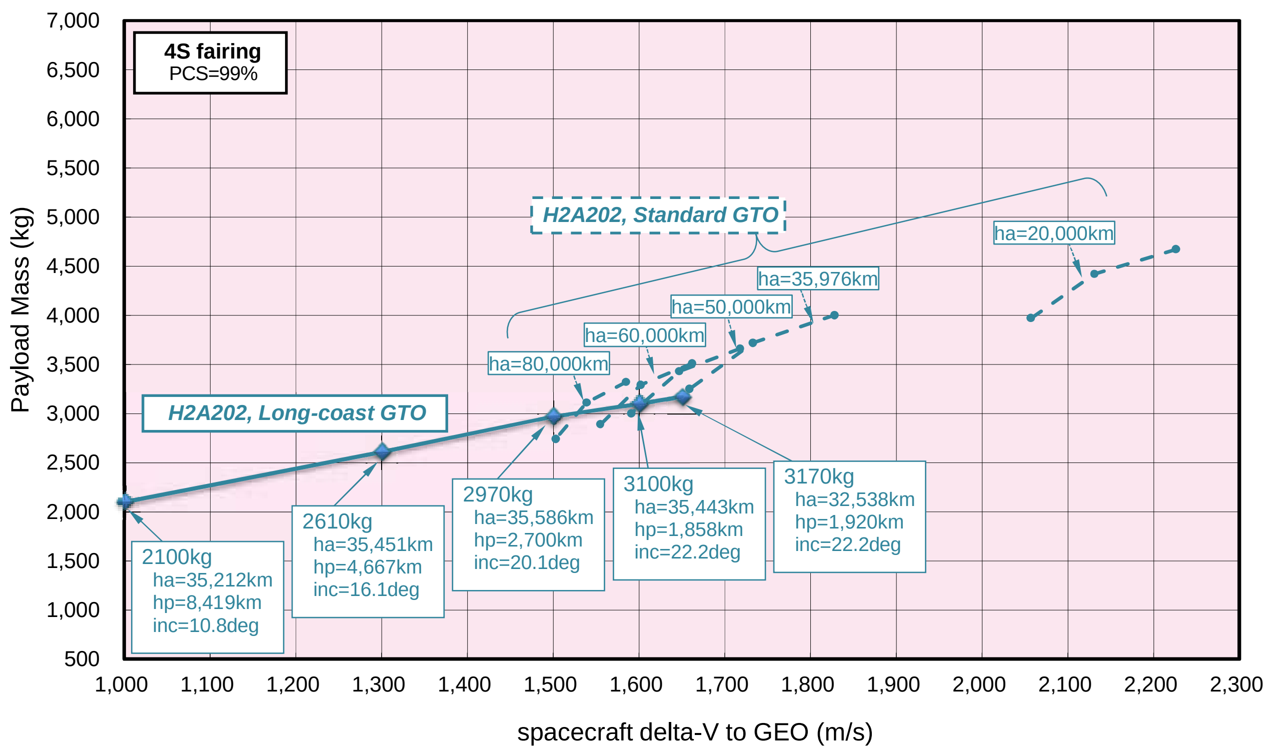


Figure 2.3-3 Long-coast GTO mission Launch Capabilities (H2A202 with model 4S Fairing)

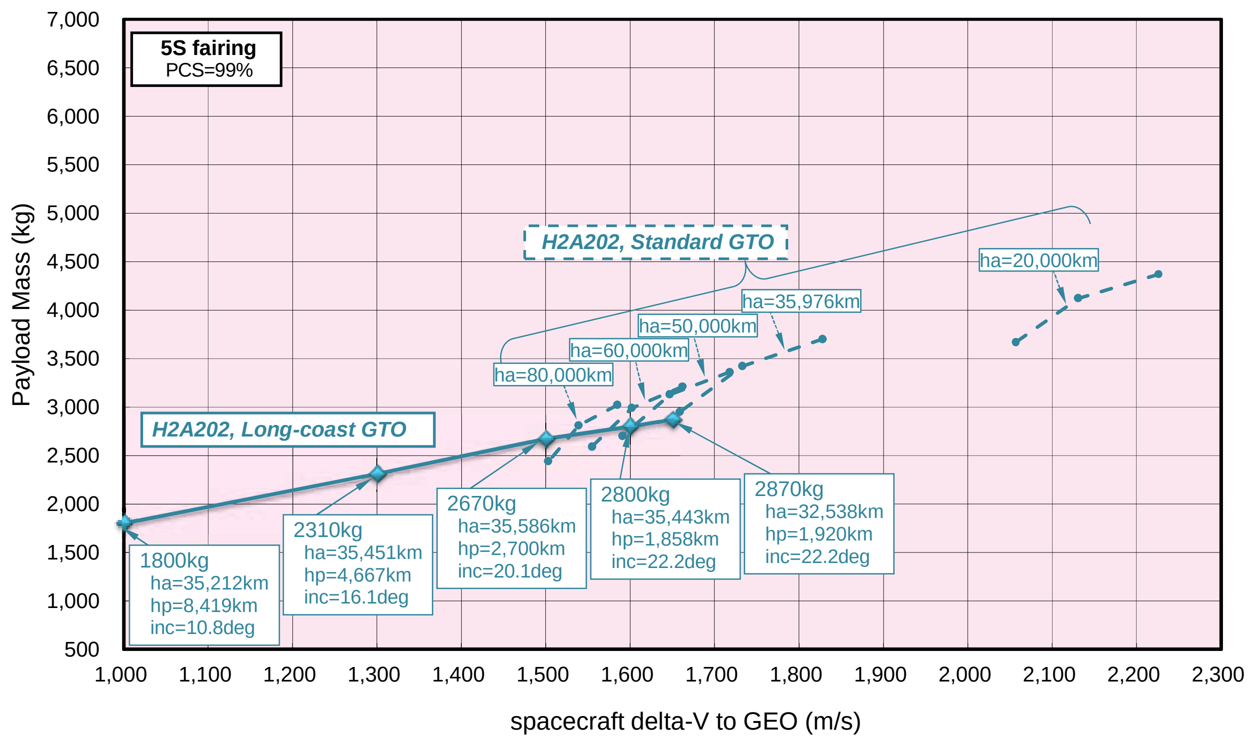


Figure 2.3-4 Long-coast GTO mission Launch Capabilities (H2A202 with model 5S Fairing)

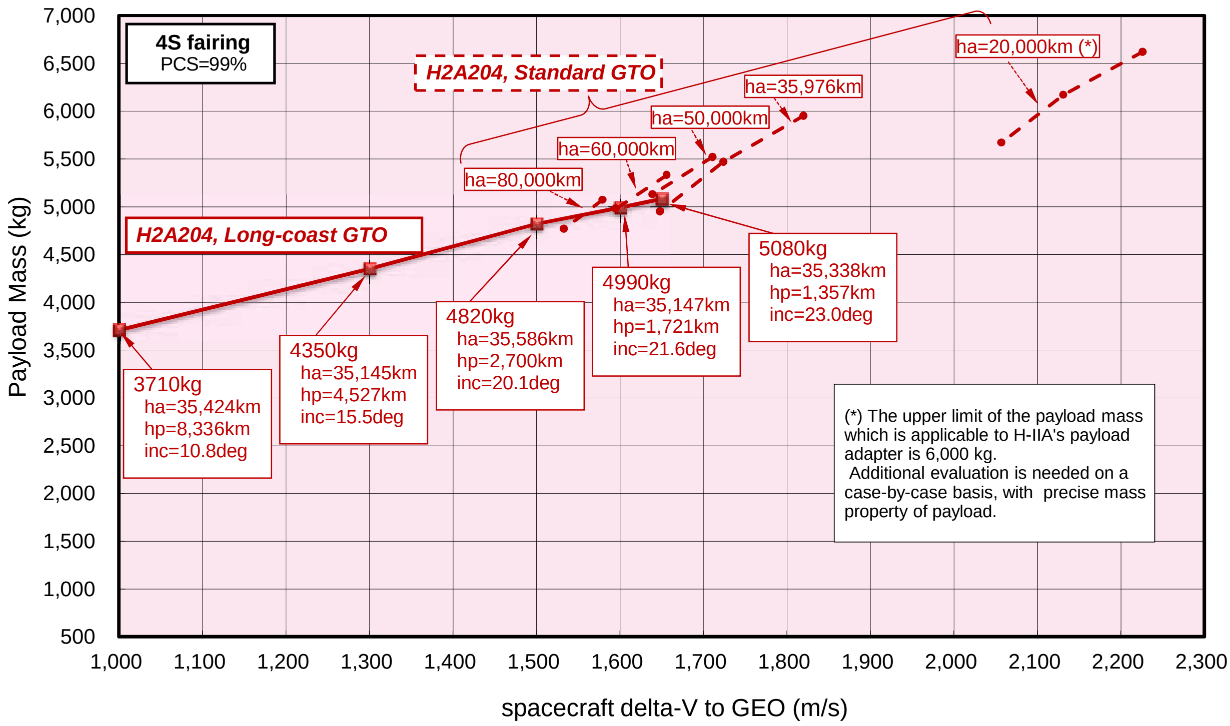


Figure 2.3-5 Long-coast GTO mission Launch Capabilities (H2A204 with model 4S Fairing)

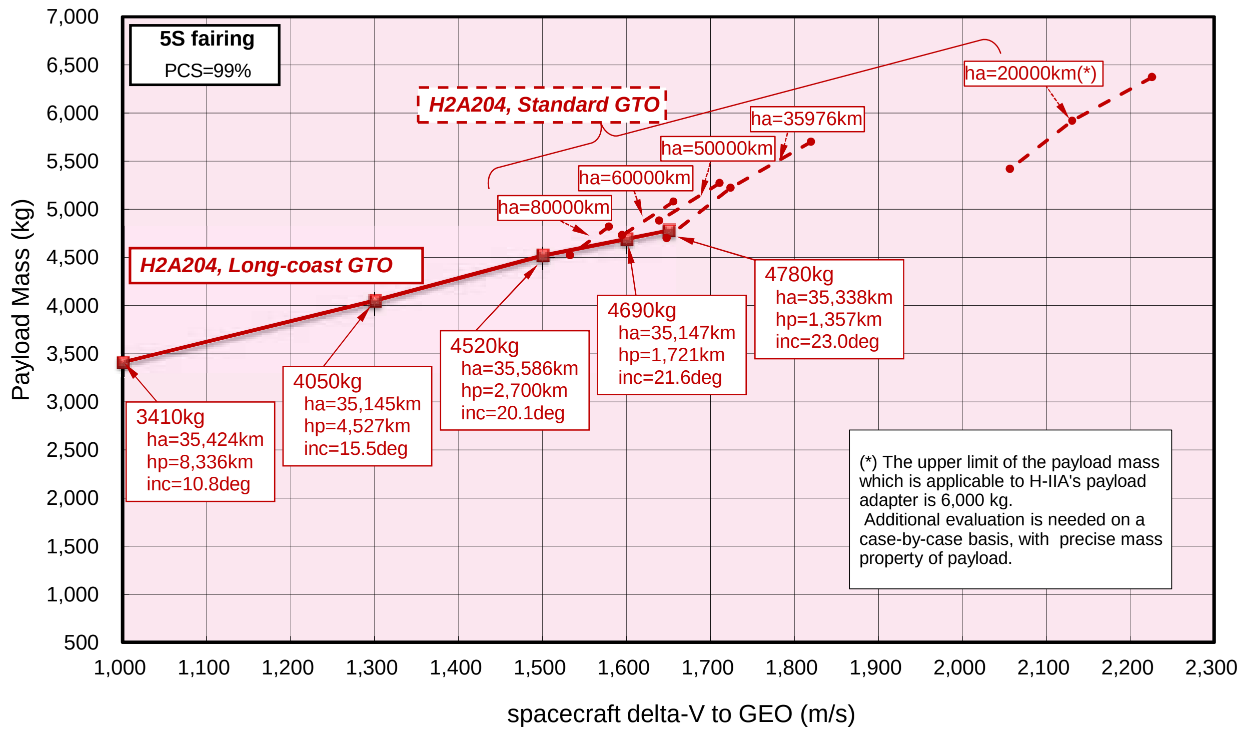


Figure 2.3-6 Long-coast GTO mission Launch Capabilities (H2A204 with model 5S Fairing)

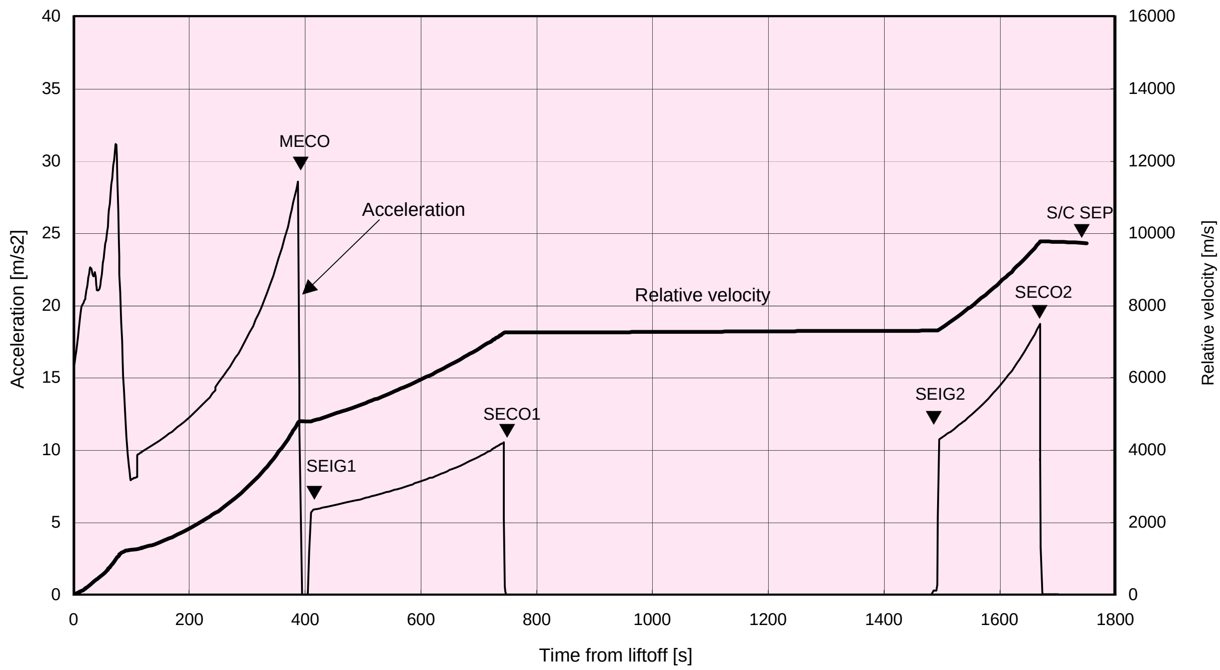


Figure 2.3-7　Typical flight parameters for Standard GTO mission (H2A202) Acceleration and Relative Velocity

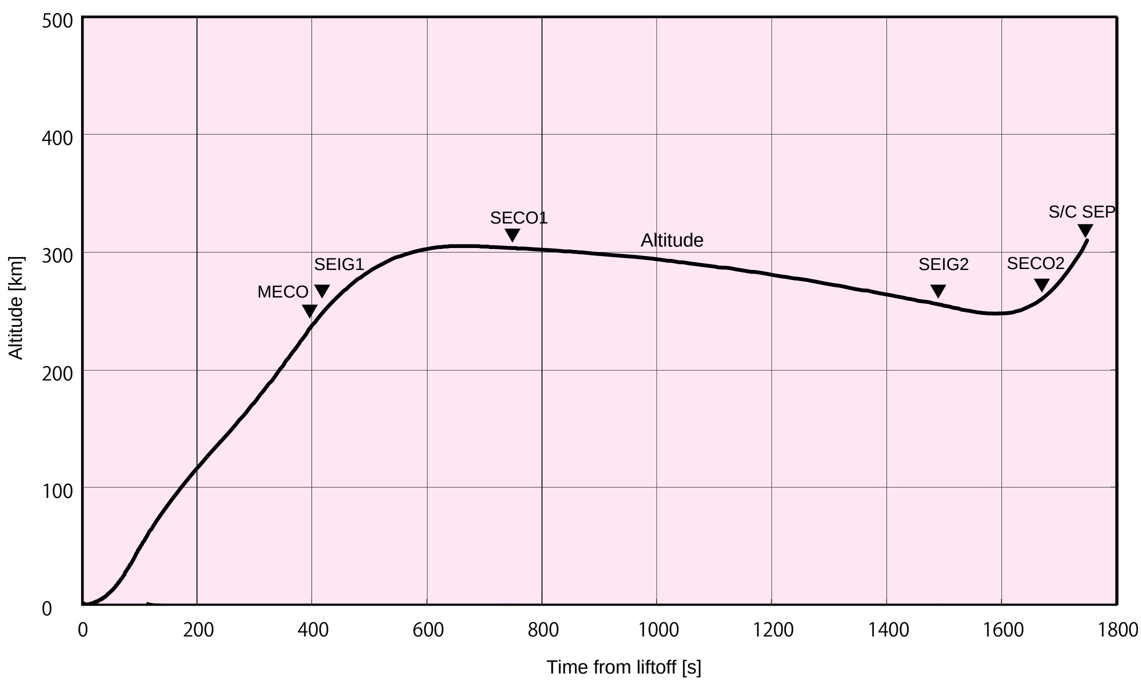


Figure 2.3-8　Typical flight parameter for Standard GTO mission (H2A202) Altitude

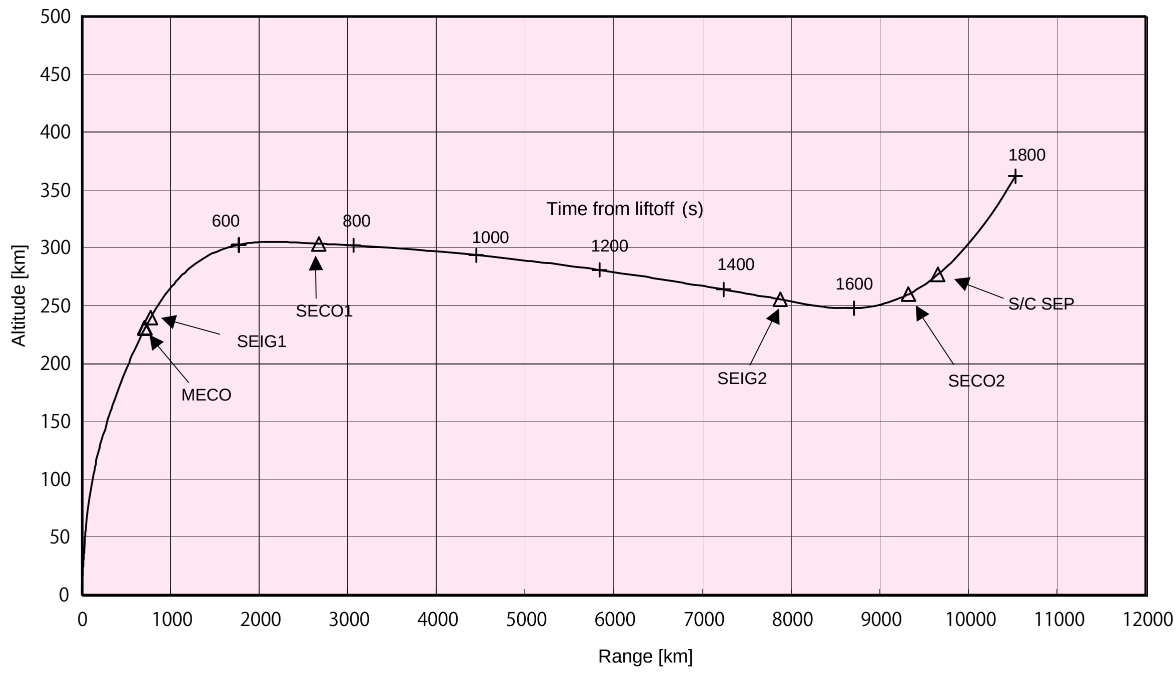


Figure 2.3-9 Typical flight trajectory for Standard GTO mission (H2A202) Altitude ～ Range

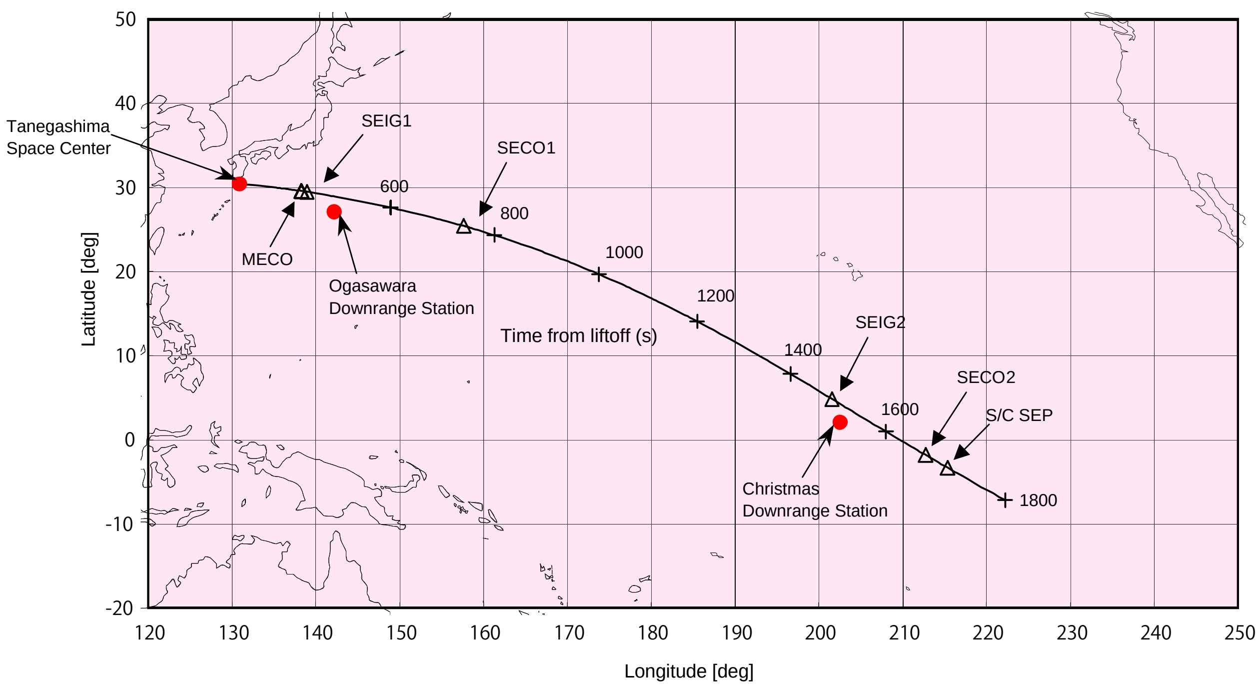


Figure 2.3-10 Typical flight trajectory for Standard GTO mission (H2A202) Latitude ～ Longitude

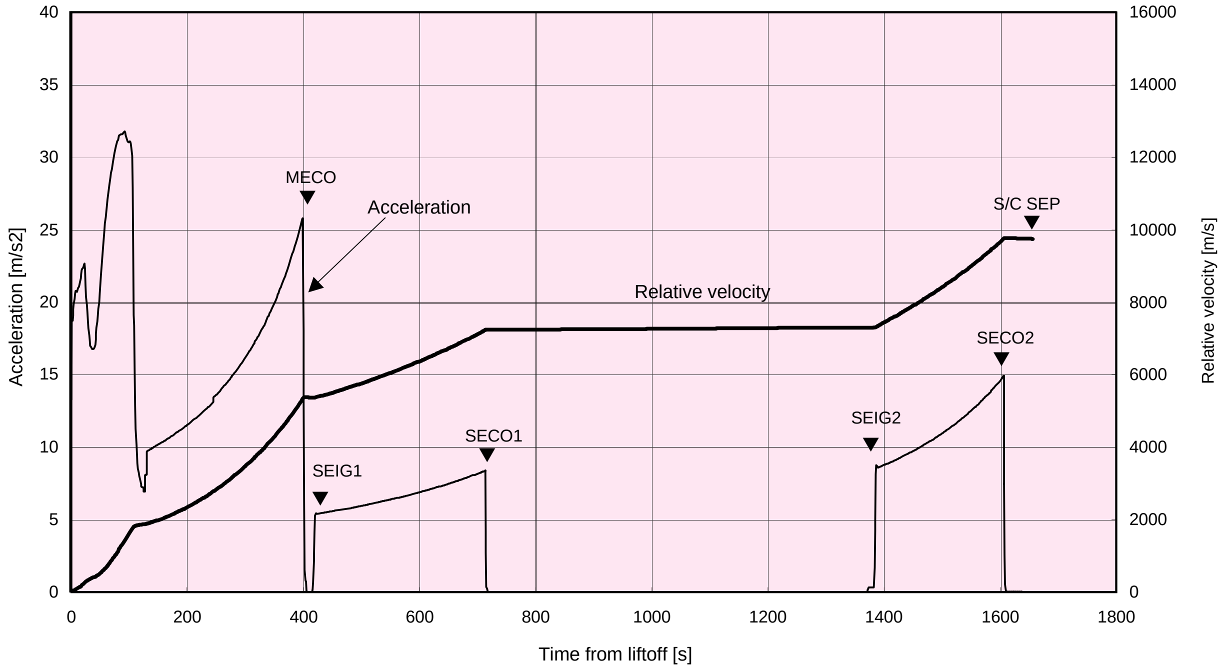


Figure 2.3-11 Typical flight parameters for Standard GTO mission (H2A204) Acceleration and Relative velocity

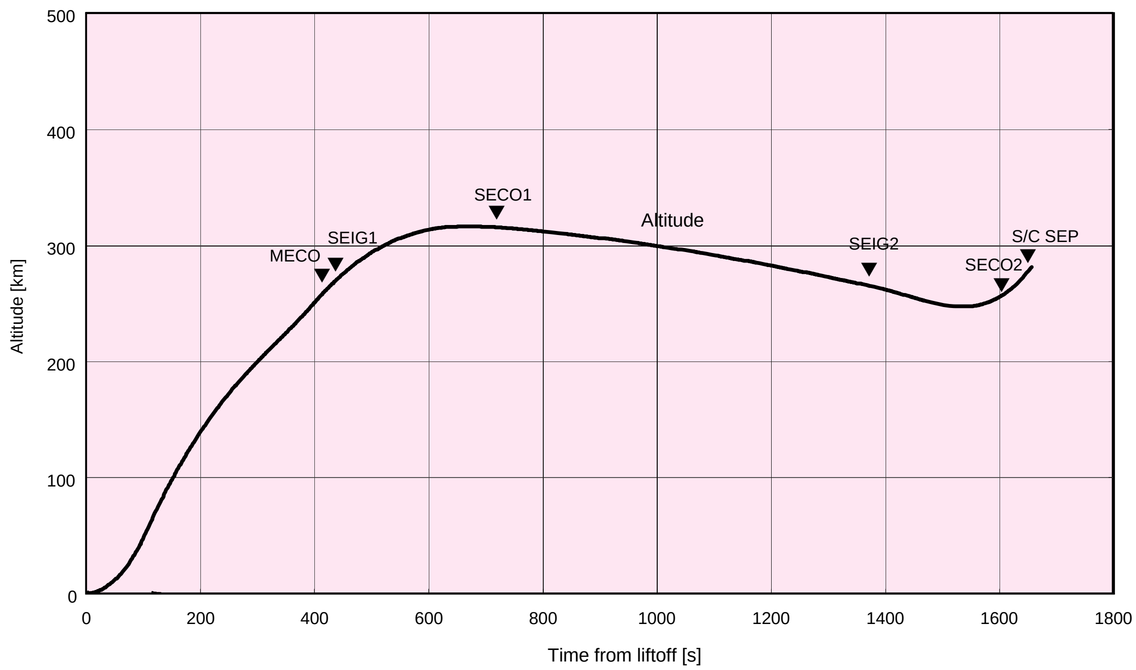


Figure 2.3-12 Typical flight parameter for Standard GTO mission (H2A204) Altitude

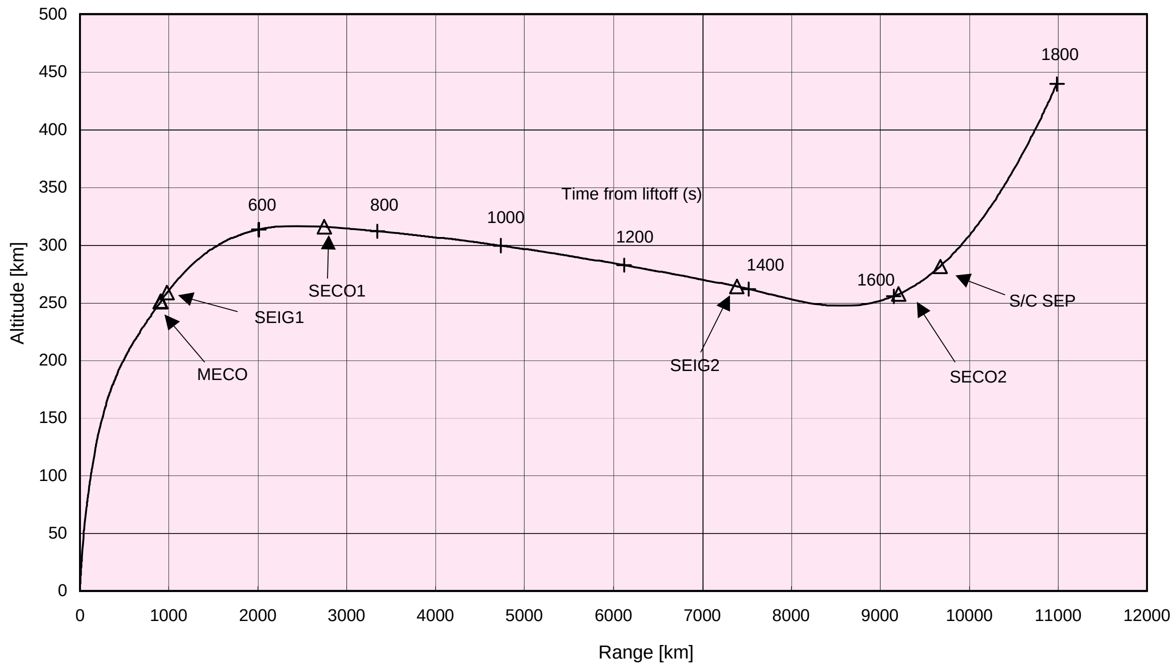


Figure 2.3-13 Typical flight trajectory for Standard GTO mission (H2A204)
Altitude ～ Range

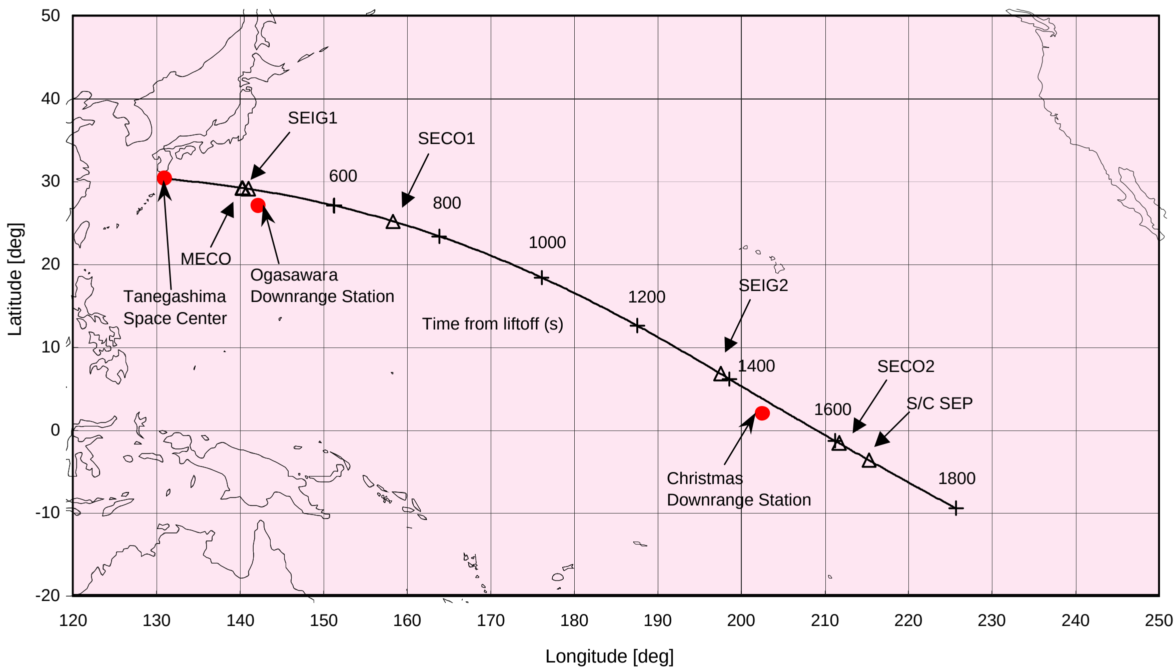


Figure 2.3-14 Typical flight trajectory for Standard GTO mission (H2A204)
Latitude ～ Longitude

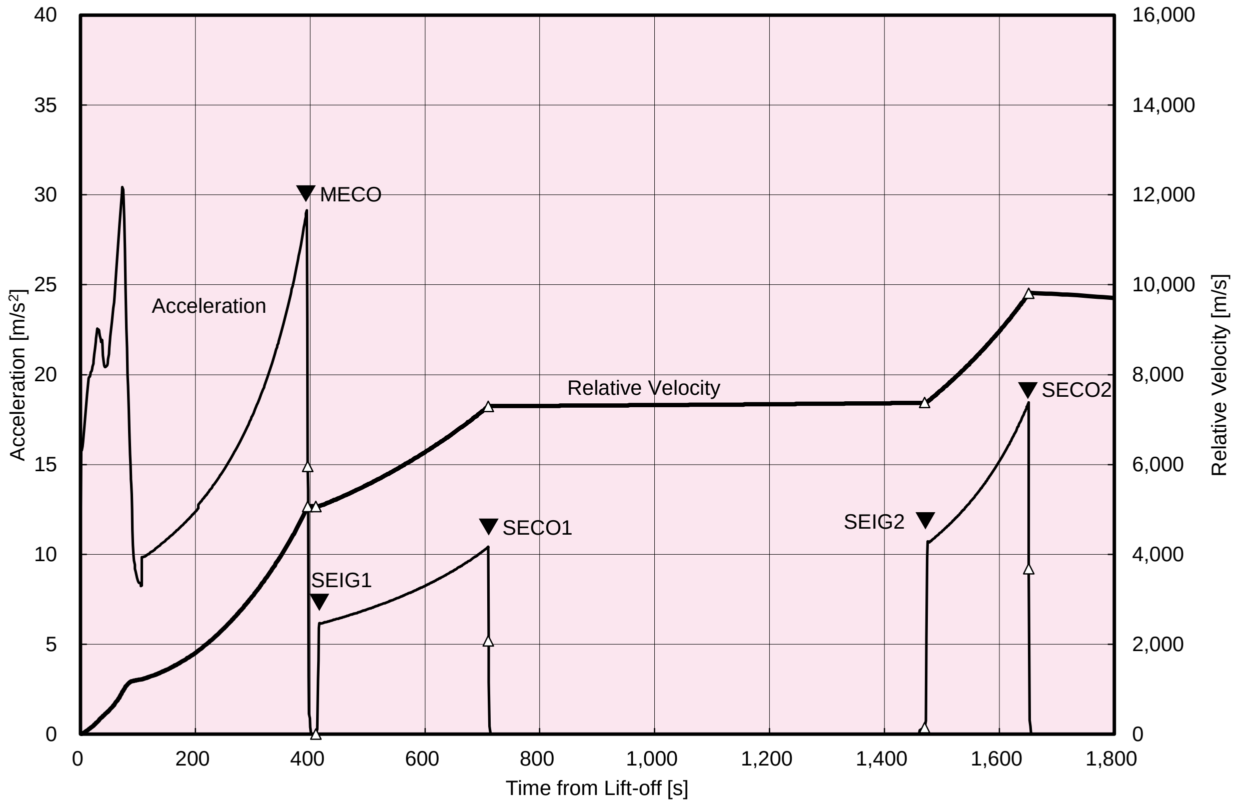


Figure 2.3-15 Typical flight parameters for Long-coast GTO mission (H2A202) Acceleration and Relative velocity (Lift-off~SECO2)

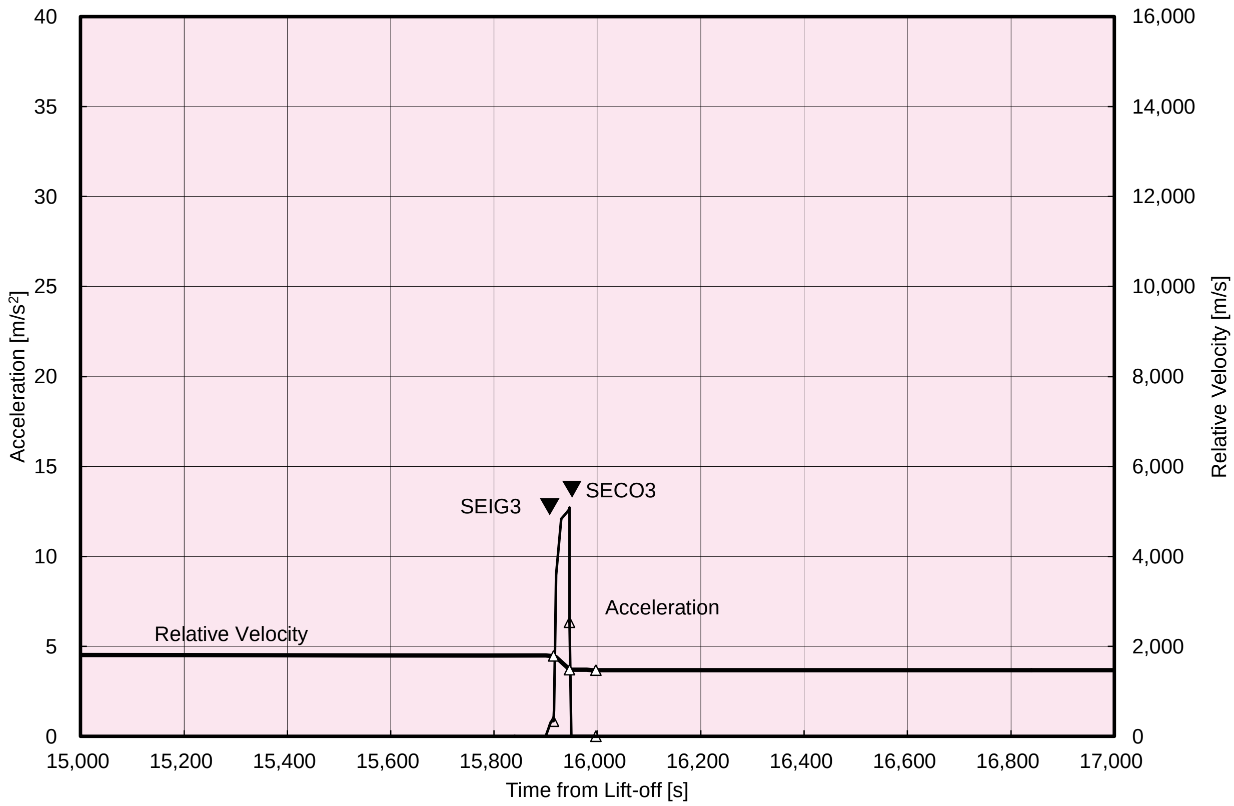


Figure 2.3-16 Typical flight parameters for Long-coast GTO mission (H2A202) Acceleration and Relative velocity (~3rd Burn)

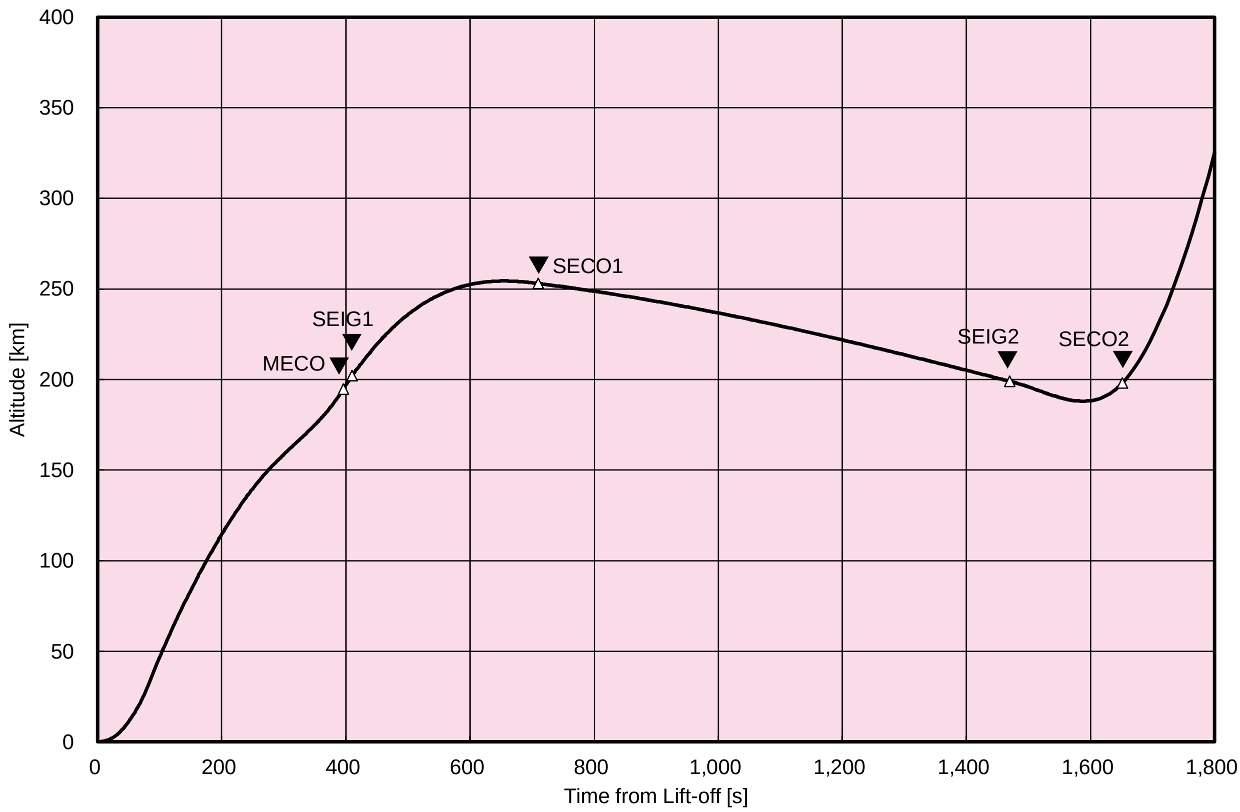


Figure 2.3-17 Typical flight parameters for Long-coast GTO mission (H2A202) Altitude (Lift-off~SECO2)

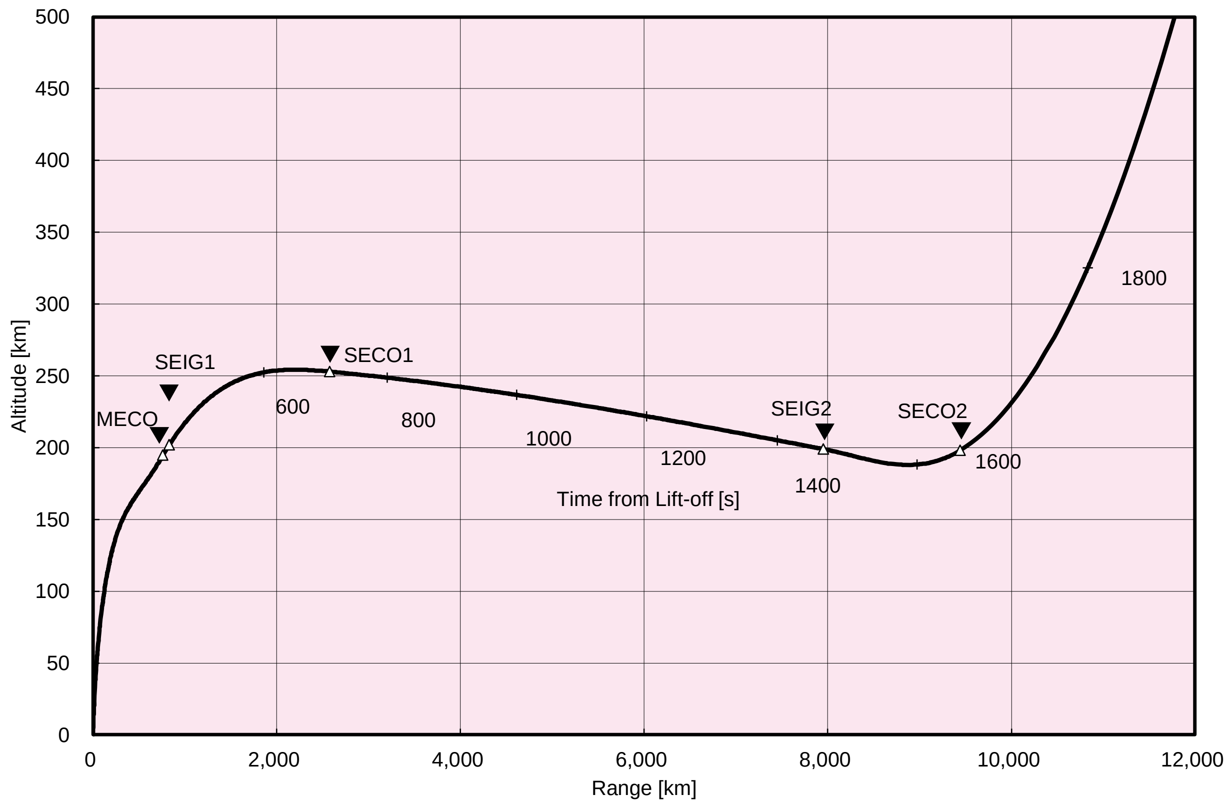


Figure 2.3-18 Typical flight trajectory for Long-coast GTO mission (H2A202) Altitude ～ Range (Lift-off~SECO2)

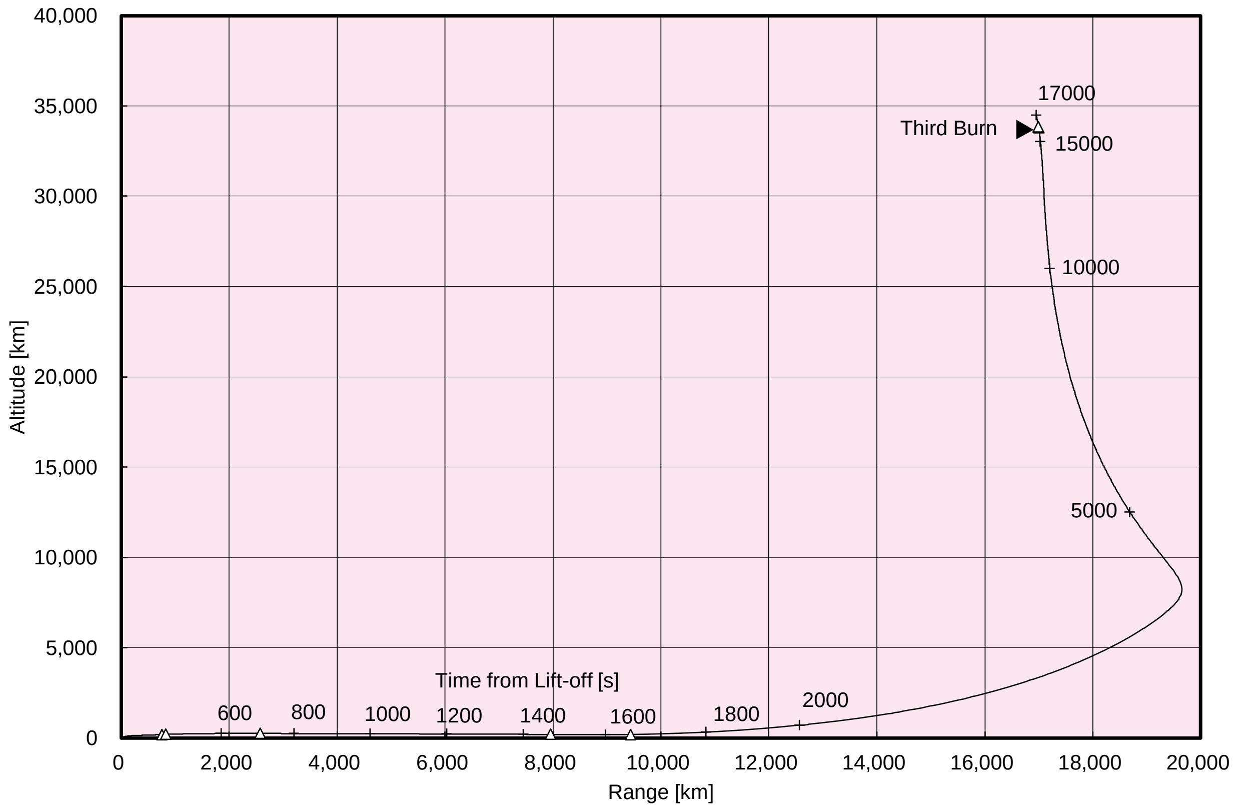


Figure 2.3-19 Typical flight trajectory for Long-coast GTO mission (H2A202) Altitude ～ Range (Lift-off~SECO3)

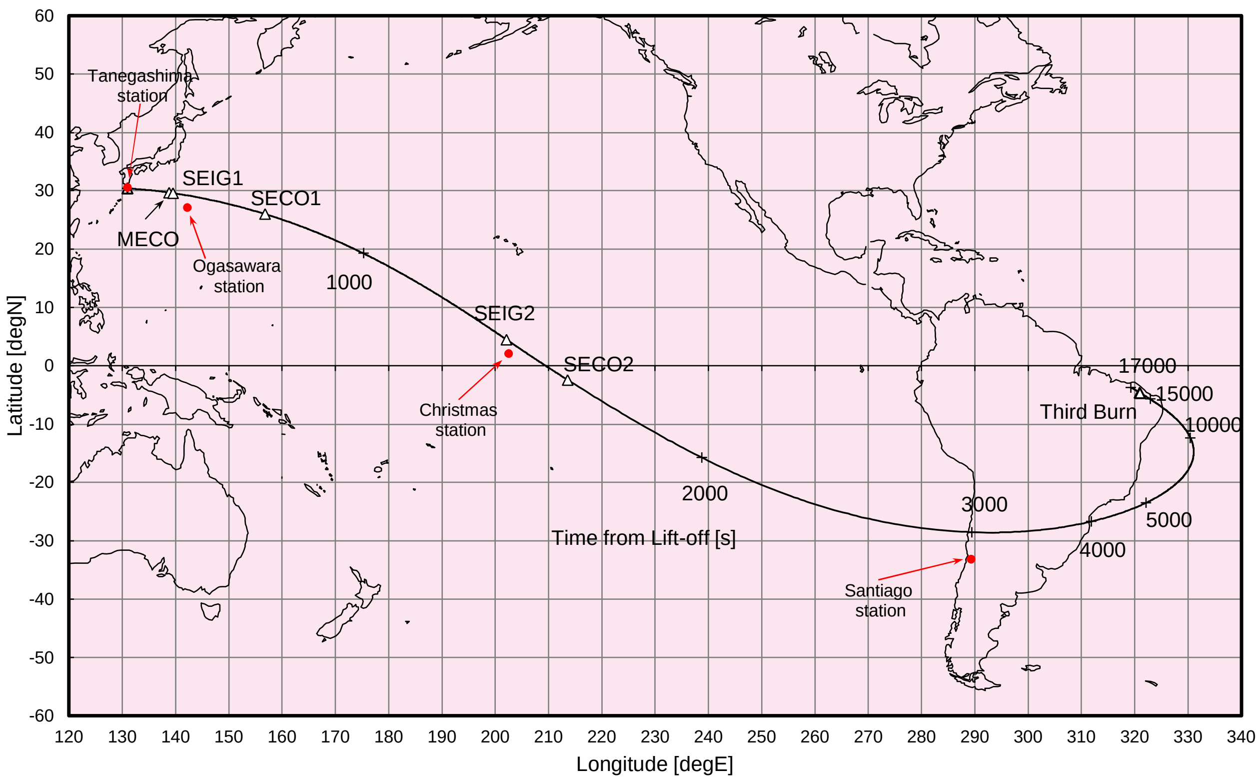


Figure 2.3-20 Typical flight trajectory for Long-coast GTO mission (H2A202) Latitude ～ Longitude

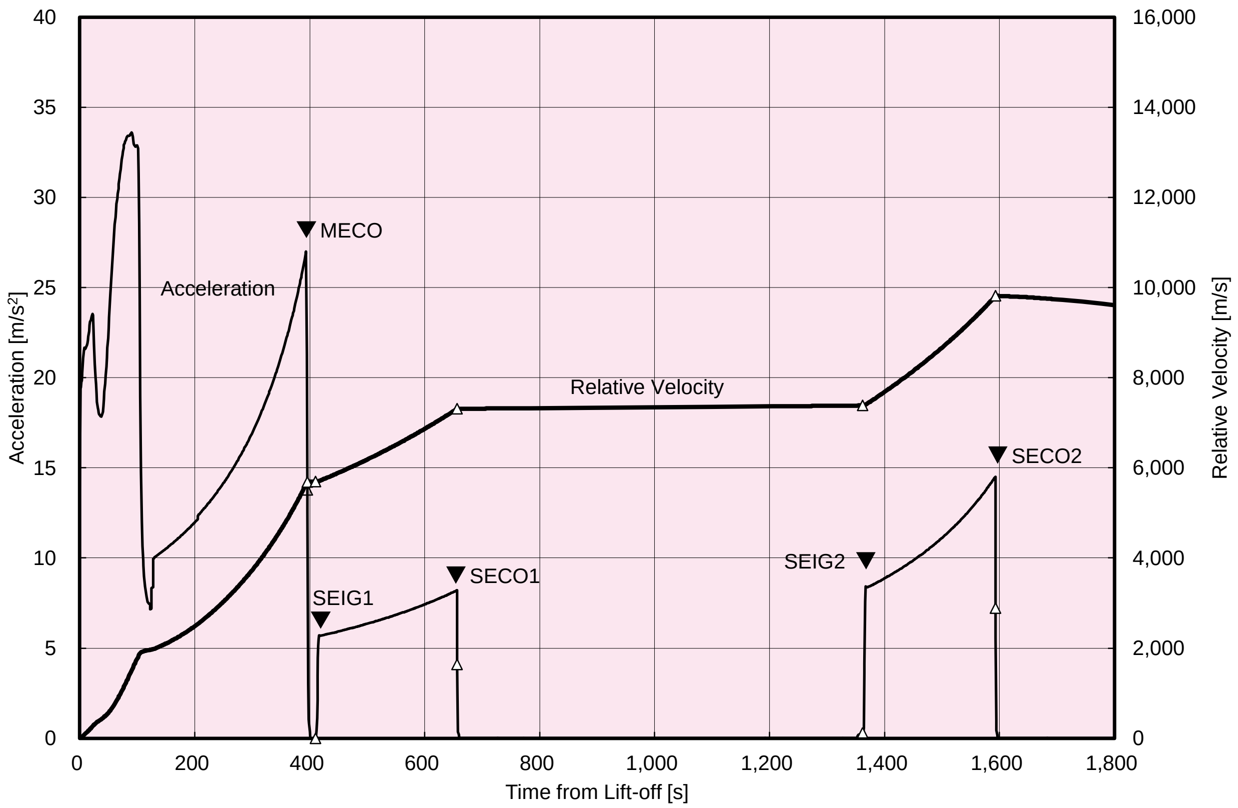


Figure 2.3-21 Typical flight parameters for Long-coast GTO mission (H2A204) Acceleration and Relative velocity (Lift-off~SECO2)

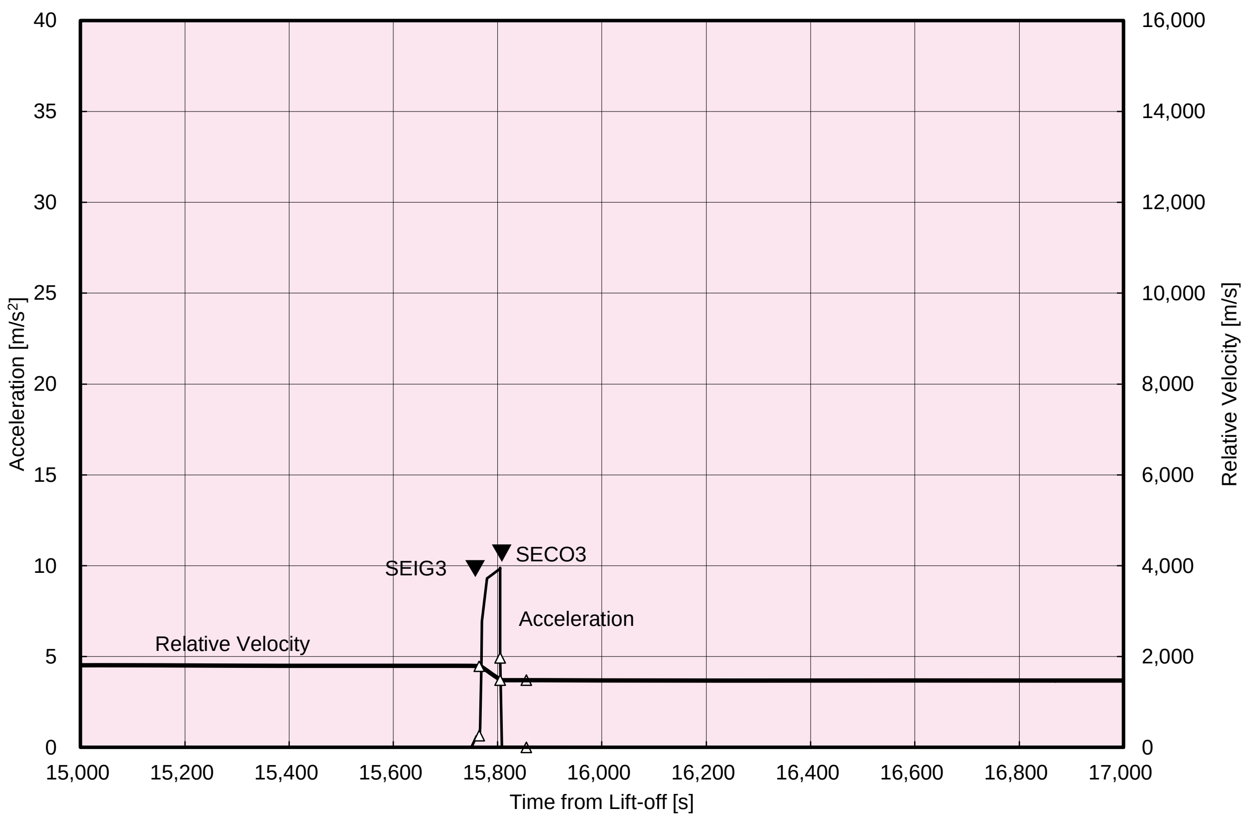


Figure 2.3-22 Typical flight parameters for Long-coast GTO mission (H2A204) Acceleration and Relative velocity (~3rd Burn)

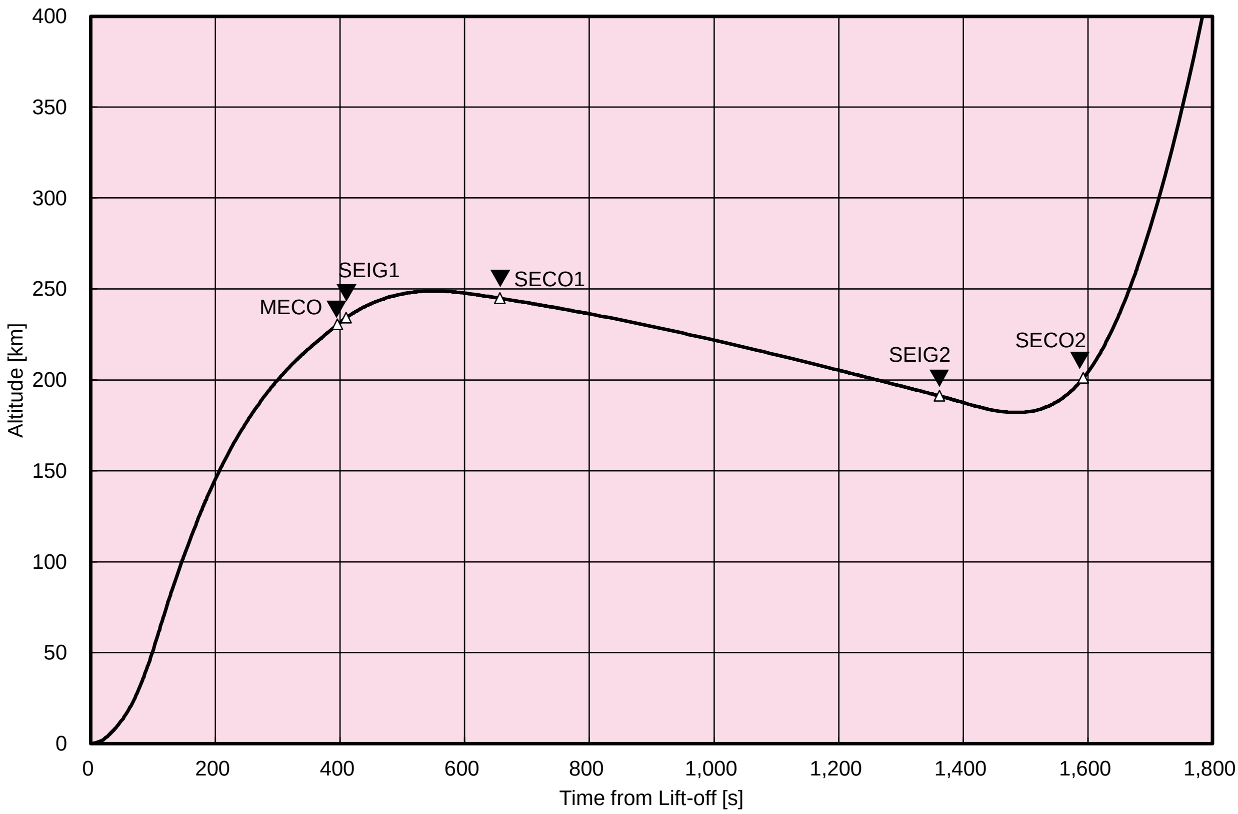


Figure 2.3-23 Typical flight parameters for Long-coast GTO mission (H2A204) Altitude (Lift-off~SECO2)

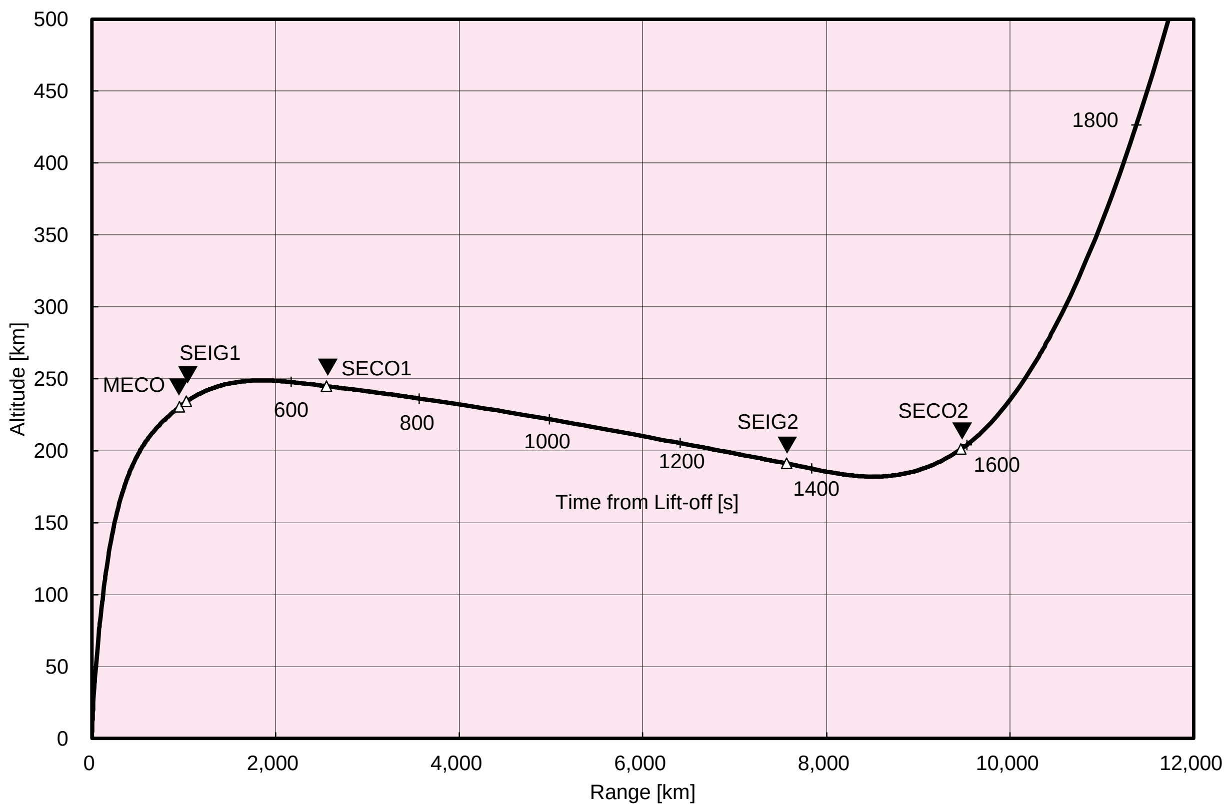


Figure 2.3-24 Typical flight trajectory for Long-coast GTO mission (H2A204) Altitude ～ Range (Lift-off~SECO2)

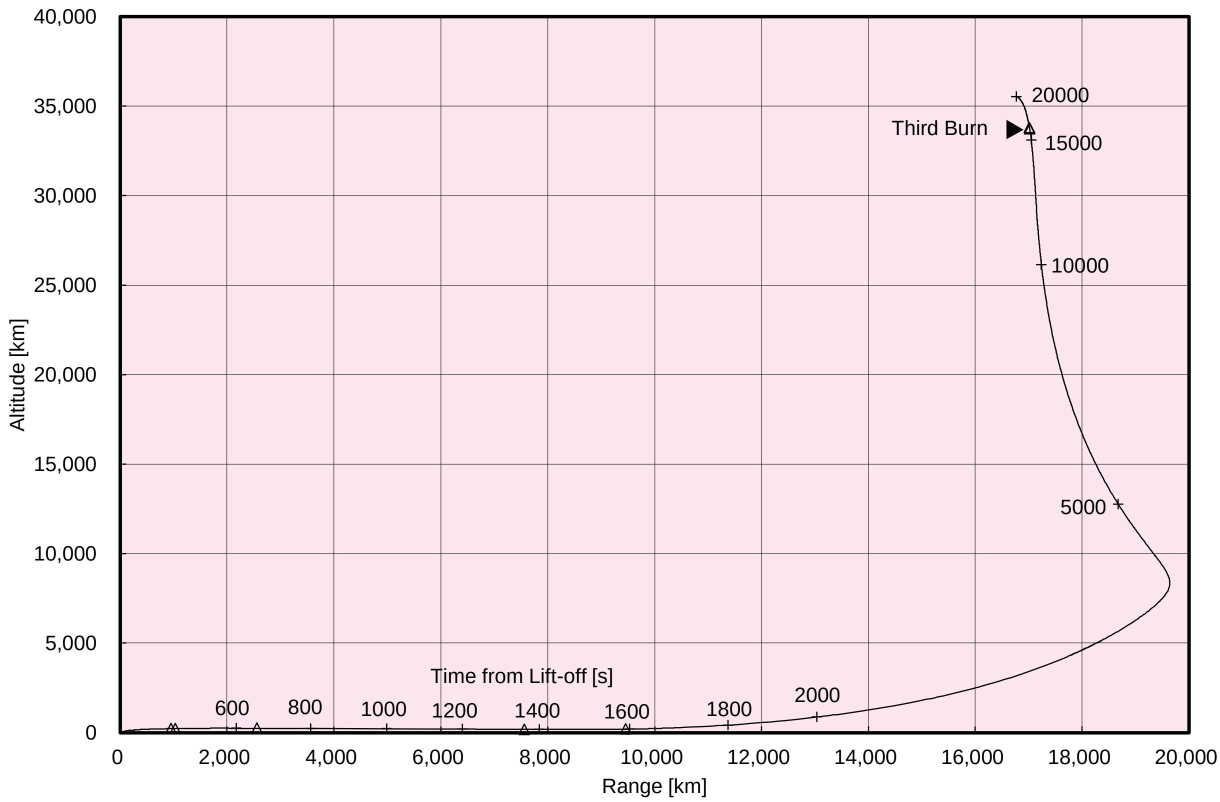


Figure 2.3-25 Typical flight trajectory for Long-coast GTO mission (H2A204) Altitude ～ Range (Lift-off~SECO3)

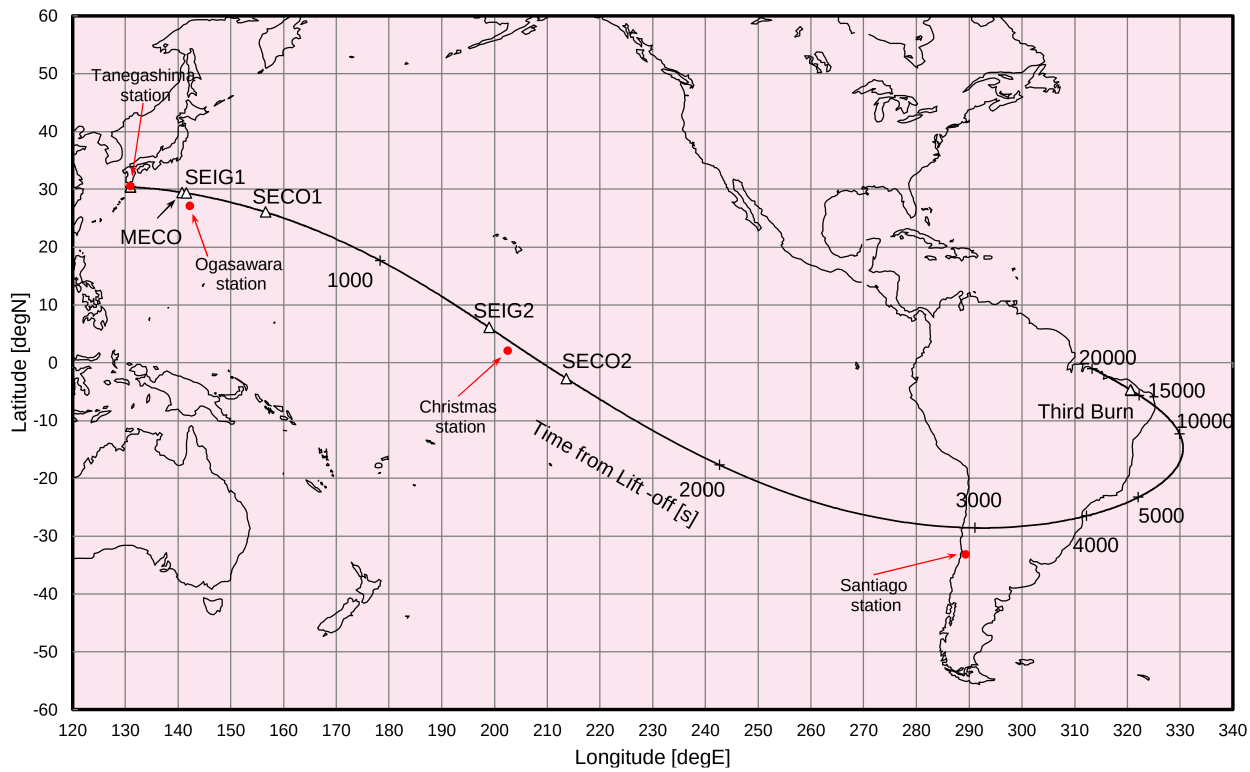


Figure 2.3-26 Typical flight trajectory for Long-coast GTO mission (H2A204) Latitude ～ Longitude

2.4 Sun Synchronous Orbit (SSO) Mission

2.4.1 Launch capability

Payload mass with model 4S fairing for SSO mission is approximately 3,300 kg based on orbital parameters in Section 2.4-2.

With model 5S fairing, the payload mass decreases by 300 kg approximately.

Launch capability of H2A202 with model 4S fairing for SSO mission is shown in Figure 2.4-1.

2.4.2 Typical orbital parameters

Typical orbital parameters for SSO mission are as follows :

Circular orbit altitude	h	=	800.0 km
Inclination	i	=	98.6 deg

2.4.3 Injection accuracies

Typical injection accuracies for SSO mission are as follows based on parameters in Section 2.4.2.

Semi-major axis	a	=	± 10.0 km
Inclination	i	=	± 0.18 deg
Eccentricity	e	=	0 ~ 0.001

(These values are 3-sigma level.)

2.4.4 Typical trajectory

Typical flight trajectory of H2A202 with model 4S fairing for SSO mission is shown in Figure 2.4-2.

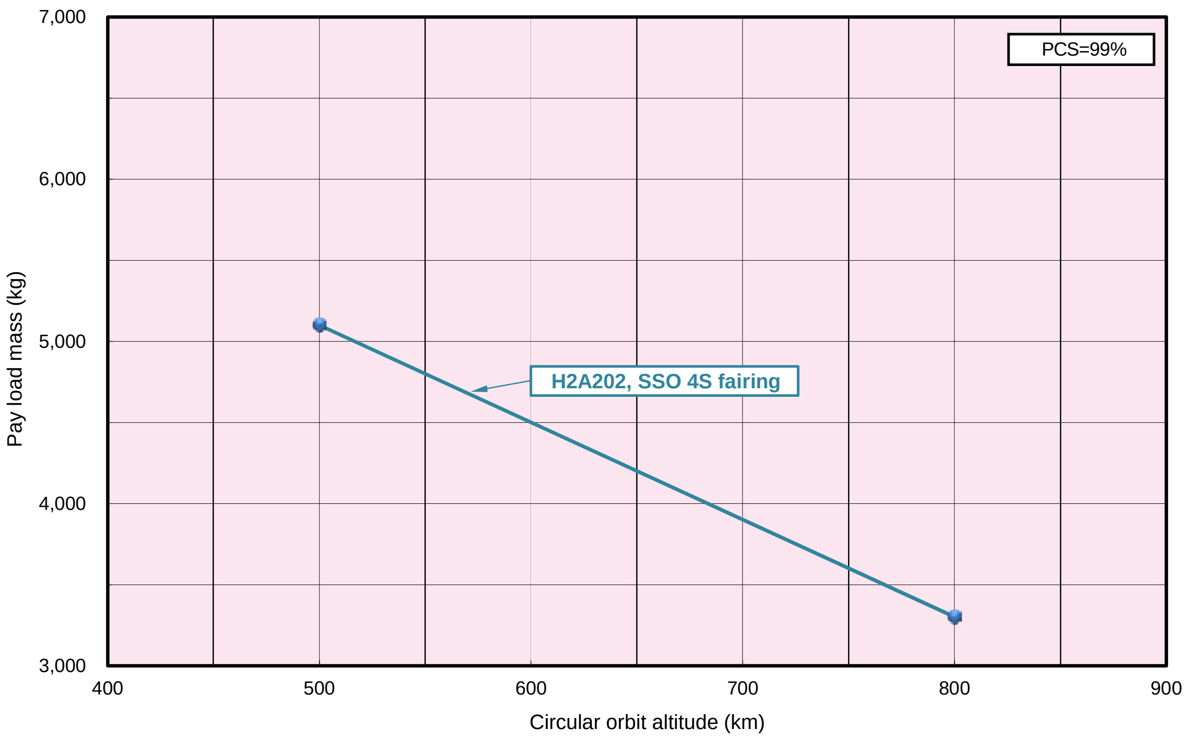


Figure 2.4-1 Launch capability for SSO mission (H2A202 and H2A204 with Model 4S fairing)

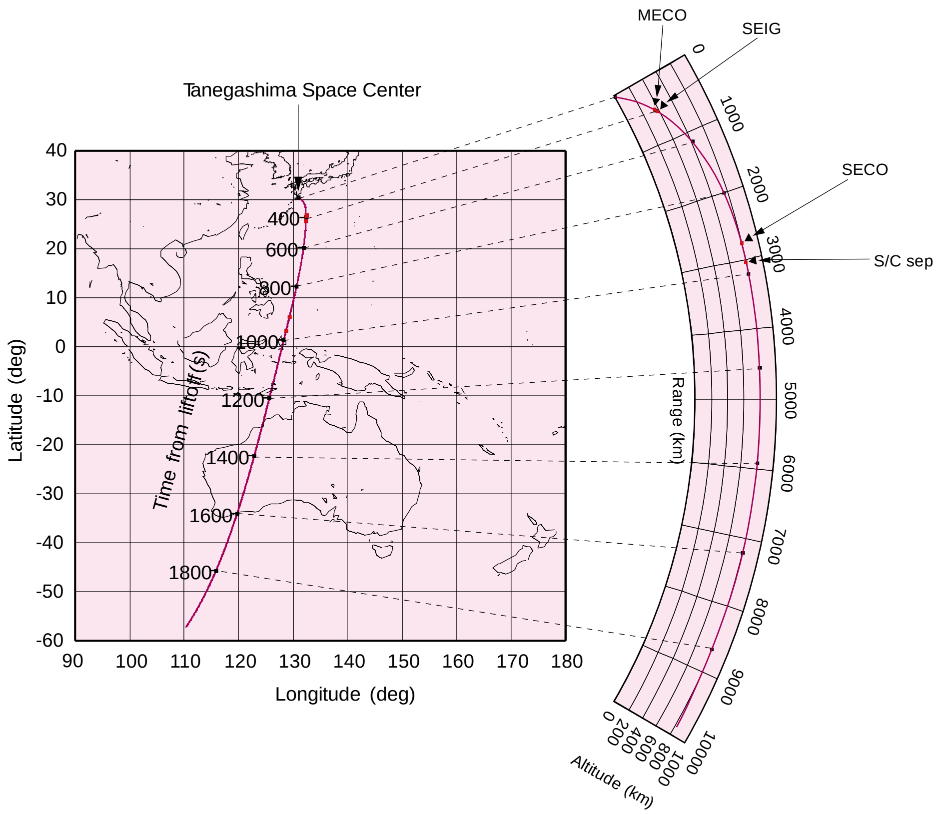


Figure 2.4-2 Typical flight trajectory for SSO mission (H2A202 with Model 4S fairing)

2.5 Low Earth Orbit (LEO) Mission

2.5.1 Launch capability

Payload mass of H2A202 with model 5S fairing for LEO mission (inclinations of 30.4 degree and 51.6 degree) is shown in Figure 2.5-1 and Figure 2.5-2

With model 4S fairing, payload mass will be greater than 5S fairing depending on the injection orbit.

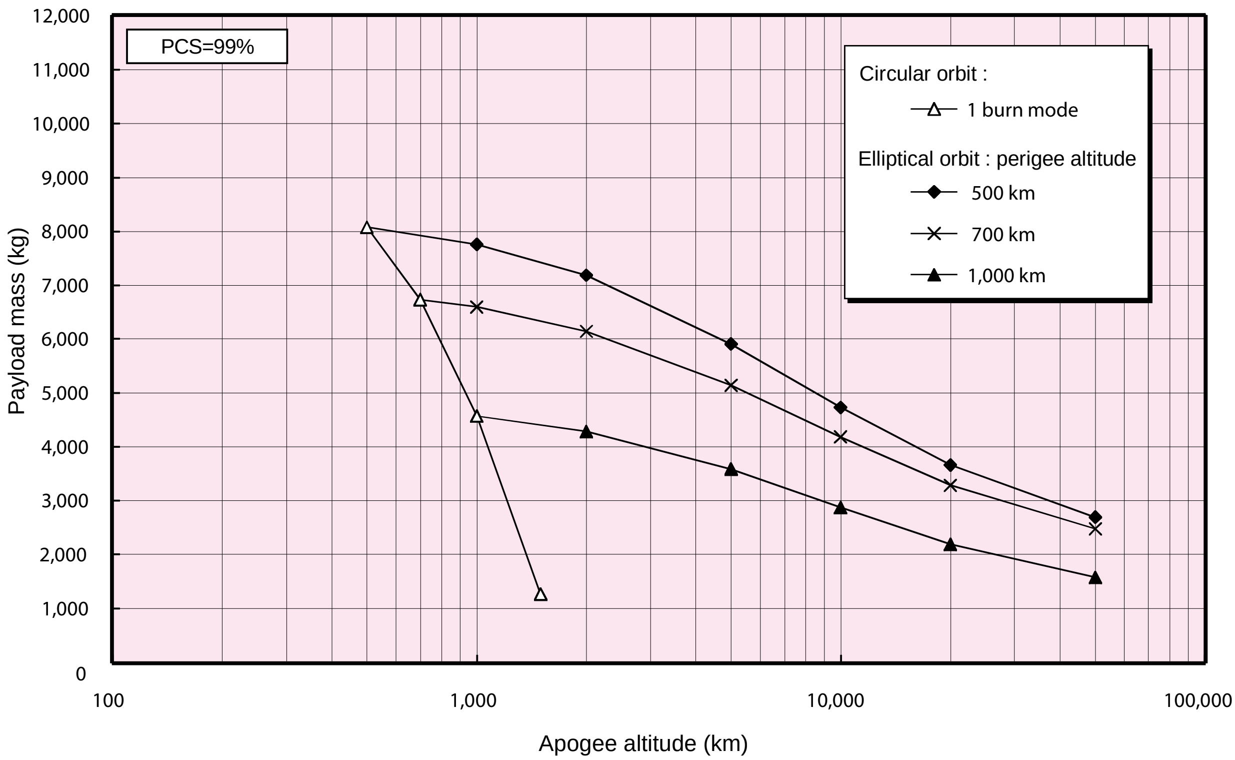


Figure 2.5-1 Launch capability for LEO mission (H2A202 with Model 5S fairing) (inclination 30.4 deg)

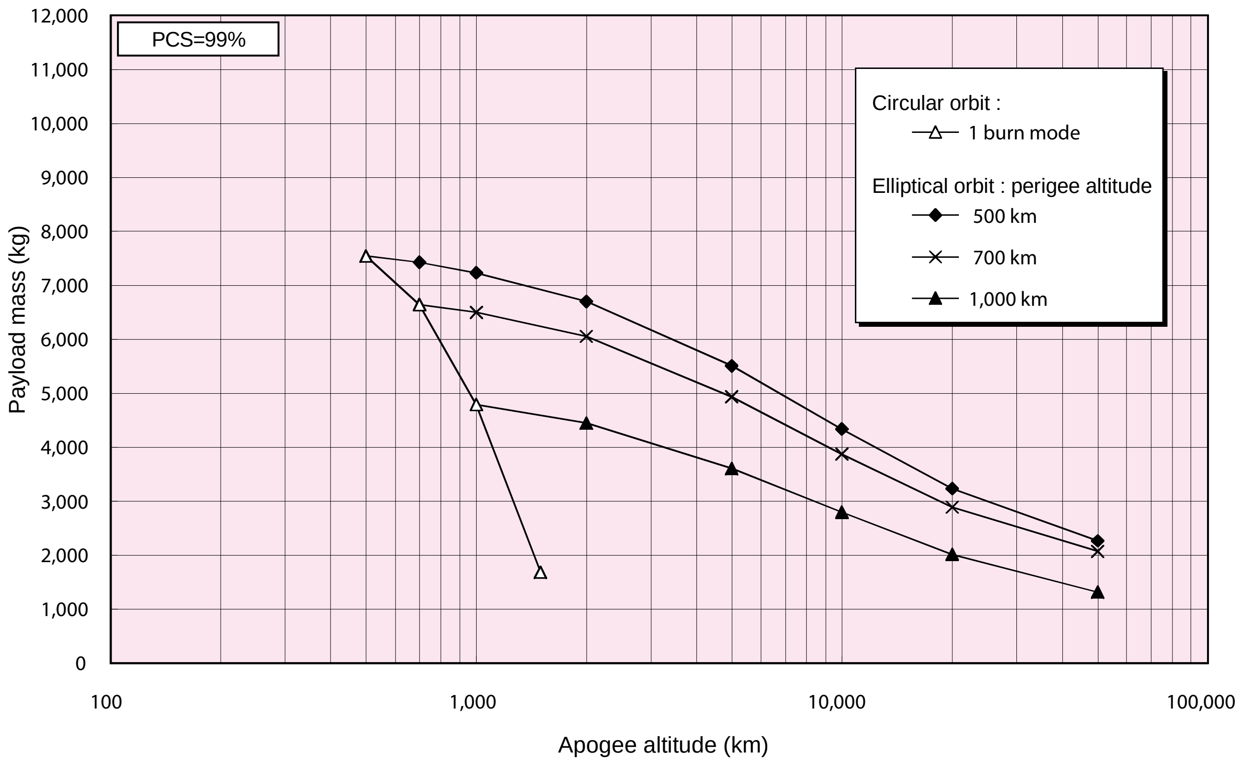


Figure 2.5-2 Launch capability for LEO mission (H2A202 with Model 5S fairing) (inclination 51.6 deg)

2.6 Earth Escape Mission

2.6.1 Launch capability

Payload mass of H2A202 with model 4S and 5S fairing for Earth Escape mission is shown in Figure 2.6-1.

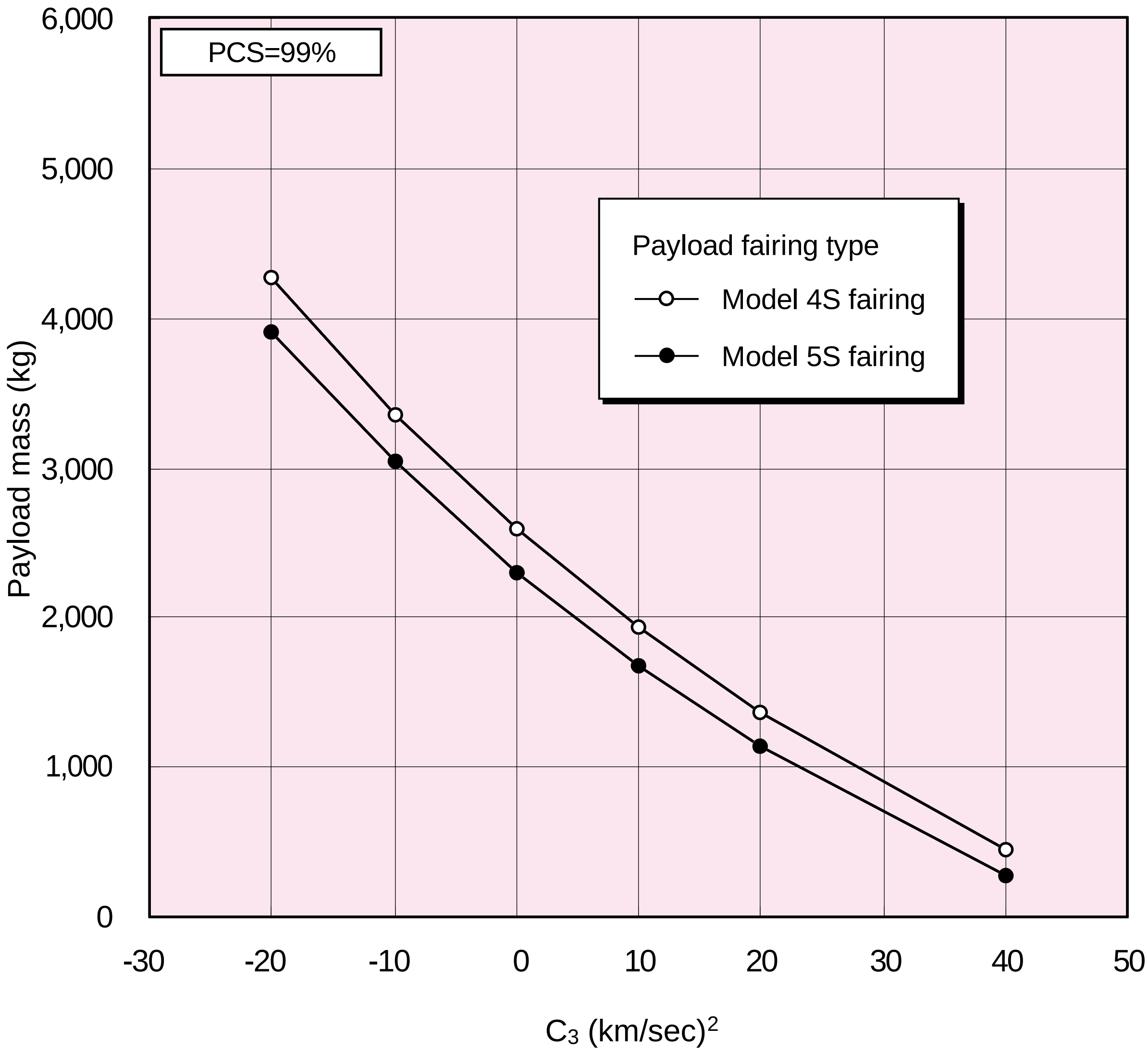


Figure 2.6-1 Launch capability for earth escape mission (H2A202)

2.7 Spacecraft Orientation and Separation

2.7.1 General description

When injected into the planned orbit, the second stage provides the spacecraft with required orientation and spin-up (if required, see Section 2.7.3) prior to separation.

Following the spacecraft separation, CCAM is performed to avoid collision and contamination to the spacecraft.

2.7.2 Separation sequence

(1) Before separation

The second stage attitude is controlled by the guidance and control system according to the mission plan (desired direction, program rate and sequence) with respect to three axis direction using the reaction control (gas jet) system (RCS).

(2) Separation

The second stage guidance control computer (GCC2) sends the separation command to the second stage sequence distribution box (SDB2), and SDB2 sends an ignition current to the separation system (pyrotechnics of clamp band or separation nuts).

The separation is monitored by two (2) separation switches located at the top of the adapter. The separation switches sense the spacecraft separation, and telemetry sends the separation event data to the ground station.

The separation mechanism is by separation spring.

(3) After separation

After spacecraft separation, the second stage conducts the CCAM using the RCS and the residual GH_2 in the LH_2 tank (residual GHe in the ambient helium bottle is used.)

2.7.3 Spin-up performance

The second stage RCS can provide a roll spin for spacecraft with angular velocity up to 5 rpm clockwise or counterclockwise before spacecraft separation. If SCO requires spacecraft spin-up, details should be coordinated with LSP.

2.7.4 Pointing accuracy

Pointing accuracy (deviation from the nominal value) just before spacecraft separation is shown below.

Without pointing maneuver	: Less than ± 10 deg
With pointing maneuver	: Less than ± 3 deg

<Note> These values are:

-around roll, pitch and yaw axes.

-accuracies before spin-up occurs if spin-up is requested.

2.7.5 Relative separation velocity

Relative velocity at spacecraft separation is approximately 0.5 to 1.0 m/s. Other velocity is possible per SCO's request as far as it does not affect launch vehicle collision avoidance maneuvers.

For analyzing the spacecraft/launch vehicle relative orbit, SCO is requested to provide a spacecraft orbit and attitude control plan after separation.

2.7.6 Separation tip-off rate

Separation tip-off rate varies depending on the spacecraft physical characteristic.

The spacecraft is ejected by the spring ejection mechanism installed on payload adapter and typical tip-off rate is less than 2.0 deg/s.

2.7.7 Dual launch sequence

Separation sequence for dual launch is shown as below.

- LEO (upper)-GTO (lower) mission : Figure 2.7-1
- GTO (upper)-GTO (lower) mission : Figure 2.7-2

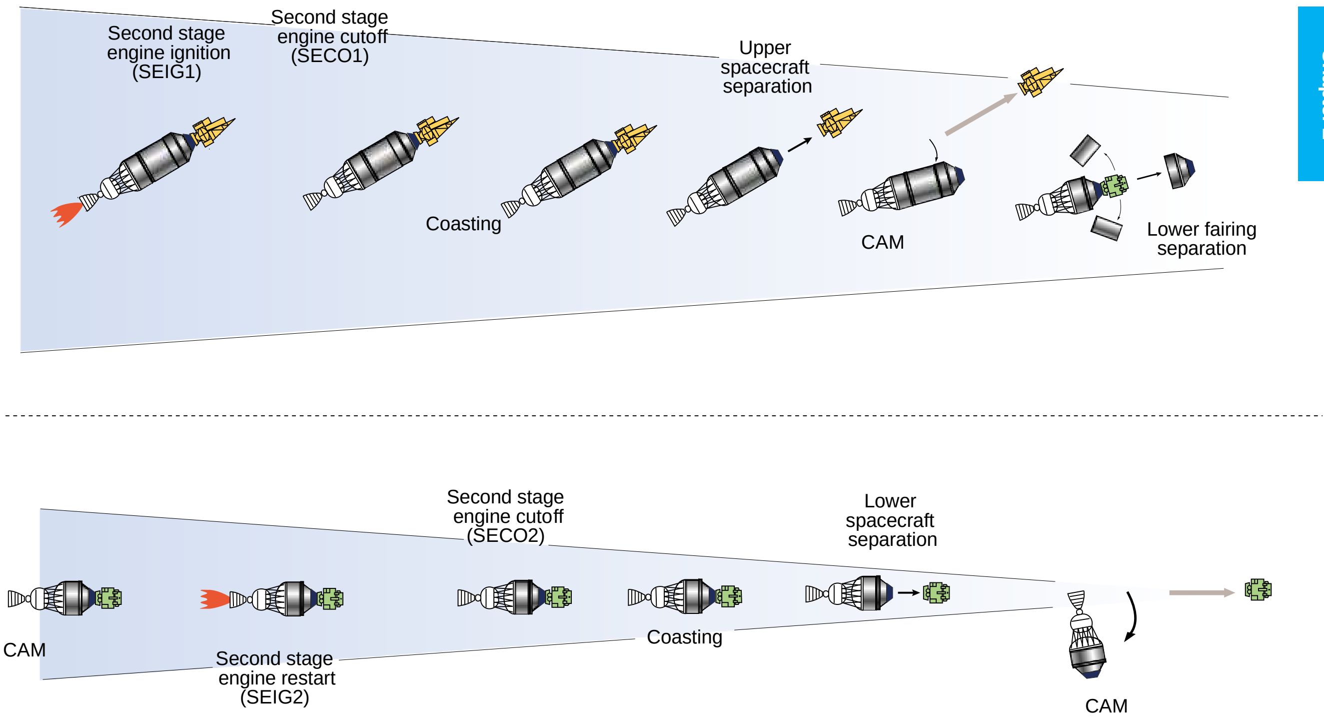


Figure 2.7-1　Sample image of the separation sequence for dual launch on LEO-GTO mission

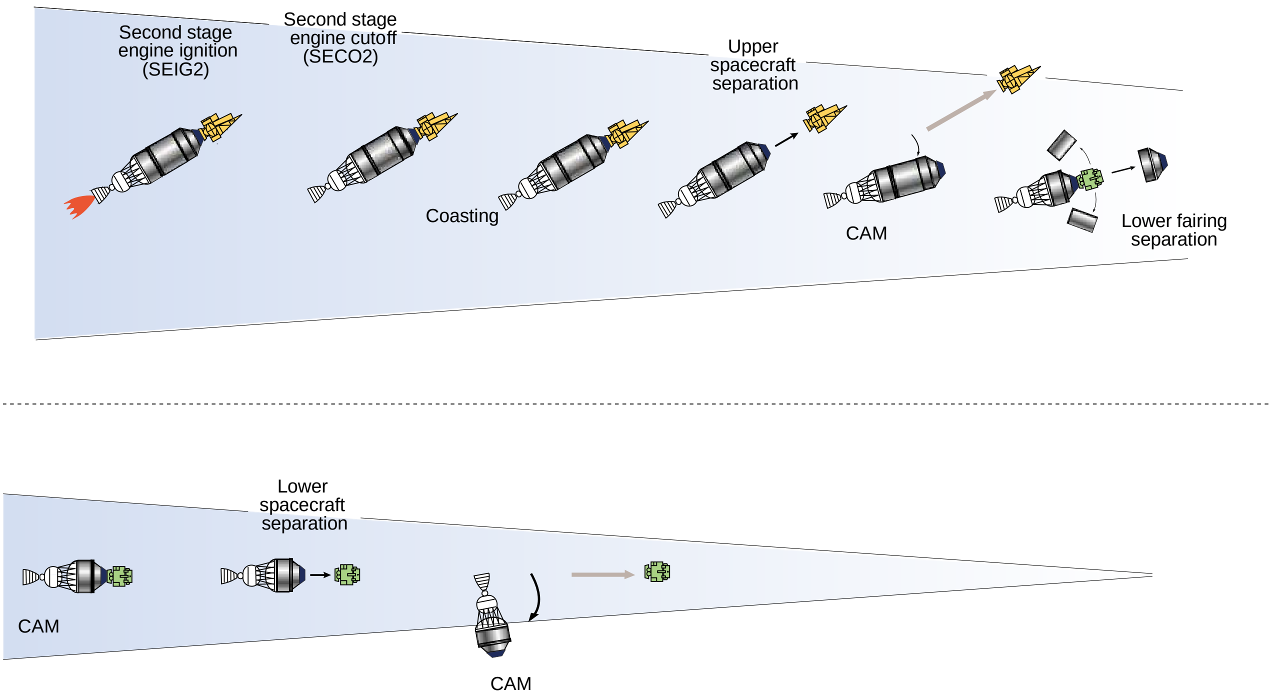


Figure 2.7-2 Sample image of the separation sequence for dual launch on GTO-GTO mission

Chapter 3　ENVIRONMENTS

3.1　General

This chapter describes the environments to which the spacecraft will be exposed in prelaunch and flight phases. The spacecraft has to be designed and tested before launch according to these conditions. Therefore, spacecraft test requirements are also described.

3.2　Mechanical Environments

3.2.1　General

During liftoff and in-flight phases, the spacecraft is exposed to static and dynamic loads caused by the launch vehicle. These environments cover the phases of ground transportation in Tanegashima Island (including transportation to the launch pad), encapsulation of the spacecraft and mounting spacecraft on the second stage. The load factors described in Section 3.2.2 to 3.2.5 should be considered as limit loads applied to the spacecraft.

3.2.2　Combined load factor

Figure2.3-7, Figure2.3-11, Figure2.3-15, Figure2.3-16, Figure2.3-21, and Figure2.3-22 show typical longitudinal static acceleration flight time history.

During liftoff and in-flight phases, dynamic acceleration excited by aerodynamic factors (such as winds, buffeting at transonic phase) and / or forces of the propulsion systems (thrust buildup or tail-off transients of the first stage main engine or SRB-A, etc.) is imposed on the spacecraft. So a combination of static acceleration (the quasi-static acceleration) and low-frequency dynamic acceleration should be considered as the design limit load factors for the spacecraft primary structure.

For the H-IIA launch vehicle, the combined load factor at liftoff and at main engine (of the first stage) cutoff (MECO) transient cover the maximum loads of the spacecraft primary structure during liftoff and in-flight phases.

Table 3.2-1 shows the load factors of 3-sigma high values. Lateral and longitudinal loads may act simultaneously during any phase.

For secondary structures of the spacecraft which have low natural frequencies, load factor on the structures may exceed the above load factors. The acceleration distribution within the spacecraft should be determined using the Coupled Loads Analysis (CLA) results. For more information regarding the CLA, please contact LSP .

3.2.3 Sinusoidal vibration

The spacecraft is exposed to vibration environment that may be divided into two general frequency ranges as follows:

(1) low-frequency sinusoidal vibration

(2) high-frequency random vibration

In this section, (1) is described. And (2) is described in Section 3.2.4 or 3.2.5.

The levels shown in Table3.2-2 are enveloped levels of the low frequency vibrations which are exerted during liftoff and in-flight phases, in particular at liftoff, maximum Qα' (which means the maximum product of dynamic pressure and total angle of attack), the first and the second stage separation, second stage steady state burn, etc. These levels are prescribed at the spacecraft interface (spacecraft separation plane). The second stage steady state burn longitudinal vibration mode appears remarkably on lightweight spacecraft. To reduce the vibration, passive damping device is available for lightweight spacecraft. For more information regarding the sinusoidal vibration, please contact LSP.

These conditions do not include the influence of steady acceleration, therefore additional evaluations for this influence are necessary.

If there is possibility during testing that the structure is subjected to overloads due to differences between the flight configuration and the vibration test configuration, a notching procedure will be allowed to avoid overloads. Notching conditions should be based on the CLA. When sinusoidal vibration environmental test cannot simulate the dynamic load of the flight condition sufficiently, please contact LSP.

3.2.4 Random vibration

Spacecraft structure experiences the random vibration (high-frequency), which is primarily caused by the acoustic noise described in Section 3.2.5. If random vibration conditions at the base of the spacecraft are required, please contact LSP.

3.2.5 Acoustics

The spacecraft is exposed to an acoustic environment during the first stage phase until the vehicle ascends to the altitude where an atmospheric influence can be disregarded. Random vibrations are generated by the noise of the first stage main engine and the SRB-A, and the pressure vibration caused by buffeting and boundary layer noise during the phase of the transonic flight and the high dynamic pressure.

Table 3.2-3 and Table 3.2-4 show the acoustic environment level inside the payload fairing for H2A202 and H2A204 respectively. This is the envelope level during launch and flight and defined as 2-sigma high. This level is uniform around the spacecraft. The spacecraft should endure this level for 40 seconds. The reference point 0 dB of sound pressure level (SPL) is equivalent to 20 μPa. The filling effect is not included in these acoustic environment levels.

Regarding acoustic environment, please contact LSP for details.

3.2.6 Shock

Table 3.2-5 shows a summary of pyrotechnic shock events during flight on all kinds of the H-IIA launch vehicle. A type and a location of each separation system are different according to each shock event. In these separation systems, the spacecraft separation device located at the spacecraft separation plane produces the highest shock.

Figure 3.2-1 and Figure 3.2-2 show a typical separation shock spectrum at the spacecraft separation plane with 1194M adapter and 1666MA adapter, respectively. These shock spectrums are exerted uniformly in all directions.

Low shock adapters (937LS-H, 1194LS-H, 1666LS-H) can be provided. Figure 3.2-3 shows a typical spacecraft separation shock spectrum at the spacecraft separation plane with 1194LS-H adapter.

If SCO desires to use another familiar adapter, LSP can procure the customer requested adapter. Please contact LSP for details.

Table 3.2-1 Combined loads (3σLimit level)

Event		Acceleration		Remarks
Liftoff	Combined loads * for compression	Longitudinal	- 31.4 m/s^2	-1.7 G (steady) + (-1.5 G (dynamic))
		Lateral	± 17.7 m/s^2	± 1.8 G
	Combined loads * for tension	Longitudinal	- 1.0 m/s^2	- 0.1 G
		Lateral	± 17.7 m/s^2	± 1.8 G
At MECO	Immediately before MECO	Longitudinal	- 39.2 m/s^2	- 4.0 G
		Lateral	± 4.9 m/s^2	± 0.5 G
	MECO Transit	Longitudinal	+ 9.8 m/s^2	+ 1.0 G
		Lateral	± 9.8 m/s^2	± 1.0 G

*): Maximum load at the top of the payload adapter
Lateral: ± may act in either direction
As for the longitudinal loads, all of them are defined with the tension loads as positive.

Table 3.2-2 Sinusoidal vibration (3σLimit level)

Item	Spacecraft Weight	Sinusoidal vibration Level	Frequency	Note
Longitudinal	over 2ton	9.8 $m/s^2_{0\text{-}P}$	for 5 to 30 Hz	
		7.8 $m/s^2_{0\text{-}P}$	for 30 to 100 Hz	
	1ton	9.8 $m/s^2_{0\text{-}P}$	for 5 to 30 Hz	
		19.6 $m/s^2_{0\text{-}P}$	for 30 to 40 Hz	
		7.8 $m/s^2_{0\text{-}P}$	for 40 to 100 Hz	
	0.5ton	9.8 $m/s^2_{0\text{-}P}$	for 5 to 30 Hz	
		24.5 $m/s^2_{0\text{-}P}$	for 30 to 40 Hz	
		7.8 $m/s^2_{0\text{-}P}$	for 40 to 100 Hz	
Lateral	All	6.9 $m/s^2_{0\text{-}P}$	for 5 to 18 Hz	
		5.9 $m/s^2_{0\text{-}P}$	for 18 to 100 Hz	

Excitation is applied at the base of the adapter with a 4 octave/min sweep rate in up and down direction so that vibration levels at the spacecraft interface are equal to the above levels.

Table 3.2-3 Sound pressure level inside fairing with acoustic blanket (H2A202)

Center Frequency (Hz)	SPL (dB)
31.5	125
63	126.5
125	131
250	133
500	128.5
1000	125
2000	120
4000	115
8000	113
OASPL	137.5

The reference point 0 dB = 20 μPa
This level is defined as 2-sigma high

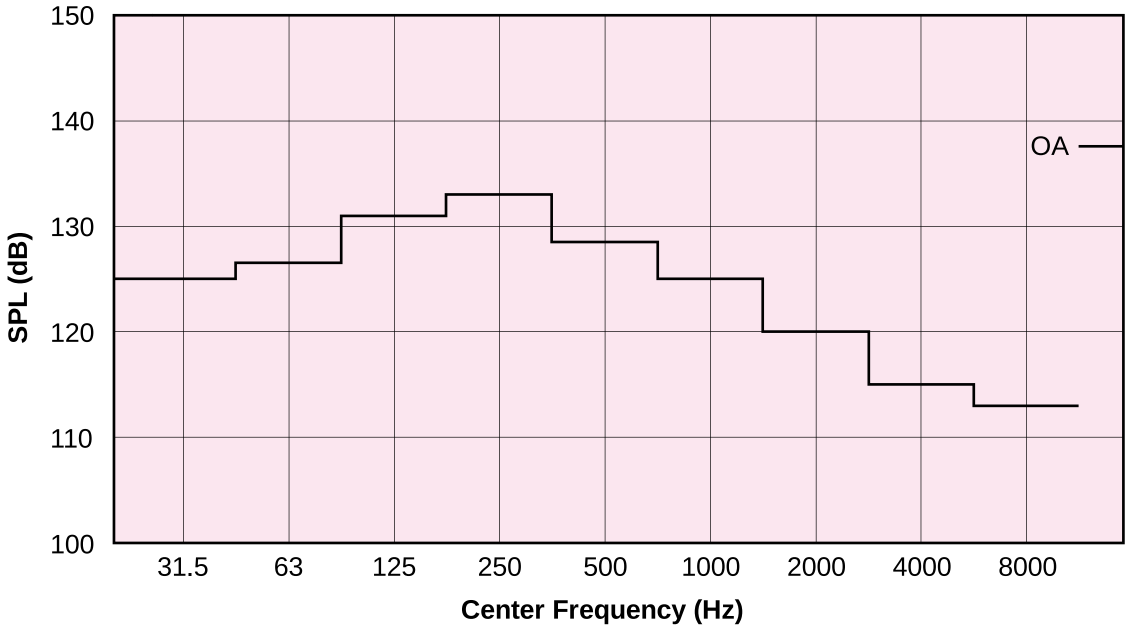


Table 3.2-4 Sound pressure level inside fairing with acoustic blanket (H2A204)

Center Frequency (Hz)	SPL (dB)
31.5	128
63	129.5
125	134
250	136
500	131.5
1000	128
2000	123
4000	118
8000	116
OASPL	140.5

The reference point 0 dB = 20 µPa
This level is defined as 2-sigma high

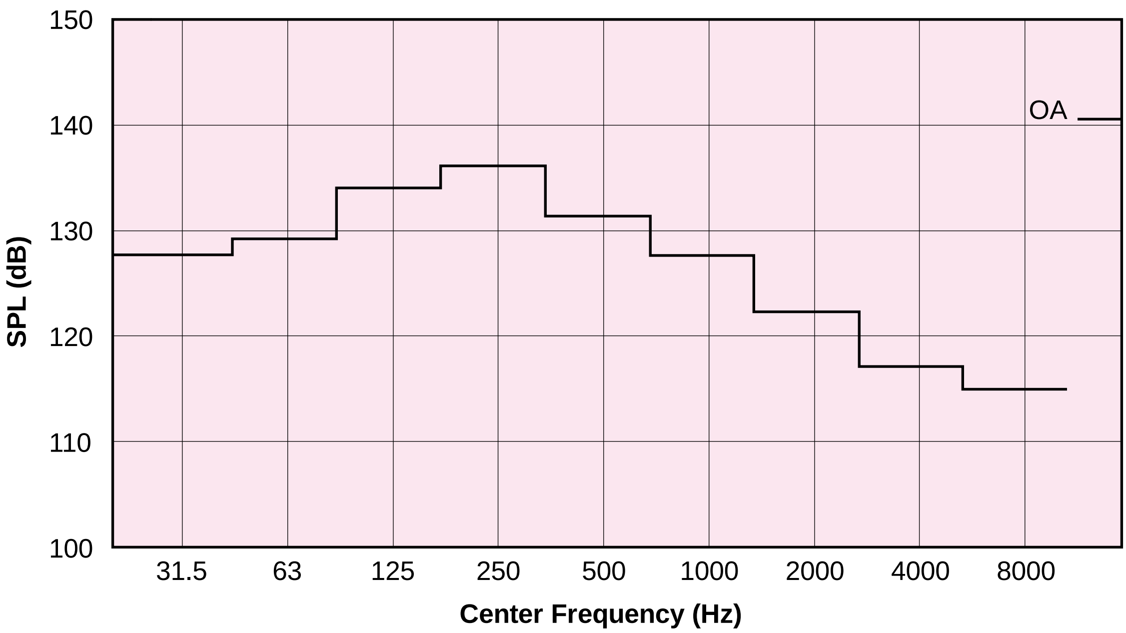
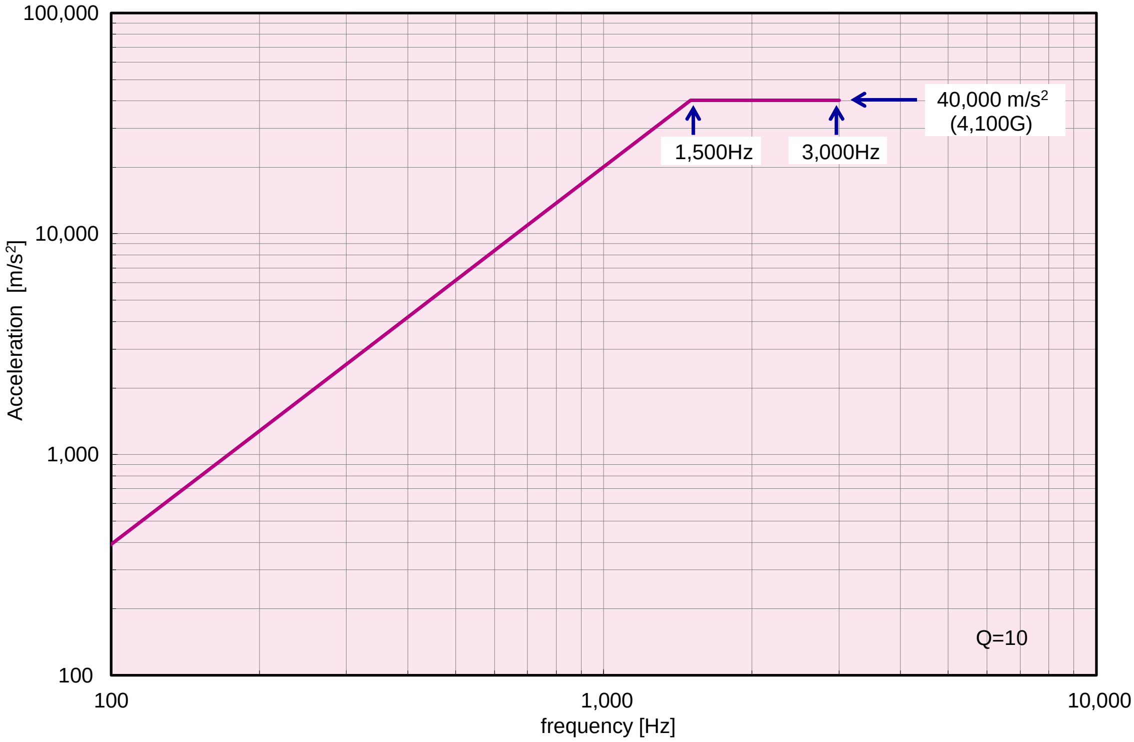


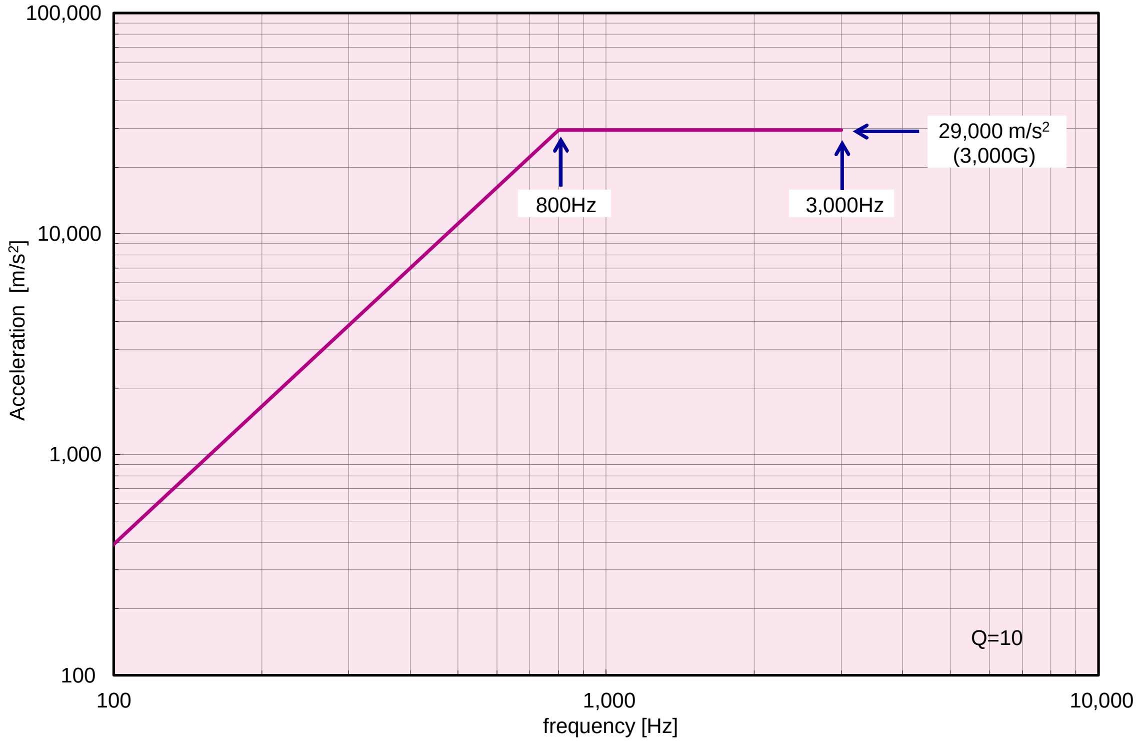
Table 3.2-5 Summary of pyrotechnic shock events

Pyrotechnic shock events	H2A202	H2A204	Remarks
SRB-A separation	✓	✓	
Fairing separation	✓	✓	upper fairing in dual launch
First stage separation	✓	✓	
Spacecraft separation	✓	✓	upper spacecraft in dual launch
Fairing separation (lower)	✓	✓	dual launch only
Spacecraft separation (lower)	✓	✓	dual launch only



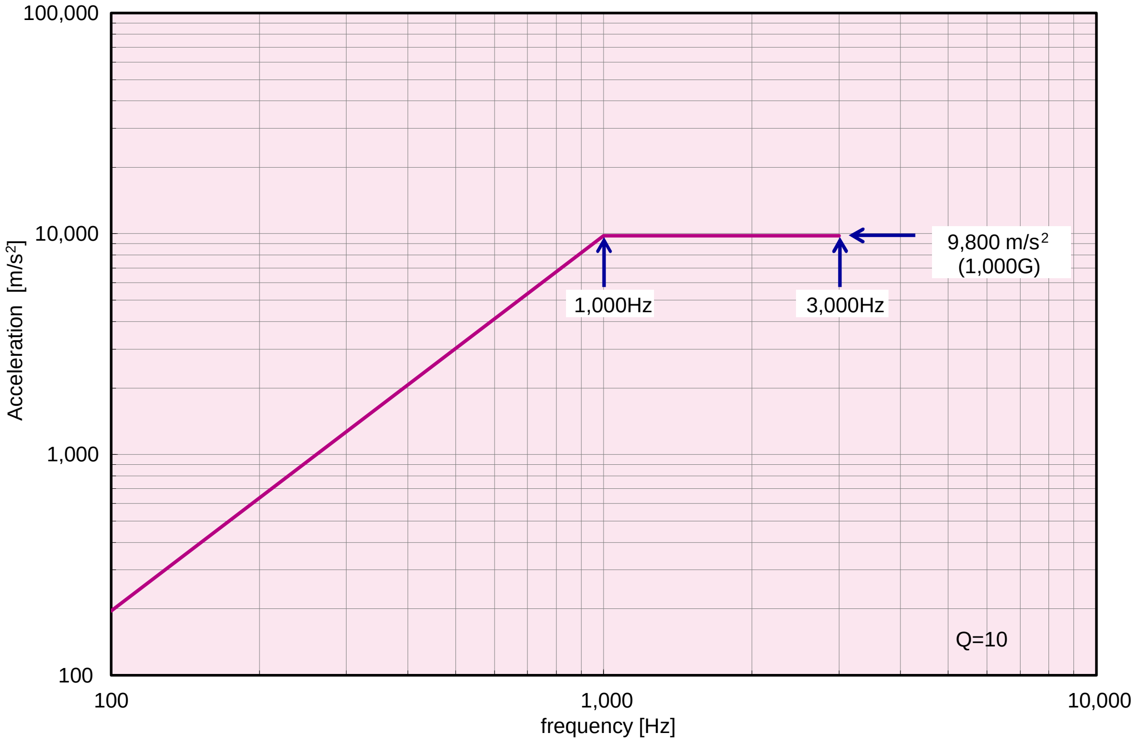
* Separation shock spectrum of 1194M is measured 50mm above separation plane.

Figure 3.2-1 Typical separation shock spectrum with 1194M adapter



* Separation shock spectrum of 1666MA is measured 125mm above separation plane.

Figure 3.2-2 Typical separation shock spectrum with 1666MA adapter



* Separation shock spectrum of 1194LS-H is measured 50mm above separation plane.

Figure 3.2-3 Typical separation shock spectrum with 1194LS-H adapter

3.3 Thermal Environments

3.3.1 General

Concerning thermal environments, following four phases have to be considered:

(1) the phase of spacecraft preparation in the Ground Support Equipment (GSE) buildings and transport between these buildings (Refer to the GSE Manual for details)

(2) the phase of spacecraft encapsulation into the payload fairing and mounting on the launch vehicle in the Vehicle Assembly Building (VAB).

(3) the phase of ML roll-out to the launch pad and the final prelaunch

(4) the in-flight phase

The phases from (1) to (3) are concerned with prelaunch environment and (4) are concerned with launch and flight environments.

3.3.2 Prelaunch environment

The spacecraft thermal environment is controlled during prelaunch activities such as spacecraft integration, functional test, propellant loading, mating to the payload adapter (PLA), encapsulation in the fairing, mounting on the launch vehicle, final checkout, ML roll-out to the launch pad, etc.

Environments during these activities are shown in Table 3.3-1.

From ML roll-out until liftoff, fairing temperature is controlled by the state of fairing inlet, not inside the fairing, and the air dryer removes moisture from compressed air with a dew point as low as 0 ºC. If needed, fairing inlet temperature is adjustable by spacecraft temperature condition, Contact LSP for details.

3.3.3 Launch and flight environments

(1) Inside fairing (from liftoff to fairing separation)

The spacecraft is protected by the payload fairing during ascent to a nominal altitude of approximately 130 km (430 kft). But aerodynamic heating heats the fairing surface and the spacecraft receives a time-dependent radiant heating environment from the internal surface of this fairing prior to fairing separation.

Figure 3.3-1 and Figure3.3-2 show typical internal surface temperature profiles of the Model 4S fairing or Model 4/4D upper fairing for H2A202 and H2A204 mission. And Figure 3.3-3 shows typical internal surface temperature profiles of the Model 5S fairing for a H2A202 mission. Further, Table 3.3-2 shows the maximum temperature and total hemispherical emittance of internal surfaces of the payload fairing during ascent flight. The maximum heat flux of H2A202 is less than 500 W/m^2, and that of H2A204 is less than 600 W/m^2.

Figure 3.3-1, Figure3.3-2, Figure 3.3-3, and Table 3.3-2 are reference.

(2) After fairing separation

The nominal timing for fairing separation on all flights is determined so that the maximum free molecular heat flux does not exceed 1135 W/m^2. Typical free molecular heating profile for GTO mission is shown in Figure 3.3-4 as reference. This heat flux is evaluated as a free molecular flow acting on a plane surface perpendicular to the velocity vector and is based on a standard atmospheric model. The spacecraft thermal environment after fairing separation includes free molecular heating, solar radiation, earth radiation, earth albedo, radiation to the second stage and deep space and thermal flux conducted from the forward end of the second stage through the spacecraft adapter. Solar and earth thermal radiation can be controlled as required by the spacecraft by selecting launch time, vehicle attitude (including rolls) and proper mission design. Thermal heat conducted from the second stage, and influence of RCS plumes and expelled LH_2 / GH_2 and LOX / GOX are usually small. These values are estimated in the mission analysis.

Table 3.3-1 Air conditioning capabilities

Phase	Air conditioning item					Note
	Typical temp.	Temp. control Capability	Typical Relative humidity	Relative humidity Capability	Cleanliness	
Spacecraft assemble area(STA2,STA1,SFA2)	22 ± 3 ºC	15-30 ºC	50 ± 10 %	Below 60%	Class 100,000	Redundant facilities air conditioning
Transport from STA2 to SFA	N/A	N/A	N/A	N/A	Class 100,000	To maintain environment by using GN_2
Propellant loading area SC encapsulation area (SFA,SPLB,SFA2)	21 ± 3 ºC	15-30 ºC	50 ± 10 %	Below 60%	Class 100,000	Redundant facilities air conditioning
Transport from SFA or SFA2 to VAB	21 ± 3 ºC	10-25 ºC	A dew point of as low as 0°C	Below 60%	Class 5,000	By fairing air conditioning dolly
VAB payload area	24 ± 3 ºC (in summer) 18 ± 3 ºC (in winter)	15-30 ºC	50 ± 10 %	Below 60%	Class 100,000	Redundant facilities air conditioning
Fairing air inlet at VAB	21 ± 3 ºC	10-25 ºC	50 ± 10 %	Below 60%	Class 5,000	By VAB air conditioning Backed up by fairing air conditioning dolly
Fairing air inlet during ML roll-out	21 ± 3 ºC	10-25 ºC	A dew point of as low as 0°C	Below 60%	Class 5,000	By fairing air conditioning dolly
Fairing air inlet at LP	21 ± 3 ºC*3	10-25 ºC	A dew point of as low as 0°C*1 (0%*2)	Below 60%	Class 5,000	*1) normal operations by LP air conditioning *2) Backed up by GN_2

*3 : Inlet temperature is adjustable within system capability according to Spacecraft requirements.

Table 3.3-2　Typical maximum temperatures and emittance of fairing internal surface

Section	Max. temperature（℃）		Emittance
	H2A202	H2A204	
Nose cap	140	160	0.15
Nose cone	100	130	0.15
Cylinder	120	140	0.15
Acoustic blanket	34	60	0.85

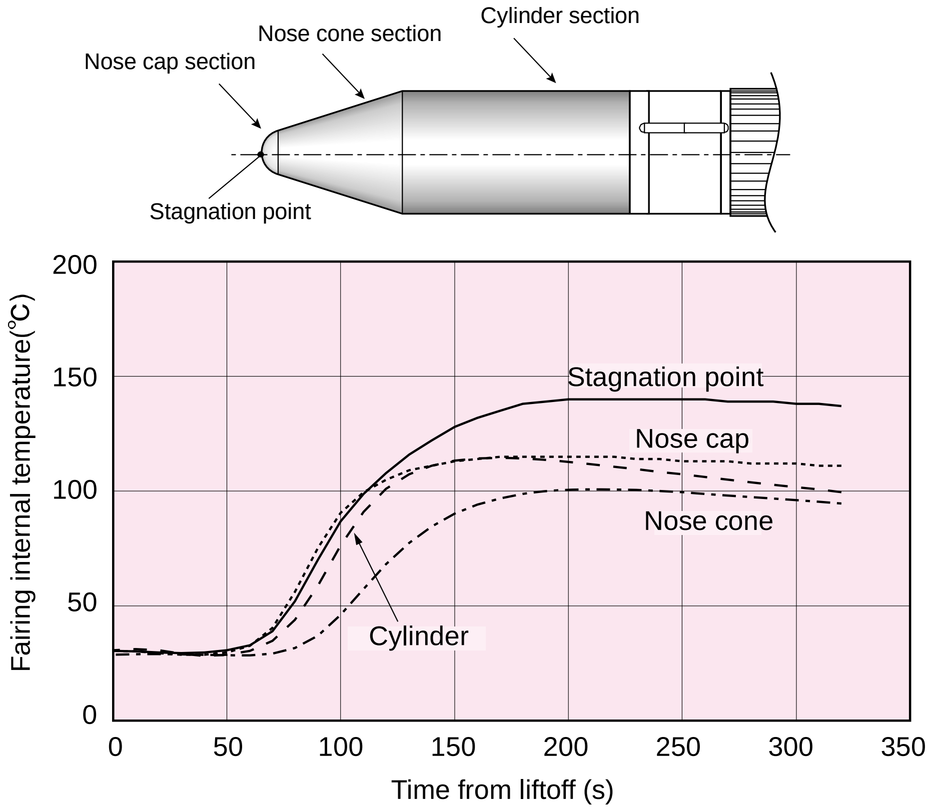


Figure 3.3-1 Typical internal surface temperature profiles of Model 4S or Model 4/4D upper fairing for H2A202

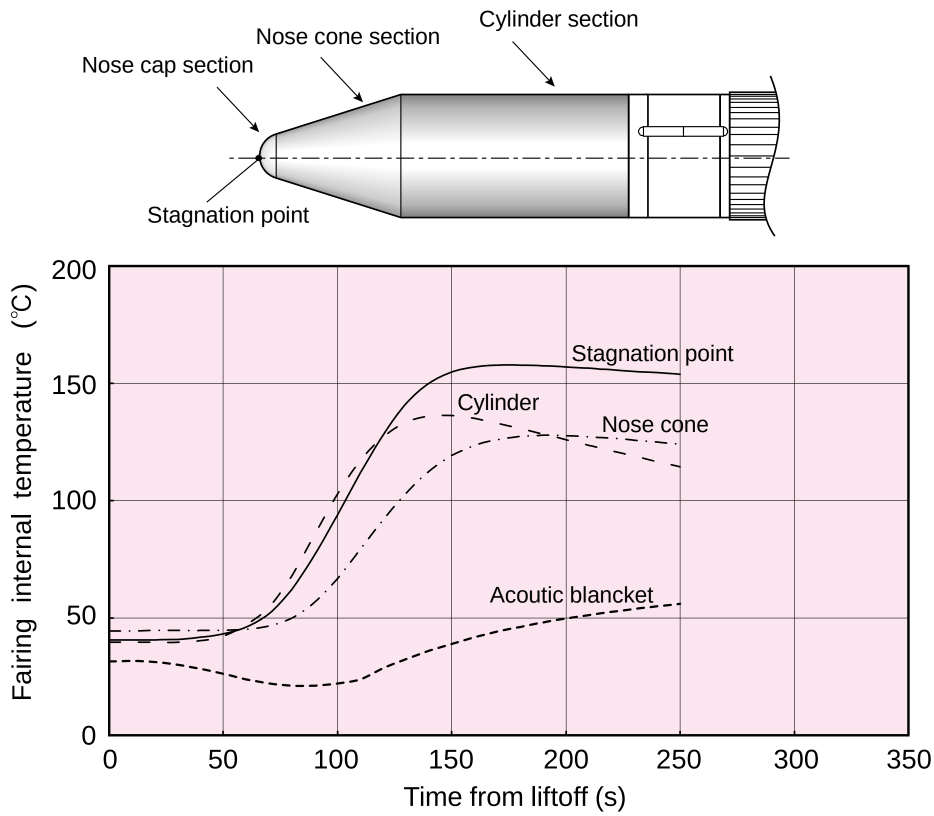


Figure 3.3-2 Typical internal surface temperature profiles of Model 4S or Model 4/4D upper fairing for H2A204

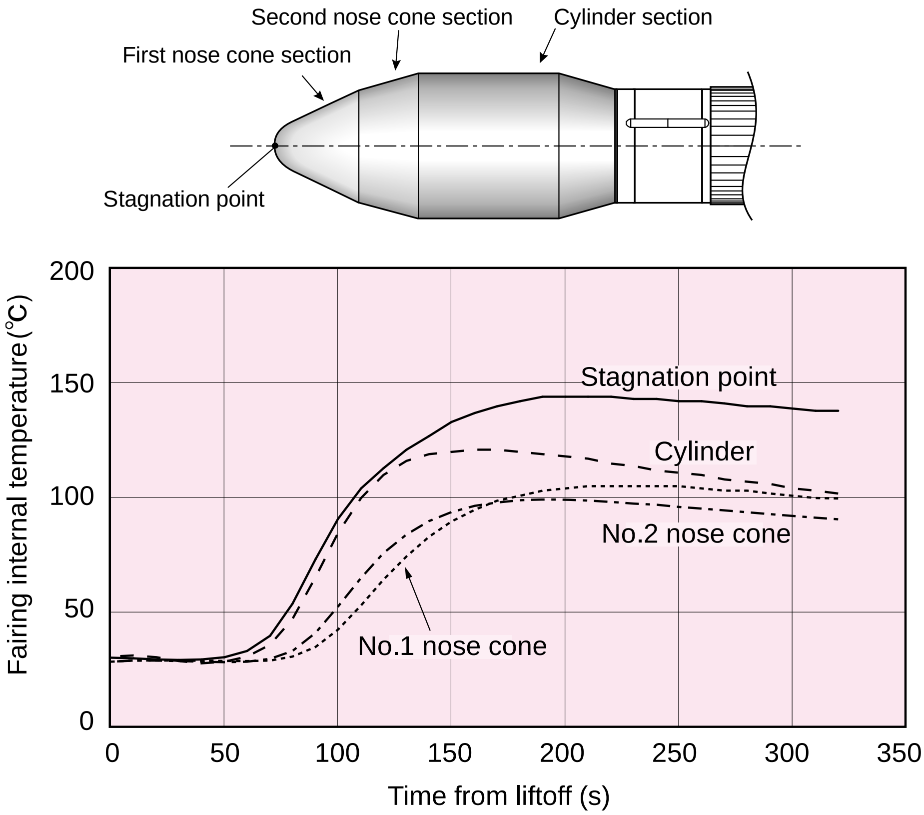


Figure 3.3-3 Typical internal surface temperature profiles of Model 5S fairing for H2A202

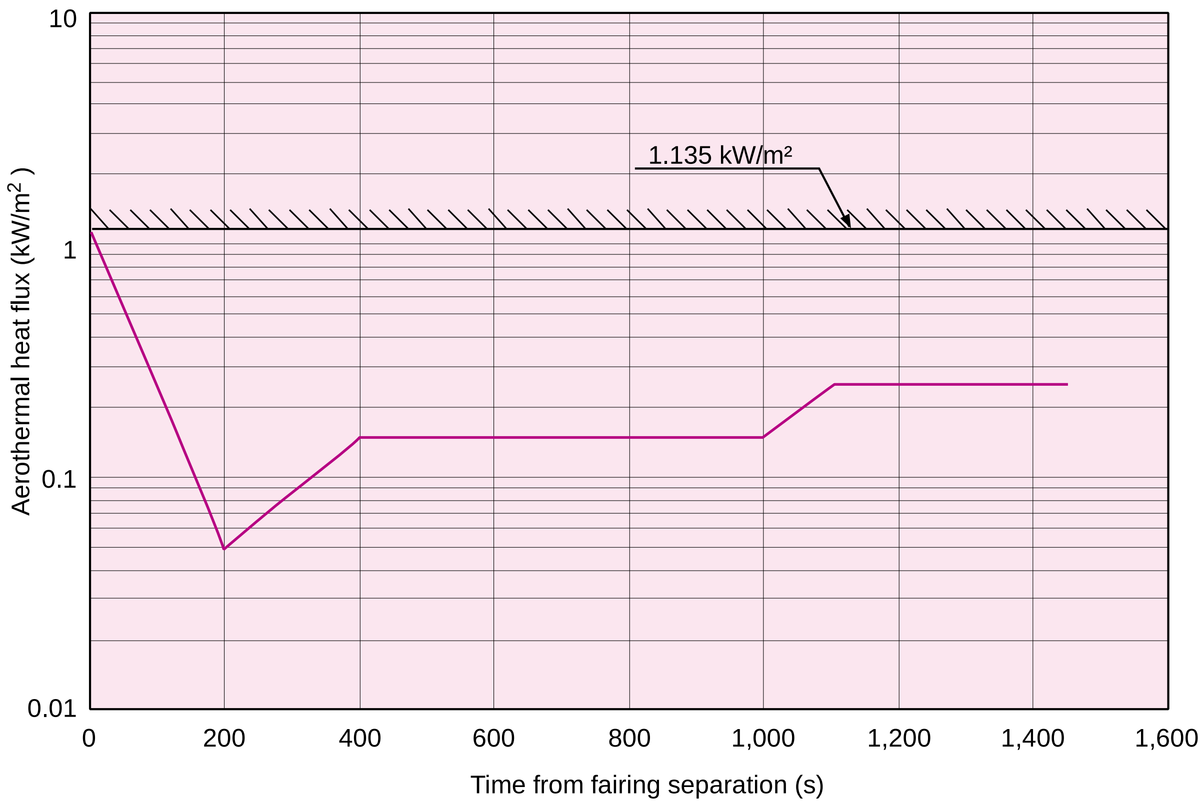


Figure 3.3-4 Typical aerothermal heat flux of the free molecular flow (GTO mission)

3.4 Fairing Internal Pressure Environment

Air inside the payload fairing is vented during the ascent phase through one-way flapper valve. Typical predicted fairing internal pressure profile is shown in Figure 3.4-1.

The pressure decay rate varies while the launch vehicle is in the transonic flight phase. The typical fairing internal pressure decay at the transonic phase continues for 9sec and the pressure decays 25kPa in that duration. Maximum pressure decay rate does not exceed 4.51 kPa/s at that period. The vent area of the launch vehicle payload adapter and fairing is designed assuming that the spacecraft vents some amount of internal volume through the payload adapter. For more information regarding the fairing internal pressure, please contact LSP.

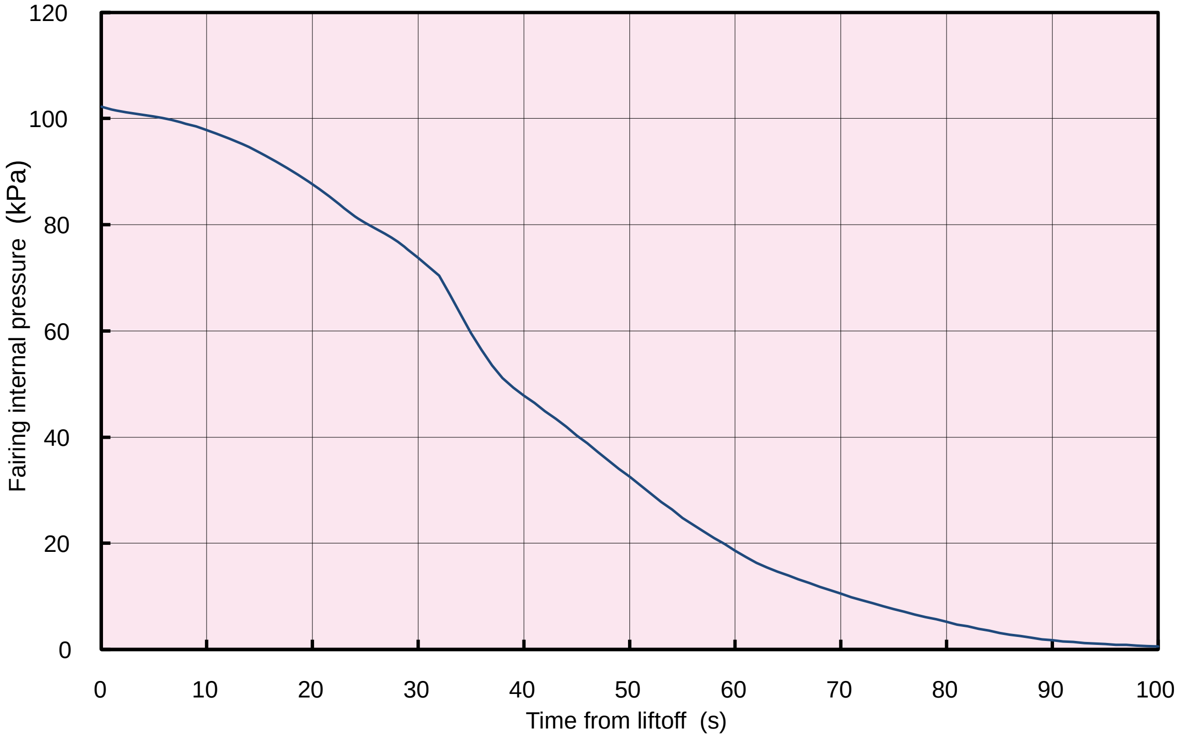


Figure 3.4-1 Typical fairing internal pressure profile

3.5 Contamination and Cleanliness

3.5.1 Prelaunch contamination and cleanliness

The launch vehicle hardware which may affect the spacecraft environment is designed, manufactured and handled according to strict contamination control guidelines. This hardware is defined as a contamination critical item, and includes the payload support structure, the payload adapter and the interior surface of the payload fairing. Air supplied to the spacecraft is also controlled strictly according to the contamination control guidelines.

(1) Contamination control before spacecraft encapsulation

All spacecraft processing area is cleaned up before spacecraft arrival. Before spacecraft encapsulation, the faring, PSS, and PLA inner surfaces are cleaned up and verified with the level "VC+UV"(Visibly Clean Plus Ultraviolet). The surface particle count is performed and particle cleanliness level is recorded to maintain cleanliness. Air is also controlled at less than Class 100,000 conditions through the HEPA filter.

(2) Contamination control after encapsulation

After the spacecraft is encapsulated in the fairing, the payload compartment inside the fairing is completely closed except access doors or large doors. Air controlled at less than Class 5,000 conditions is supplied through the air conditioning duct and maintained in higher pressure condition than around the fairing.

(3) NVR

NVR cumulative rate does not exceed 0.1mg/m^2 per day in any spacecraft processing area.

3.5.2 Flight contamination control

(1) CCAM

The sequence of the spacecraft separation and the collision / contamination avoidance maneuver is carried out as described in Section 2.7.2. In this phase, the RCS is used for attitude control and propellant settling (if required, used as retro-motors to move the vehicle away from the spacecraft after separation). As part of the maneuver, GH_2 is expelled from the LH_2 tank through the propulsive vent port to increase second stage / spacecraft separation distance. Further, for safety disposal of the second stage, LH_2 and LOX are expelled from each tank through each engine chill down port which is non-propulsive, GOX is expelled from the LOX tank through the vent port (non-propulsive), GH_2 is expelled from the LH_2 tank through the non-propulsive or propulsive vent port, and cryogenic GHe is also discharged from GHe bottles through the LOX tank.

The installation location and cant angle of RCS thrusters, vent ports and chill down ports are roughly shown in Table 3.5-1. The RCS is of a module type and two RCS modules are installed on the component equipment panel located at the aft end of the second stage LOX tank and are located in vehicle axis symmetry. One RCS module consists of four 50 N hydrazine (N_2H_4)

thrusters. (If required, six more 50 N thrusters and one 4 N thruster are added per module.) The thruster's exhaust gas is mainly composed of ammonia, nitrogen and hydrogen. Use of RCS thrusters is restricted to minimize contamination products.

The expelled products from each vent and chill down port are hydrogen, oxygen and a small amount of helium which are almost non-contaminating to the spacecraft.

(2) Outgassing

Non-metallic materials inside the fairing when exposed to a vacuum environment do not exceed the following values defined in ASTM E595.

(a) Total mass loss (%TML)	:Less than 1.0%.
(b) Collected volatile condensable materials (%CVCM)	:Less than 0.1%.

Table 3.5-1 Source of gaseous contaminant to spacecraft and its installation location

Gas producting item / quantity	Products	Purpose	Location and cant angle to X axis [*1]	Remarks
< RCS module >	Ammonia, Nitrogen, Hydrogen		Under surface of the component equipment panel	( ) means option thruster
50 N pitch / roll thruster : 4 (4)		Attitude control	90 deg	
50 N yaw thruster : 2 (2)			90 deg	
50 N settling / retention thruster : 2 (2)		Propellant settling	164.4 deg [*2]	
4 N settling / retention thruster : 0 (2)			164.4 deg [*2]	
50 N retro-thruster : 0 (4)		Retrogression	45 deg [*2]	
< LH_2 / GH_2 vent >				
Engine chill down port : 1	Hydrogen	Engine chill down and residual LH_2 expulsion	Aft end of LH_2 tank : about 90 deg	
LH_2 tank vent port : 1	Hydrogen	Residual GH_2 expulsion	164.4 deg [*2]	
< LOX / GOX vent >				
Engine chill down port : 1	Oxygen	Engine chill down and residual LOX expulsion	Aft end of LH_2 tank : 90 deg	
LOX tank vent port : 1	Oxygen	Residual GOX expulsion	Aft end of LH_2 tank : 90 deg	
< CHe vent >				
LOX tank vent port : 1	Helium	Residual CHe discharge	Use LOX tank vent port through LOX tank	

<Note>

*1 : Cant angle means an angle between the X (longitudinal vehicle) axis and a discharge direction of a nozzle or a vent port.(the positive direction of X axis is the forward direction of the vehicle)

*2 : These values are changed by mission.

3.6 Radiation and Electromagnetic Environments

3.6.1 General

To ensure that electromagnetic compatibility (EMC) is achieved for each launch, the electromagnetic environment is thoroughly evaluated. The spacecraft and the launch vehicle system should prevent mutual disturbance for devices and wiring of each system, and be designed to endure any anticipated disturbance. The spacecraft system will be required to provide all data necessary to support EMC analysis employed for this purpose.

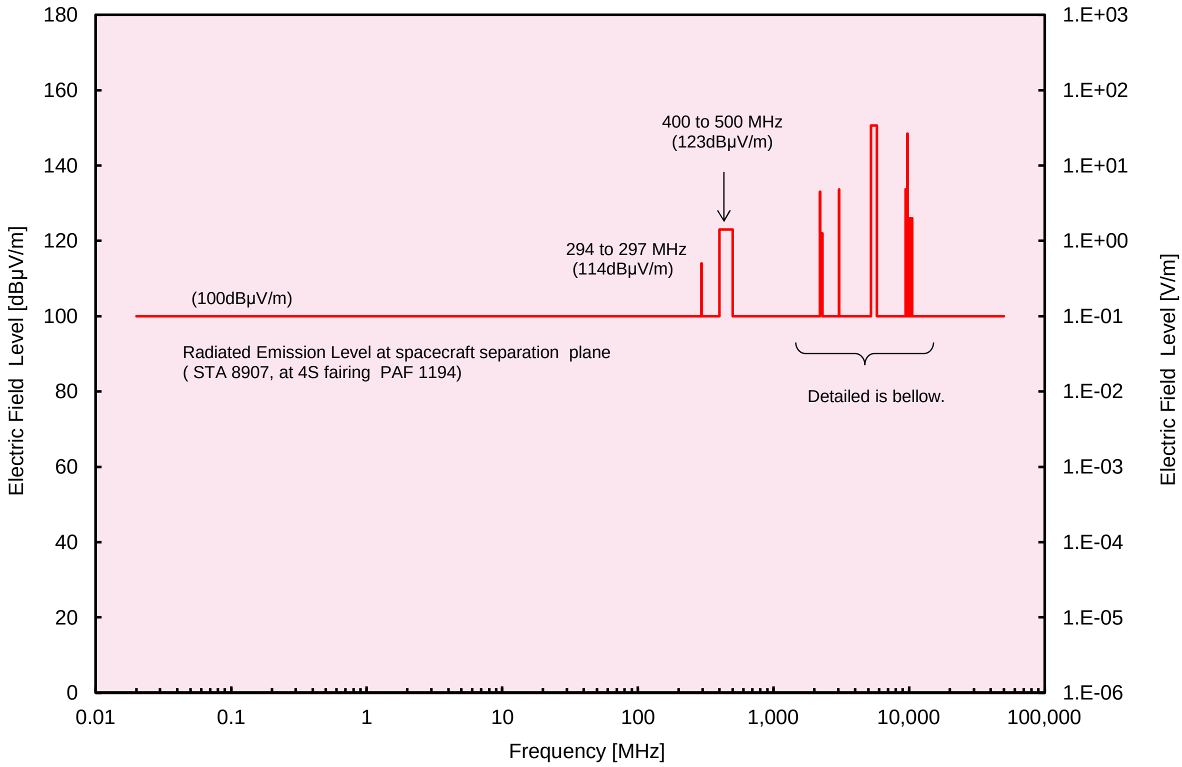
3.6.2 H-IIA radio environment

Radio environment generated by H-IIA launch vehicle on-board RF equipment, TNSC facilities, and nearby facilities are shown in Figure 3.6-1 and Figure 3.6-2. For “Hot launch” spacecraft, SCO should satisfy the following requirements.

(1) Spurious emission level from the launch vehicle and ground facilities is shown in Figure 3.6-1. The spacecraft's spurious susceptibility level should satisfy the spurious emission level in Figure 3.6-1.

(2) The spacecraft should not radiate exceeding the “H-IIA spurious susceptibility level” as the worst case of the sum spurious level shown in Figure 3.6-2.

(3) The spurious emission level and susceptibility level are defined at the spacecraft separation plane.



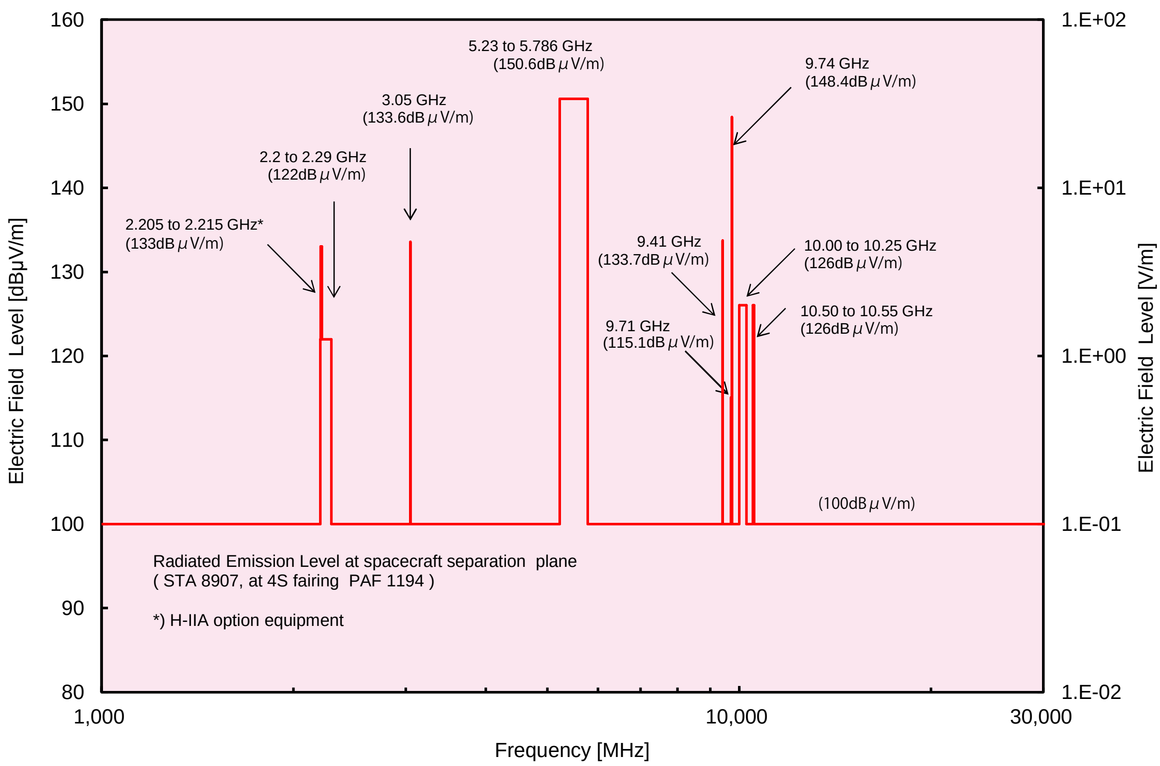


Figure 3.6-1 Spurious emission by launch vehicle and ground facilities

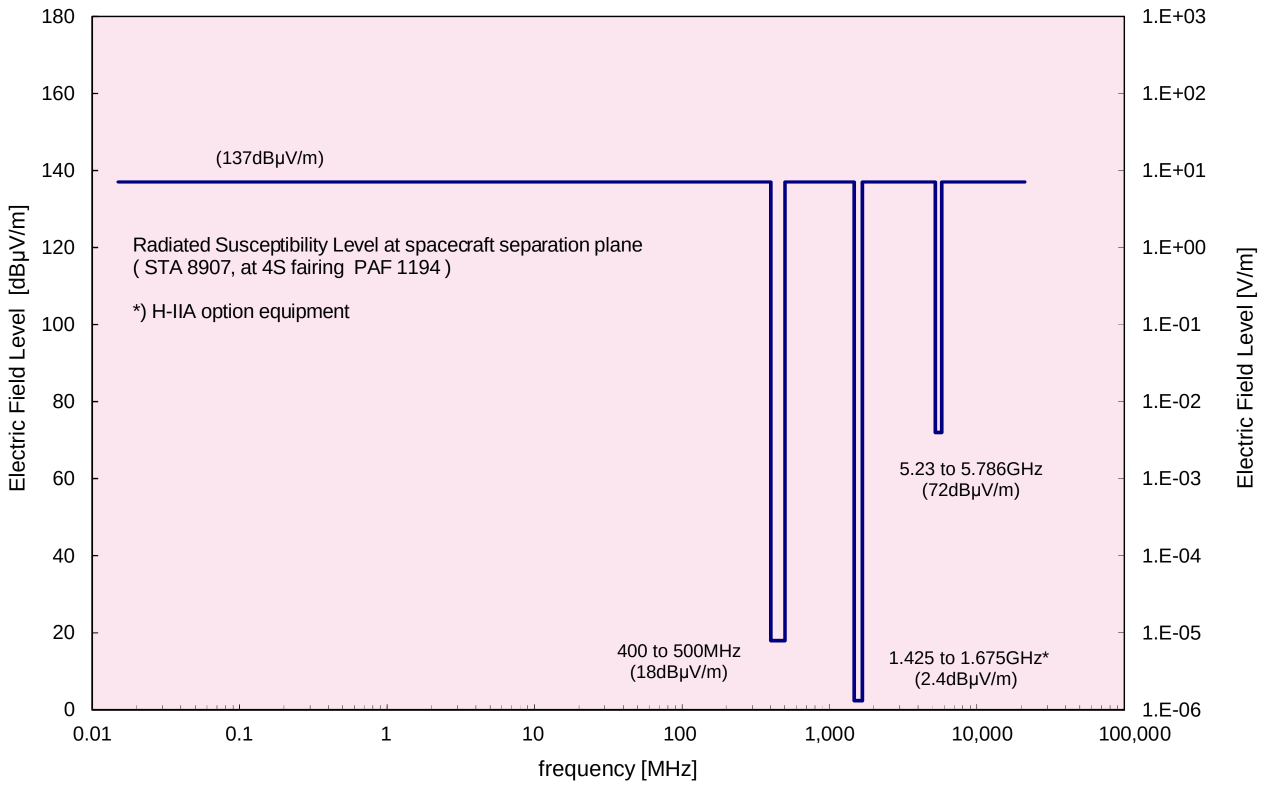


Figure 3.6-2 Spurious radiated susceptibility of launch vehicle

3.7 Spacecraft Compatibility Test Requirements

LSP requires that the spacecraft be capable of withstanding maximum expected flight loads multiplied by minimum factors of safety to preclude loss of critical function. An environmental test report is required to summarize the performed tests and to document the adequacy of the spacecraft structure for flight loads.

The spacecraft tests required for demonstration of compatibility are listed in Table 3.7-1. This table describes tests, margins, and durations appropriate for recommendation in three phases of development. The structural test model (STM) is considered a test-dedicated qualification article. The proto-flight model (PFM) is the first flight article produced without a qualification or STM program. The flight model (FM) is defined as a flight article produced after the qualification or proto-flight article.

LSP also suggests that the spacecraft organization demonstrate the spacecraft compatibility to thermal and EMI / EMC environments.

Flight hardware match mate tests are performed to verify mating interfaces and envelopes. Table 3.7-2 identifies recommended spacecraft qualification and acceptance tests to validate adequate compliance with H-IIA environments.

Table 3.7-1 Spacecraft structural tests, margin, and duration

Test	STM (Qualification)	PFM (Protoflight)	FM (Flight)
Static Level Analyses	1.25 x Limit (DLF or CLA)	1.25 x Limit (CLA)	1.0 x Limit (CLA)
Acoustic Level Duration	Limit + 3 dB 80 seconds	Limit + 3 dB 40 seconds	Limit Level 40 seconds
Sine Vib Level Sweep Rate	1.25 x Limit 2 Oct / minutes	1.25 x Limit 4 Oct / minutes	1.0 x Limit 4 Oct / minutes
Shock	2 Firings	2 Firings	1 Firing

<Note>
DLF : Design load factor
CLA : Coupled load analysis

Table 3.7-2 Spacecraft qualification and acceptance tests requirement

	Acoustic	Shock	Sine Vib	EMI / EMC	Model Survey	Static Loads	Match Mate Test
Qualification	✓	✓	✓	✓	✓	✓	
Acceptance	✓		✓		✓		✓

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Chapter 4 SPACECRAFT INTERFACE

4.1 General

This chapter describes interface requirements between the spacecraft and the H-IIA launch vehicle. In this chapter, spacecraft stiffness, spacecraft balance, mechanical interface, electrical and RF link interface are included. All interface information given is a baseline. For more information regarding interface modifications or tailoring, please contact LSP.

4.2 Launch Vehicle Coordinate System

Figure 4.2-1 shows the H-IIA launch vehicle coordinate system used in this document. The H-IIA launch pad direction (axis-II direction) is 125deg(CW) from north.

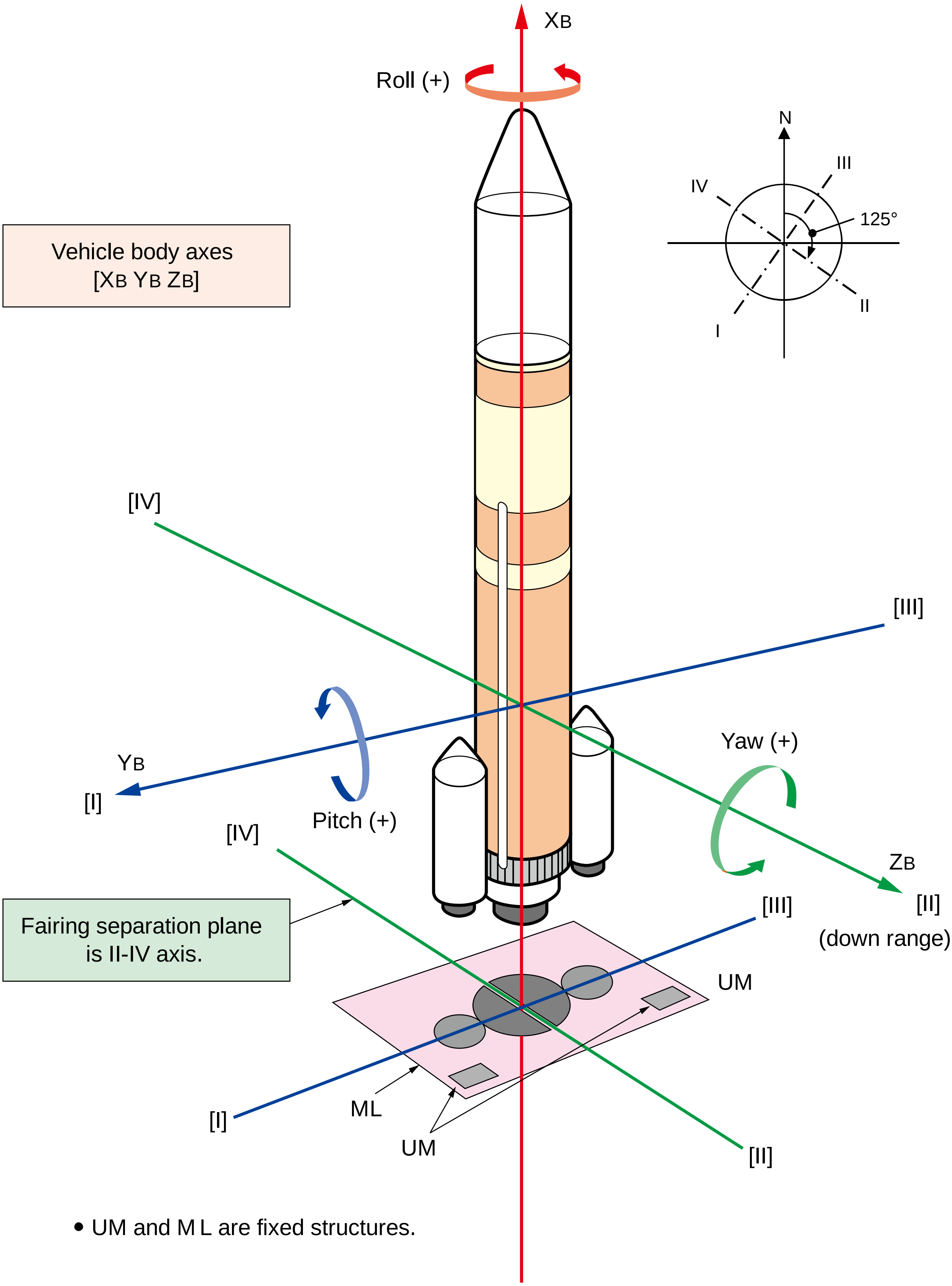


Figure 4.2-1 H-IIA Launch Vehicle coordinate system

4.3 Stiffness of spacecraft

To avoid dynamic coupling modes in the low-frequency between the launch vehicle and the spacecraft during the ascent phase, the spacecraft should be designed with a structural stiffness which satisfies the fundamental frequency requirements.

Under the assumption that the spacecraft is connected rigidly to the separation plane, its primary structure fundamental frequency should be as follows:

(1) Lateral direction	$\geq$ 10 Hz
(2) Longitudinal direction	$\geq$ 30 Hz

If the spacecraft does not satisfy above conditions, SCO should discuss loads, environmental conditions and usable volume, etc., with LSP using a result of Coupled Loads Analysis (CLA) at the technical interchange meetings, and confirm that there are no problems.

4.4 Balance

4.4.1 General

This section describes balance requirements for the spacecraft design. These are related to the payload adapter strength and the center of gravity (CG) of the spacecraft. The CG of the spacecraft affects angular rates at the spacecraft separation and spin-up unbalance for the spin-up spacecraft.

In this manual, the spacecraft CG height means a distance from the spacecraft separation plane to the spacecraft CG.

4.4.2 Height limit of the center of gravity

The height limit of the spacecraft CG is defined according to the strength requirement of each payload adapter. Therefore the height limit of CG depends on the used adapter type. The allowable range of the spacecraft CG height for each type of the payload adapter is described in Appendix 4.

4.4.3 Balance requirements

4.4.3.1 Static balance

The CG offset of the spacecraft in radial direction should be as follows:

(1) Separation without spin-up (or low angular rates separation motion)
Less than 25 mm from the launch vehicle center axis in radial direction

(2) Separation with spin-up (5 rpm in roll axis)
Less than 25 mm from the launch vehicle center axis in radial direction

4.4.3.2 Dynamic balance

For the spacecraft which requires spin-up, the angle between the principal roll inertia axis of the spacecraft and the launch vehicle roll axis (or the spacecraft spin axis) should be less than 3°. For more information regarding the separation with spin-up, please contact LSP.

4.5 Payload Fairing

This section describes the H-IIA launch vehicle payload fairing, especially the configuration, usable volume, and mission modification of fairings. The spacecraft is located on top of the launch vehicle second stage with a payload adapter, and encapsulated within the payload fairing for launch environment protection. For dual launch, the upper spacecraft is located on top of the lower fairing using an adapter for the upper spacecraft.

4.5.1 Fairing types

There are 3 standard payload fairing models.

(1) 4 m diameter dedicated launch model (Model 4S*)
(2) 5 m diameter dedicated launch model (Model 5S*)
(3) 4 m diameter dual launch model (Model 4/4D-LC*)
- upper : long type
- lower : clamshell type

* "4" indicates 4 m diameter, "5" indicates 5 m diameter.
"S" indicates single launch, "D" indicates dual launch.

Table 4.5-1 shows characteristics of payload fairings. Usable volumes of the following payload fairings take into account dynamic displacement of spacecraft which satisfies frequency requirements described in Section 4.3.

4.5.1.1 Model 4S

This payload fairing is available for a dedicated launch of 3.7 m diameter spacecraft.
Figure 4.5-1 shows the Model 4S.
Figure 4.5-2 and Figure4.5-3 show the usable volume of the Model 4S fairing.

4.5.1.2 Model 5S

This payload fairing is available for a dedicated launch of 4.6 m diameter spacecraft.
Figure 4.5-4 shows the Model 5S.
Figure 4.5-5 shows the usable volume of the Model 5S fairing.

4.5.1.3 Model 4/4D-LC

This payload fairing is available for a dual launch of 3.7 m diameter spacecraft.
Figure 4.5-6 shows the Model 4/4D-LC.

Figure 4.5-7 shows the usable volume of the 4/4D-LC upper fairing.
Figure 4.5-8 shows the usable volume of the 4/4D-LC lower fairing.

4.5.1.4 Piggy-back spacecraft

The H-IIA launch vehicle has capability to install maximum of 4 piggy-backs on the PSS. In case of piggy-back installation, LSP will coordinate with SCO in advance.

Figure 4.5-9 shows the accommodations of piggy-back spacecraft.

4.5.2 Stay out zone around the payload adapter

Each PLA has the stay-out zone. Details of the stay-out zone are shown in Appendix 4.

4.5.3 Large door

Two large doors for launch operations of the H-IIA launch vehicle are located at the bottom of the cylindrical part for 4S and 4/4D-LC model, and located at boat tail part for 5S model. The large door is shown in Appendix4. After encapsulation of the spacecraft in the fairing, SCO can access the spacecraft through these doors. Please contact LSP for coordination on using these doors.

Size of large door	: 600 mm x 600 mm
Number of large door	: 2

4.5.4 Mission modifications

4.5.4.1 Access door

After encapsulation of the spacecraft in the fairing, SCO can access the spacecraft through access doors. The access door and allowable access door areas are shown in Appendix 4.

The distance between centers of ø 600mm access doors should be more than 1,200mm and the distance between centers of ø 450mm access doors (options) should be more than 1,000mm. Access doors should normally be located at the cylindrical section of the payload fairing. Access doors should be located in $\pm 120^{\circ}$ area from IV axis in order for SCO to access the spacecraft by diving board through the access door in VAB. Purge gases can be provided through access doors.

Standard size of access door	: 600 mm in diameter (ø 600)
Optional size of access door	: 450 mm in diameter (ø 450)
Standard number of the access doors of ø 600	: 2

4.5.4.2 Umbilical connectors

The umbilical connectors are mounted on the interface plane of the payload adapter. Details of umbilical connectors are described in Section 4.7.6.

Standard number of umbilical connectors : 2

(Option)

If SCO requires, the lanyard umbilical connectors via payload fairing can be used. Details are described in Section 4.7.6.5.

4.5.4.3 Radio transparent window

After encapsulation of the spacecraft in the fairing, a radio transparent window is available to link with the radio signals between the spacecraft and the spacecraft GSE. The radio transparent window is shown in Appendix 4. It will be installed in the same area as the access door.

Material of radio transparent window	: Glass fiber reinforced plastic (GF skin honeycomb sandwich)
Size of radio transparent window	: 450 mm diameter
Standard number of radio transparent windows	: 1

(Option)

Instead of using a radio transparent window, an internal / external antenna connected through a coaxial cable is available to link with the radio signals between the spacecraft and the spacecraft GSE. It can receive and transmit radio signals before liftoff as a substitute for transparent windows.

If SCO require RF link in flight, external antennas are available on outside of fairing as an option.

4.5.4.4 Internal antenna

After encapsulation of the spacecraft in the fairing, an internal antenna inside the fairing is available to link with the radio signals between the spacecraft and the spacecraft GSE. It can receive radio signals from the spacecraft and transmit radio signals from the spacecraft GSE. Typical installation of an internal antenna is shown in Appendix 4. The internal antenna is connected to Ground Support Equipment (GSE) through coaxial lines.

Standard number of internal antennas : 1

4.5.4.5 Air conditioning

After the spacecraft is encapsulated in the PLF, the conditioned air is supplied through the air conditioning duct. In order to vent the air inside the PLF, Air inlet and vent valves (flapper valves) are installed on the PLF. Ventilation airflow velocity near the spacecraft surface is not exceed 2m/s. When fairing is used for dual launch, eight (8) vent holes are installed between the upper and lower fairings.

Air flow rate	: 50 m^3/min each (dual)
	: 100 m^3/min (dedicated)
Size of air inlet	: 400 mm diameter
Standard number of air inlet	: 1 for Model 4S, 5S, 4/4D-LC upper and lower (each)
Number of flapper valves	: 2 for Model 4S, 4/4D-LC upper and lower (each)
	: 4 for Model 5S

4.5.4.6 Acoustic blankets

Acoustic blankets are attached to the inner surface of the fairing to reduce acoustic sound pressure level. Acoustic blankets are made of glass fiber and the cover material. Thickness and size of blanket are determined according to SCO's requirements. Typical configuration of an acoustic blanket is shown in Appendix 4. The standard blanket is 10 mm thick.

Table 4.5-1 Characteristics of payload fairings

Items / Model	Launch	External		Portion of Fairing	Usable volume		Note
		Height	Diameter		Height	Diameter	
4S	Dedicated	12.0	4.07	-	10.23	3.7	
5S	Dedicated	12.0	5.1	-	9.12	4.6	
4/4D-LC	Dual	16.0	4.07	upper	8.23	3.7	
				lower	5.36	3.7	

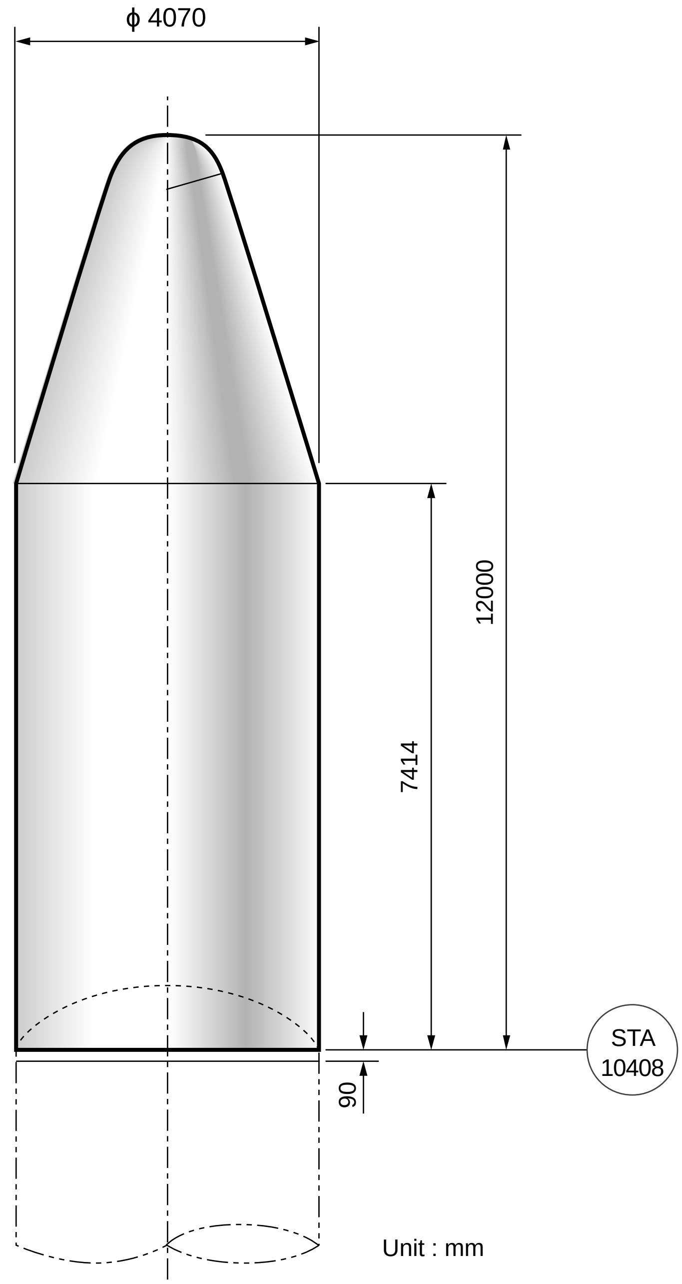
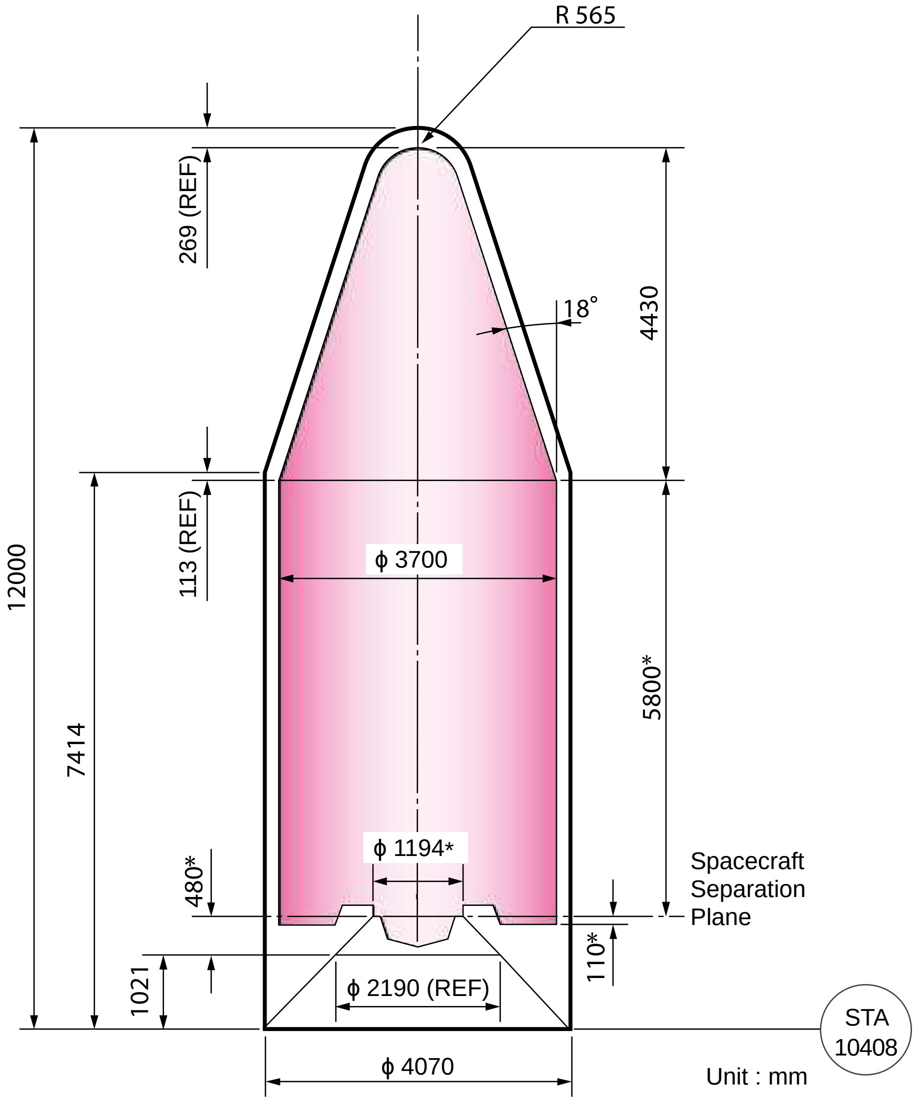


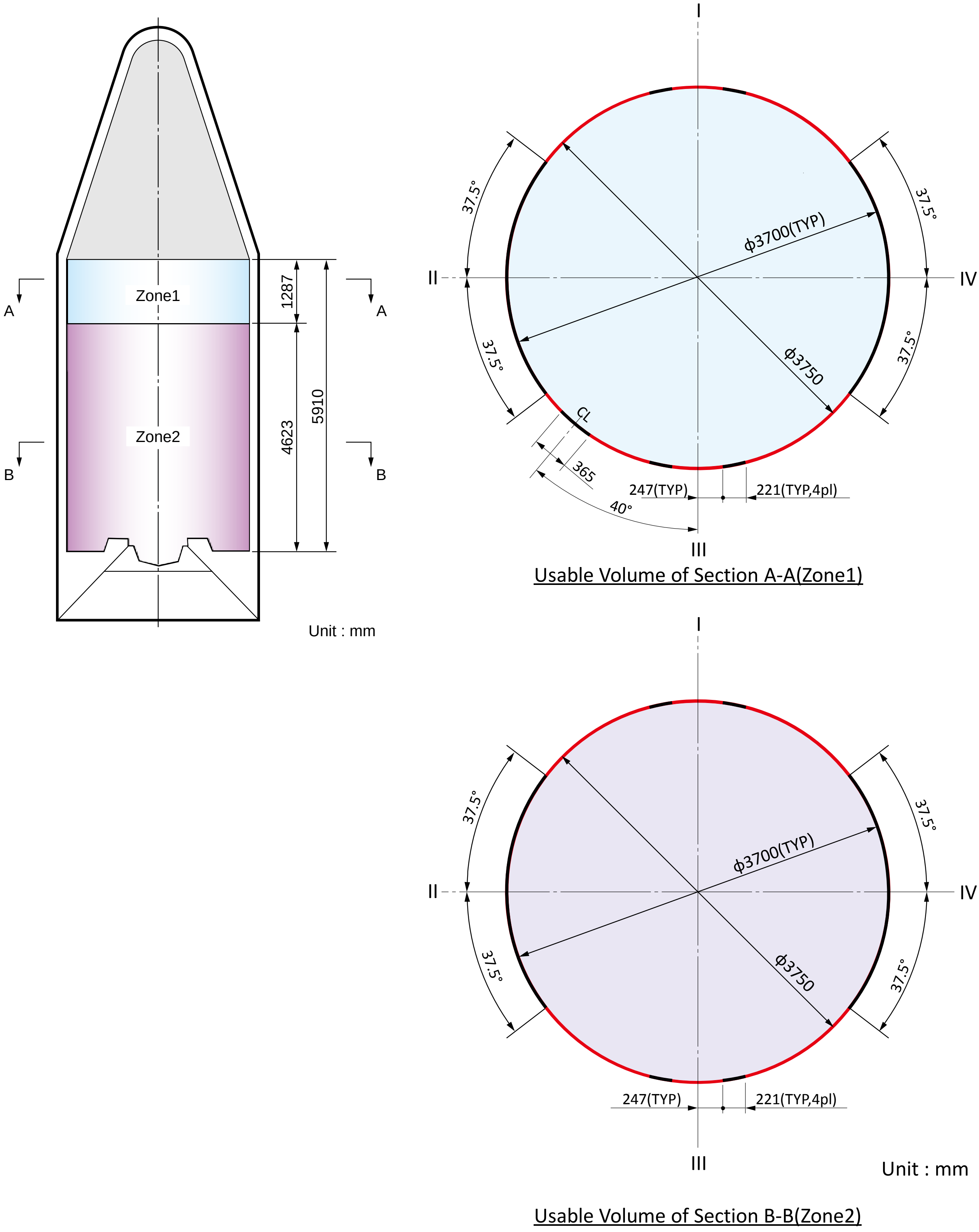
Figure 4.5-1 Model 4S



* These values will vary with adapter model.

Note) The radius of usable volume of the 4S fairing cylinder part is ϕ3700mm ~ϕ3750mm.
Details are shown in Figure 4.5-3.
If Spacecraft will protrude from the usable volume, please contact LSP for more detailed information.

Figure 4.5-2 Usable volume of Model 4S (1/2)



Usable volume diameter = ϕ3700mm : shown in black
Usable volume diameter = ϕ3750mm : shown in red

Figure 4.5-3 Usable volume of Model 4S (cylinder area)

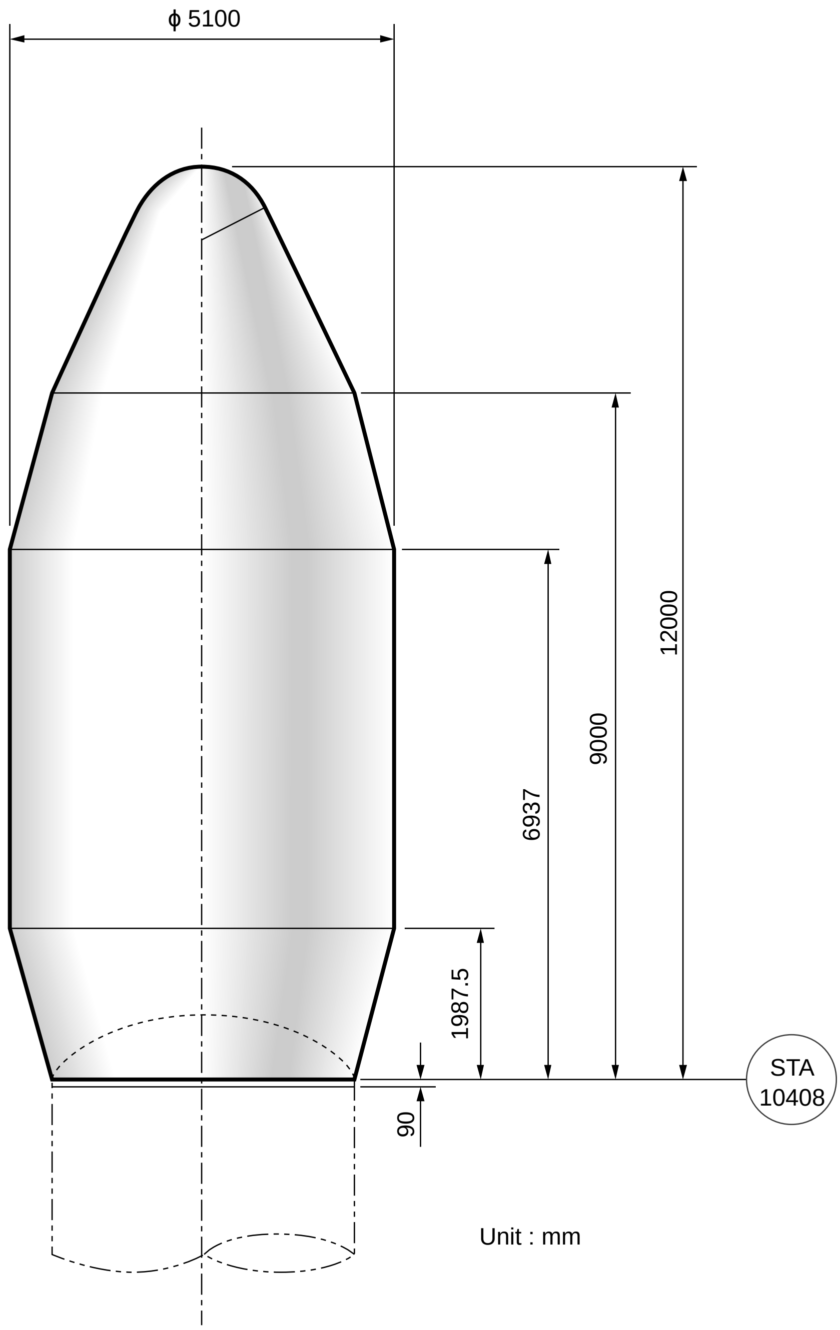


Figure 4.5-4 Model 5S

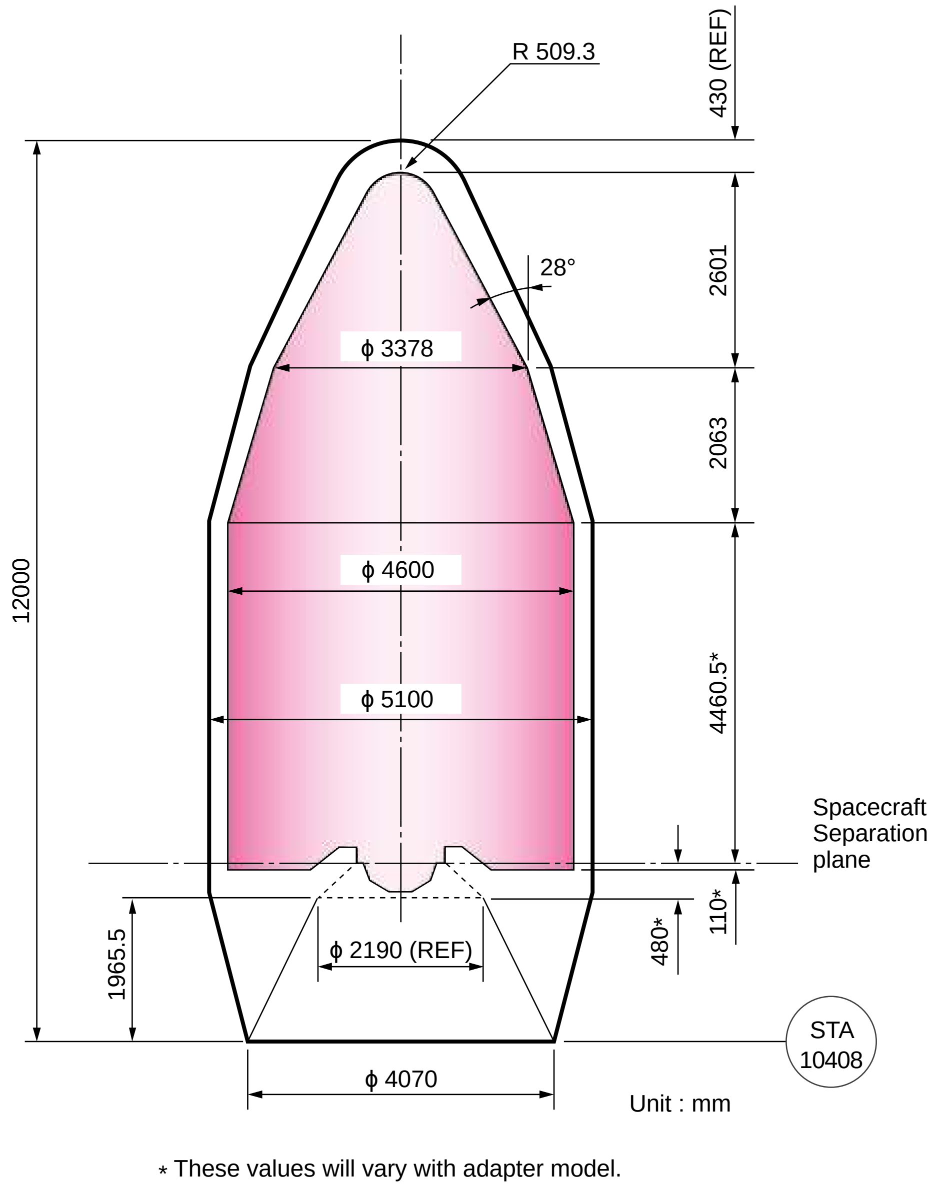


Figure 4.5-5 Usable volume of Model 5S

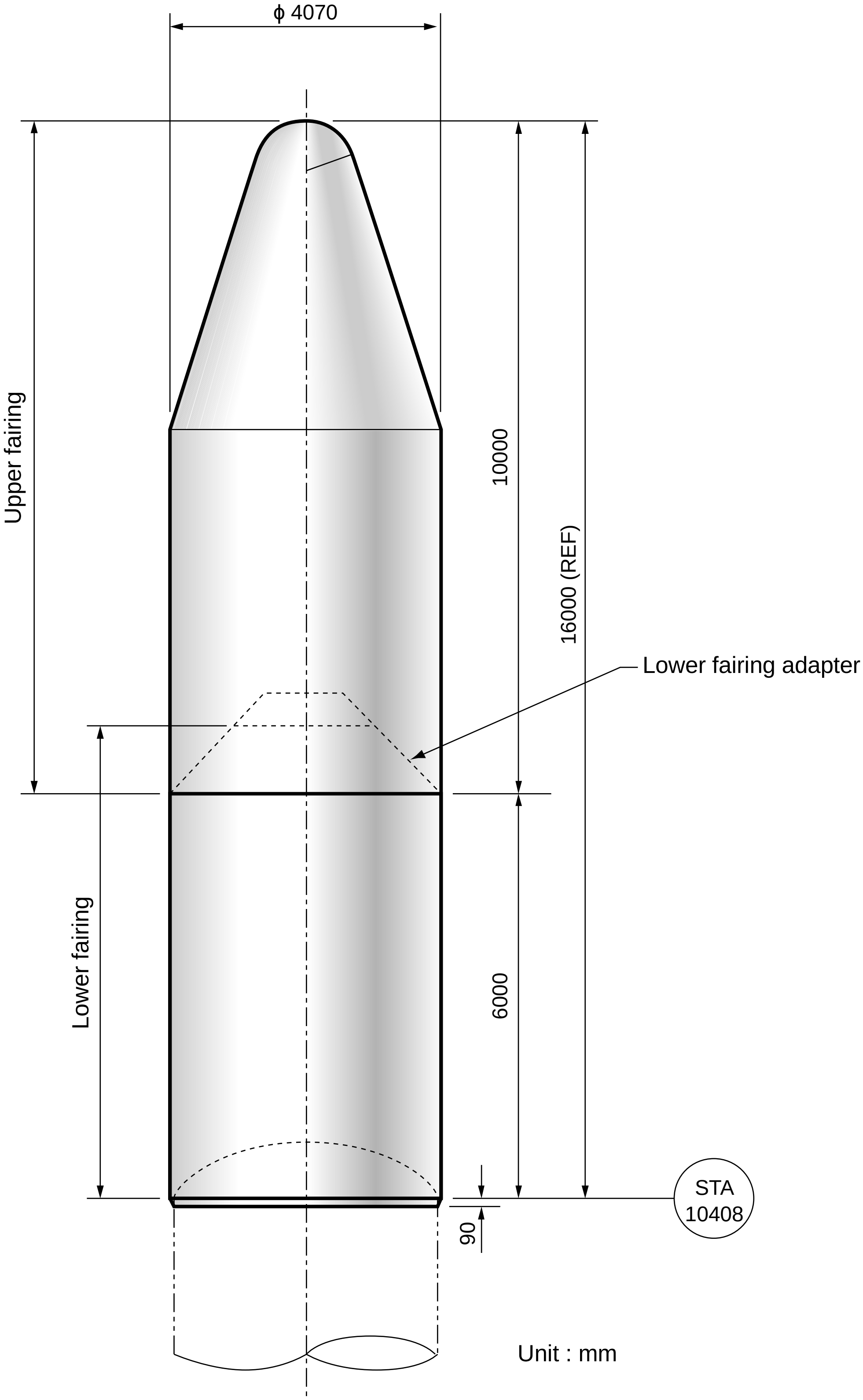
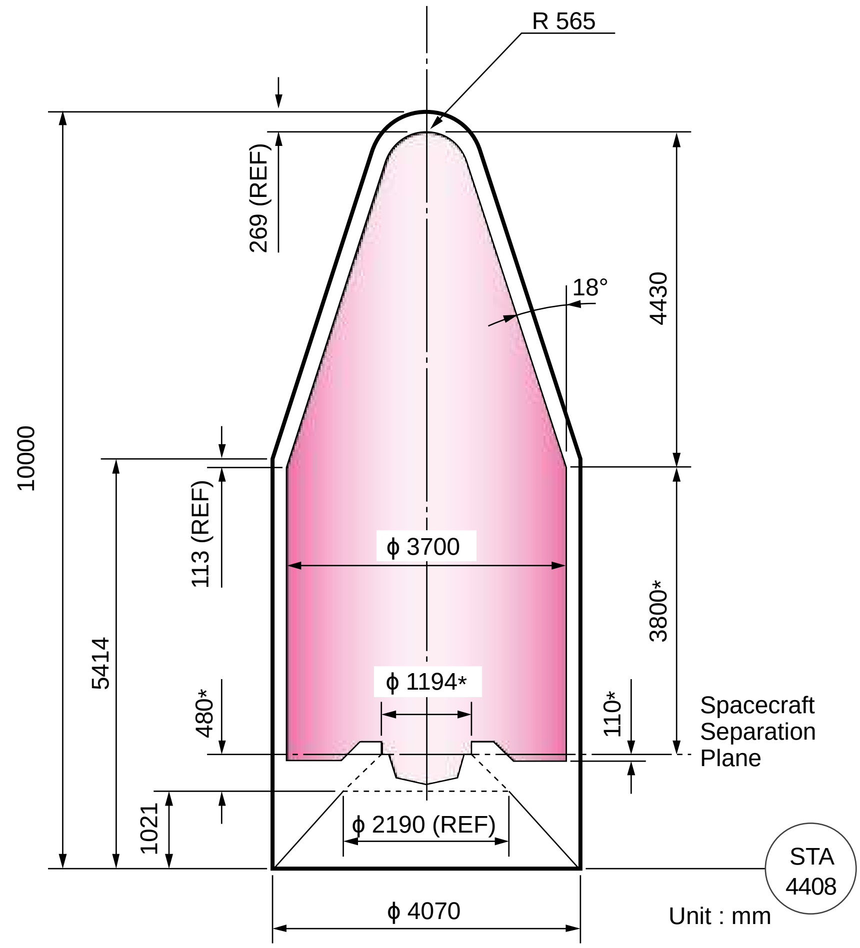
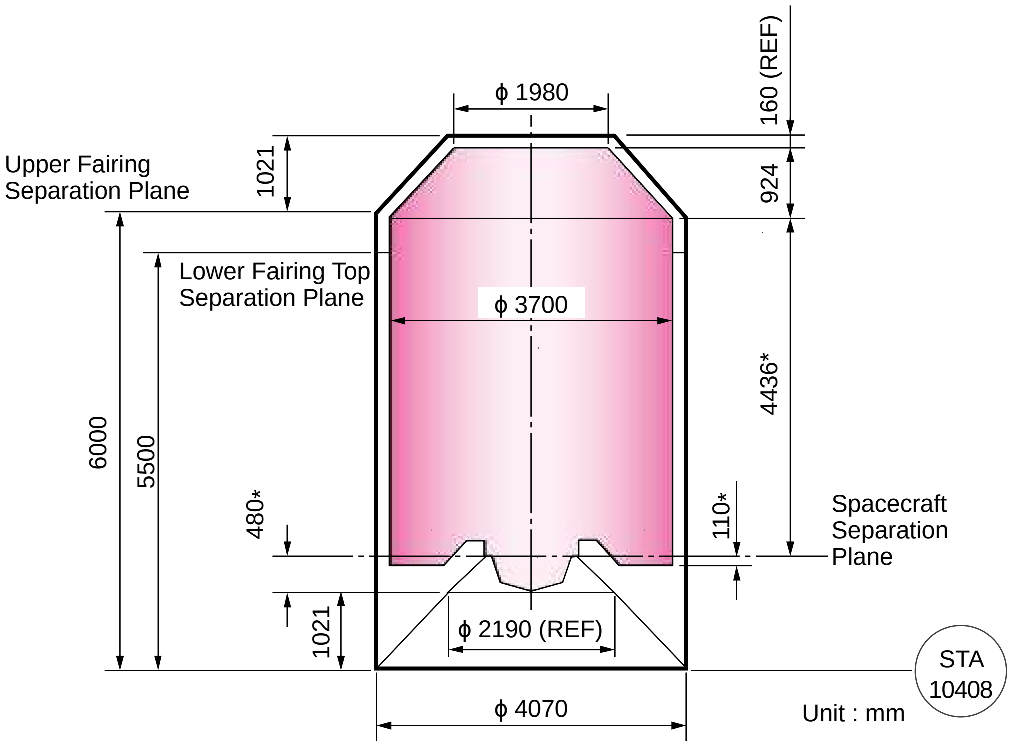


Figure 4.5-6 Model 4/4D-LC



* These values will vary with adapter model.

Figure 4.5-7 Usable volume of 4/4D-LC upper fairing



* These values will vary with adapter model.

Figure 4.5-8 Usable volume of 4/4D-LC lower fairing

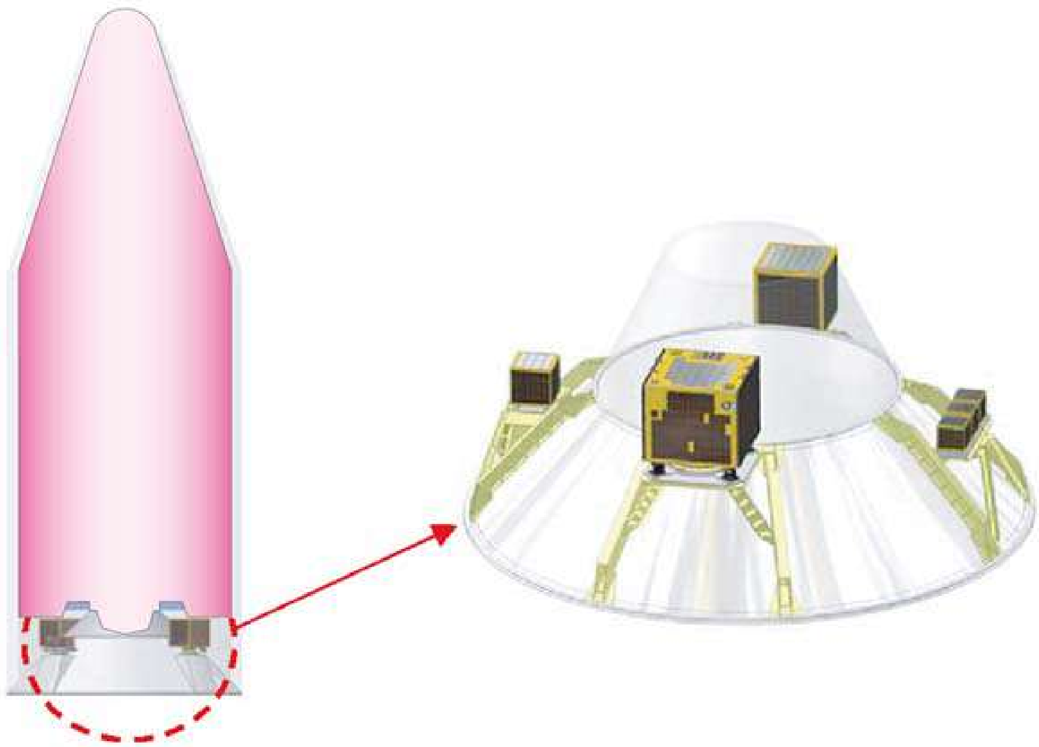


Figure 4.5-9 Accommodations of piggy-back spacecraft

4.6 Payload Adapter

4.6.1 Adapter types

Table 4.6-1 shows characteristics of the payload adapters for the H-IIA launch vehicle. Detailed information related to the payload adapter is presented in Appendix 4.

(1) “1194” stands for that the interface diameter at the top of the adapter is 1194 mm.
(2) “M” stands for that the adapter has a clamp band separation mechanism.
(3) “LS-H” stands for that the adapter has a low shock clamp band mechanism.

If SCO desires another familiar adapter which is not listed in Table 4.6-1, LSP can procure SCO requested adapter. Please contact LSP for details.

4.6.2 Mechanism of payload adapter

The payload adapter mechanism is described below.

4.6.2.1 Separation mechanism

The H-IIA launch vehicle provides clamp bands as the separation mechanism on the adapter. The V flange of the rear frame of the spacecraft is connected to the top front of the adapter, and bound by clamp bands with concave V shaped blocks. At the spacecraft separation, bolts which connect the clamp bands are cut by pyrotechnic bolt cutters. For the low shock adapter, the clamp bands are released by non-explosive or slow-released separation device. The clamp band tension is designed to ensure no gaps exist between the spacecraft and the adapter frames on the ground and in flight environments.

4.6.2.2 Ejection mechanism

Separation springs are installed in the adapter to separate the spacecraft from the launch vehicle. When the separation springs are installed, static loads act on each mating structure as limit load, in addition to the acceleration load described in Section 3.2.1.

4.6.2.3 Spacecraft separation monitoring switches

The standard launch vehicle has two (2) micro-switches on top of the adapter to monitor spacecraft separation. It is possible to locate the actuator pad of the spacecraft micro-switch on top of the adapter. In addition to the separation monitor switch, video camera is equipped on the PSS below the payload adapter to ensure the monitor of the spacecraft separation. Video signals are received on ground station. Spacecraft separation motion pictures are provided to SCO at TNSC.

4.6.3 Mission modifications

4.6.3.1 Separation springs

Separation springs are installed inside the adapter to force the spacecraft away from the launch vehicle.

a) Number of separation springs : 4 to 8

4.6.3.2 Umbilical connectors

Umbilical connectors are installed on an adapter. Details are described in Section 4.7.6.

a) Number of umbilical connectors : 2

b) Location of umbilical connectors : Decided by discussion with SCO

c) Specification of umbilical connectors : Described in Section 4.7.6.4

Table 4.6-1 Characteristics of payload adapters

Items / Model	Spacecraft Interface Diameter(mm)	Height (mm)	Mass (kg)	Connecting Device	Separation Device	Note
1194M	1215	480	100	Clamp band	Pyrotechnic Separation Device	
1666MA	1666	480	100	Clamp band	Pyrotechnic Separation Device	
937LS-H	945	630	100	Clamp band	Non-Explosive Separation Device	
1194 LS-H	1215	480	100	Clamp band	Non-Explosive Separation Device	
1666 LS-H	1666	480	100	Clamp band	Non-Explosive Separation Device	

4.7 Electrical and RF Interfaces

4.7.1 General

This section describes electrical interfaces such as electrical bonding, electrical command interfaces from the launch vehicle; specifications of interface connector (umbilical connector); and RF link between spacecraft and spacecraft GSE, etc.

4.7.2 Electrical bonding

The spacecraft and launch vehicle must maintain the same electrical potential in flight, for which bonding is necessary. Grounding (or earthing) is needed to ensure safety or protection of electrical equipment and human by discharging the electrical energy to the earth on the ground. LV provides the conductive path for grounding the SC.

The bonding which is the act of joining spacecraft and launch vehicle electrically is described in this section. The bonding reference point should be located close to the separation plane of the spacecraft and launch vehicle, where the bonding should be provided by mechanical contact of both sides at connection. MIL-STD-464 class S (less than 1Ω) resistance requirement is applied to the H-IIA launch vehicle bonding. Therefore, surface finish of the spacecraft separation structure should satisfy the above requirement.

4.7.3 Command and power interfaces

Table 4.7-1 shows the electrical interfaces which the H-IIA launch vehicle provides to the spacecraft.

4.7.3.1 Pyrotechnic command

The H-IIA launch vehicle can provide pyrotechnic commands to the spacecraft for the following two functions:

(1) Spacecraft separation pyrotechnic command

When the spacecraft organization provides a separation mechanism, the launch vehicle system can provide pyrotechnic commands for spacecraft separation. Details of design conditions, etc., are to be determined at the technical interchange meeting (TIM).

This command is provided as a standard service when the spacecraft organization provides a separation mechanism.

(2) Other spacecraft related pyrotechnic commands

If the spacecraft system needs other commands, the launch vehicle system can provide additional command as an option. The launch vehicle system can provide 10 command signals in total including other pyrotechnic commands, electrical commands and dry loop commands for a dual launch. For dual launch, the launch vehicle hardware spare channels are further limited, so special arrangements must be made at the TIM.

Main electrical characteristics of the pyrotechnic command are the following and the wiring diagram for the pyrotechnic command is shown in Figure 4.7-1.

(a) Source	Battery
(b) Voltage	28 (+6, -4) VDC
(c) Ignition timing	To be determined at the TIM
(d) Pulse width of igniting signal	To be determined at the TIM
(e) Minimum igniting current	To be determined at the TIM
(f) Recommended igniting current	To be determined at the TIM
(g) Number of power cartridges	To be determined at the TIM
(h) Non-igniting current	To be determined at the TIM
(i) Bridge wire resistance	To be determined at the TIM
(j) Insulation resistance	To be determined at the TIM
(k) Insulation resistance after ignition	To be determined at the TIM

4.7.3.2 Electrical command (discrete signal)

(1) SCO can use electrical commands (discrete signals) from the launch vehicle, if necessary. Total number of command signals is as described in Section 4.7.3.1. For a dual launch, if both spacecraft require these signals, special arrangements must be made at the TIMs. These commands are an optional service to the spacecraft.

(2) Launch vehicle wiring diagram for the electrical commands is shown in Figure 4.7-2.

(3) Main electrical characteristics of the electrical command are as follows,

(a) Voltage	28 (+6, -4) VDC
(b) Load resistance	To be determined at the TIM
(c) Supply current	To be determined at the TIM
(d) Insulation resistance	To be determined at the TIM
(e) Supply timing	To be determined at the TIM
(f) Supply time (duration)	To be determined at the TIM
(g) Spacecraft circuit condition	Insulated from ground and structure

Design details will be coordinated at the TIM.

4.7.3.3 Dry loop command

(1) SCO can use dry loop commands from the launch vehicle, if necessary. Total number of command signals is as described in Section 4.7.3.1. For dual launch, if both spacecraft require these signals, special adjustment is required at the TIM. These commands are an optional service to the spacecraft.

(2) Wiring diagram of the electrical commands is shown in Figure 4.7-3.

(3) Main electrical characteristics of the electrical command are as follows,

(a) Supply timing/duration	To be determined at the TIM
(b) Resistance (open/closed)	To be determined at the TIM
(c) Voltage/Current(from spacecraft)	To be determined at the TIM
(d) Insulation resistance	To be determined at the TIM

Design details will be coordinated at the TIM.

4.7.3.4 Power supply

If SCO requires external power supply, the launch vehicle can provide it to the extent which depends on the mission. For more information regarding the in-flight power supply, please contact LSP.

4.7.4 In-flight telemetry

SCO can use the transmission of spacecraft measurement data by the L/V telemetry, if necessary. For more information regarding the in-flight telemetry, please contact LSP.

4.7.5 Umbilical interface

4.7.5.1 Umbilical lines for dedicated launch mission

The umbilical interface block diagram between the spacecraft and the ground facilities for a dedicated launch is shown in Figure 4.7-4.

Spacecraft is connected to spacecraft GSE at ML GSE room through payload adapter, payload fairing, umbilical, umbilical mast by copper lines. These hard lines are available for electrical power supply, monitoring spacecraft system, telemetry transmission and command receipt. ML GSE room and STA2/SFA2 are connected by 100Mbps single-mode fiber lines and 10/100Mbps copper lines. These umbilical facilities make it possible to remote-control and tele-communicate the spacecraft from STA2/SFA2 during prelaunch and launch countdown operations.

The interface connectors with the spacecraft are DBAS connectors. The interface connectors at ML GSE room are shown in Table 4.7-2. The specifications for interface connectors with the spacecraft (the spacecraft umbilical connector) are defined in the Interface Control Document (ICD) for spacecraft. The maximum number of connector pins is 120.

4.7.5.2 Umbilical lines for dual launch mission

The umbilical interface block diagram between the spacecraft and the ground facilities for a dual launch is shown in Figure 4.7-5.

The specifications of interface connectors for the spacecraft in the upper fairing and lower fairing are same for a dedicated launch mission in Section 4.7.5.1.

4.7.5.3 Spacecraft validation test configuration

The spacecraft validation test configuration is shown in Figure 4.7-6. Typically, SCO is allowed to check out the propellant loaded spacecraft from the remote building. The copper lines between spacecraft and GSE are prepared by SCO. The copper lines can be used for both Spacecraft/PLA composite configuration and Spacecraft/PLA/PSS composite configuration.

4.7.6 Interface connectors between spacecraft and launch vehicle

This section specifies the interface connectors between the spacecraft and the H-IIA launch vehicle.

LSP will use the DBAS connector as a standard connector shown in Section 4.7.6.4. If SCO desires to use other interface connectors or interface connectors via payload fairing, please contact LSP.

4.7.6.1 Interface connector procurement responsibility

Interface connector receptacles (the spacecraft umbilical connectors) which are installed on the spacecraft will be prepared by SCO and plugs will be prepared by LSP, respectively. However, connector procurement responsibility can be arranged per SCO's request.

4.7.6.2 Interface connectors for dedicated launch

2 interface connectors are used for electrical signals and electrical power supply of the spacecraft. Interface connectors of launch vehicle are installed on the payload adapter. Interface connectors via payload fairing are option.

LSP can install dry loop in the spacecraft connectors and fairing connectors. SCO can use dry loop for additional sensing device of spacecraft separation, fairing separation. For more information regarding the connectors specifications, please contact LSP.

Figure 4.7-7 shows interface connectors for a dedicated launch.

4.7.6.3 Interface connectors for dual launch

Interface connectors for a dual launch will be the same as dedicated launch for the upper spacecraft. Interface connectors via payload fairing are limited for upper spacecraft, and not used for the lower spacecraft.

Figure 4.7-8 and Figure 4.7-9 show interface connectors for a dual launch.

4.7.6.4 Standard interface connector specifications

The standard interface connector specifications should conform to the requirements of ESA/ ESCC SPEC. No. 3401/008, contacts SPEC. No. 3401/009. These connectors are manufactured by TE Connectivity (DEUTSCH).

4.7.6.5 Other interface connector characteristics

(1) Standard interface connector via PLA

2 interface connectors are located 180° opposite to each other on the PLA. The receptacle surface should face the separation plane. The key position of interface connectors is set toward

the external radial direction of the H-IIA launch vehicle.

(2) Interface connector via payload fairing (Option)

(a) Disconnecting angle of the plugs: The disconnecting angle of the H-IIA launch vehicle plugs with lanyard shall be within ±10° at separation from spacecraft receptacles.

(b) Location: The interface connectors shall normally be placed within ±5° from the axis III or I.

(c) Angle: The connectors shall be arranged within ±2° between the connector face and the axis II - IV of the H-IIA launch vehicle.

(d) Pulling direction: Nothing shall be located within ±15° of the pulling direction.

(e) Key position: The key position of interface connectors is set upward direction of the H-IIA launch vehicle.

If SCO requires the interface connectors via payload fairing, please contact LSP.

4.7.7 RF Link Interface

4.7.7.1 General

LSP will provide the RF link system for S-band, C-band, X-band, and Ku-band telemetry/ command between the spacecraft on the H-IIA launch vehicle and SCO's GSE. The interface conditions are defined as follows:

(1) Link path

(a) Between LP and Spacecraft Checkout room (see Figure 4.7-12)

(b) Between VAB and Spacecraft Checkout room (see Figure 4.7-10)

(c) Between SFA/SFA2 and Spacecraft Checkout room

(2) Option path(S-band only)

(a) During the ML roll-out from VAB to LP (see Figure 4.7-11)

(b) Between the spacecraft and Masuda Tracking and Communication Station (MTCS)

(3) Equipment

(a) Single equipment for C-band, X-band, and Ku-band (compatible with dedicated mission)

(b) Two equipment for S-band (compatible with both dedicated mission and dual mission)

(4) Antennas

One of these antennas can be selected when communicating with the spacecraft at LP from Spacecraft Checkout room.

(a) Fairing internal antenna

(b) ML umbilical mast antenna

(5) RF link facilities interface characteristics

RF link facilities interface characteristics are shown in Table 4.7-3.

4.7.7.2 RF link with ML/STA2 or SFA2

RF link telemetry/command between the spacecraft and STA2 or SFA2 is available during prelaunch and launch countdown operations.

(1) RF link when ML is in VAB or on Launch Pad (LP)

When the ML is in the VAB or on the LP, RF link system between ML and checkout room in STA2 or SFA2 is provided through fiber lines and RF/optical transceiver. RF/optical transceivers are installed both in the checkout room at STA2 or SFA2 and in the ML GSE room. RF link system can be operated remotely from SCO's GSE at STA2 or SFA2. (see Figure 4.7-10 and Figure 4.7-12) The following 2 routes are selectable in remote from STA2, SFA2 and ML GSE room.

- Fairing internal antenna and RF coaxial umbilical cable
- Air link with ML umbilical mast* horn antenna via fairing RF transparent window

*) LOX umbilical mast, it stands near launch vehicle I-axis side.

The RF link schematic in VAB (STA2 case) is shown in Figure 4.7-10. The RF link schematic on LP is shown in Figure 4.7-12. Two S-band RF link equipment are available for dual launch. Other band RF link equipment is available for dedicated launch.

(2) Monitor during ML roll-out (option)

(a) Monitor during ML roll-out

During ML roll-out, spacecraft and its GSE are power-off in standard operations. If the spacecraft status needs to be monitored during the ML roll-out, the spacecraft and its GSE can be power-on and SCO personnel can stay in ML GSE room to monitor the spacecraft status.

(b) Monitor during ML roll-out by S-band

During ML roll-out from VAB to LP, fiber optic network is not available. The S-band RF link during ML roll-out can be established by using RF relay equipment via the air. STA2 or SFA2 from relay equipment are connected via fiber lines (see Figure 4.7-11). The route from STA2 or SFA2 is switched at Yoshinobu Block House (B/H) after ML roll-out begins. RF relay equipment is a single set, and only usable for single spacecraft. For more information regarding the RF link system, please contact LSP.

4.7.7.3 RF Link with MTCS (for JAXA’s spacecraft)

RF link in S-band between the spacecraft and Masuda Tracking and Communication Station (MTCS) can be provided for JAXA’s spacecraft.

(1) When ML is in VAB

The spacecraft in the VAB can be linked, through VAB RF link unit, with the MTCS via Air link.

(2) During ML roll-out

The spacecraft can be linked directly from MTCS via RF transparent windows.

(3) After ML is on LP

The spacecraft can be linked directly from MTCS via RF transparent windows (same as (2) above).

4.7.8 Electrical and RF requirements

4.7.8.1 Electrical requirements

The spacecraft organization shall satisfy the following constraints in the final preparation phase up to liftoff.

(1) The spacecraft organization shall design the spacecraft so that the umbilical cable carries only low current signals at liftoff. Recommended voltage and current are 28 VDC, and less than 10 mA.

(2) The spacecraft power supply shall be switched from external to internal, and the ground power supply shall be switched off at about 5 minutes before liftoff. Details are coordinated at the TIM.

4.7.8.2 RF requirements

See section 3.6.

Table 4.7-1 Electrical interfaces

Item	Interface	Description	Related section
Command Interface	Electrical interface DBAS type connector	(1) Discrete command --- option (8ch) (2) Dry loop command --- option (2ch)	4.7.3.1 4.7.3.2 4.7.3.3
Telemetry interface	Electrical interface DBAS type connector	Spacecraft measurement data --- option	4.7.4
Umbilical interface	Electrical interface DBAS type connector	(1) Dedicated launch Wire quantity : 120 wires (2) Dual launch Wire quantity : 120 wires (each SC)	4.7.5.1 4.7.5.2

Table 4.7-2 ML payload GSE room umbilical interface connector characteristics

Connector No.			Connector Type	Note
Spacecraft side	GSE room #1 (for Lower fairing spacecraft)	GSE room #2 (for Upper fairing or dedicated spacecraft)	GSE room #1 and #2	
UMB1	J1311	J2311	MS3452L36-10SN	See Figure 4.7-4 and 4.7-5.
	J1312	J2312	MS3452L36-10SW	
UMB2	J1313	J2313	MS3452L36-10SX	
	J1314	J2314	MS3452L36-10SY	

Table 4.7-3 Spacecraft and TNSC RF link facilities Interface characteristics

Item	Band				Note
Command	S-band	C-band	X-band	Ku-band	
Frequency	2025～2120 MHz	5900～6400 MHz	7145～7235 MHz	12.75～18.0 GHz	
Power level at Spacecraft GSE	-20 dBm	-20 dBm	-20 dBm	-20 dBm	from GSE to Facilities
Power flux density at Spacecraft antenna	N/A	Below -80 dBW/m^2	N/A	Below -80 dBW/m^2	
Effective receiver sensitivity (Power level) at Spacecraft antenna	Below -90 dBm	Below -87.6dBm @6.4GHz *	Below -90 dBm	Below -96.6dBm @ 18GHz *	*) Typical frequency
Telemetry	S-band	C-band	X-band	Ku-band	Note
Frequency	2200～2300 MHz	3700～4200 MHz	8400～8500 MHz	11.00～12.75 GHz	
Spacecraft EIRP	30 dBm	40 dBm	33 dBm	40 dBm	
Power level at Spacecraft GSE	-40 dBm	-10 dBm	0 dBm	-10 dBm	from Facilities to GSE

Connector type of RF equipment facilities side: Type N Female

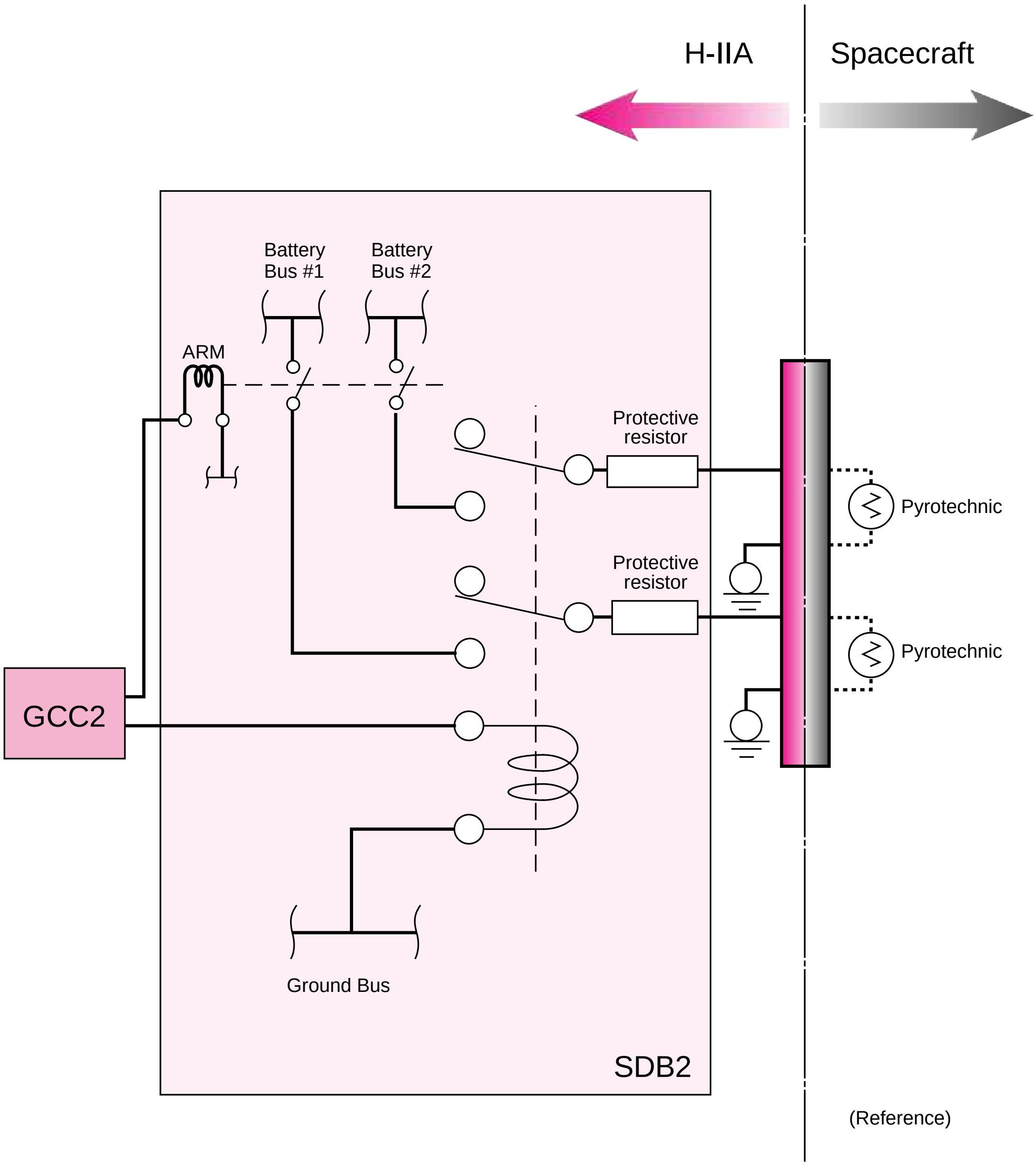


Figure 4.7-1 Pyrotechnic command wiring diagram

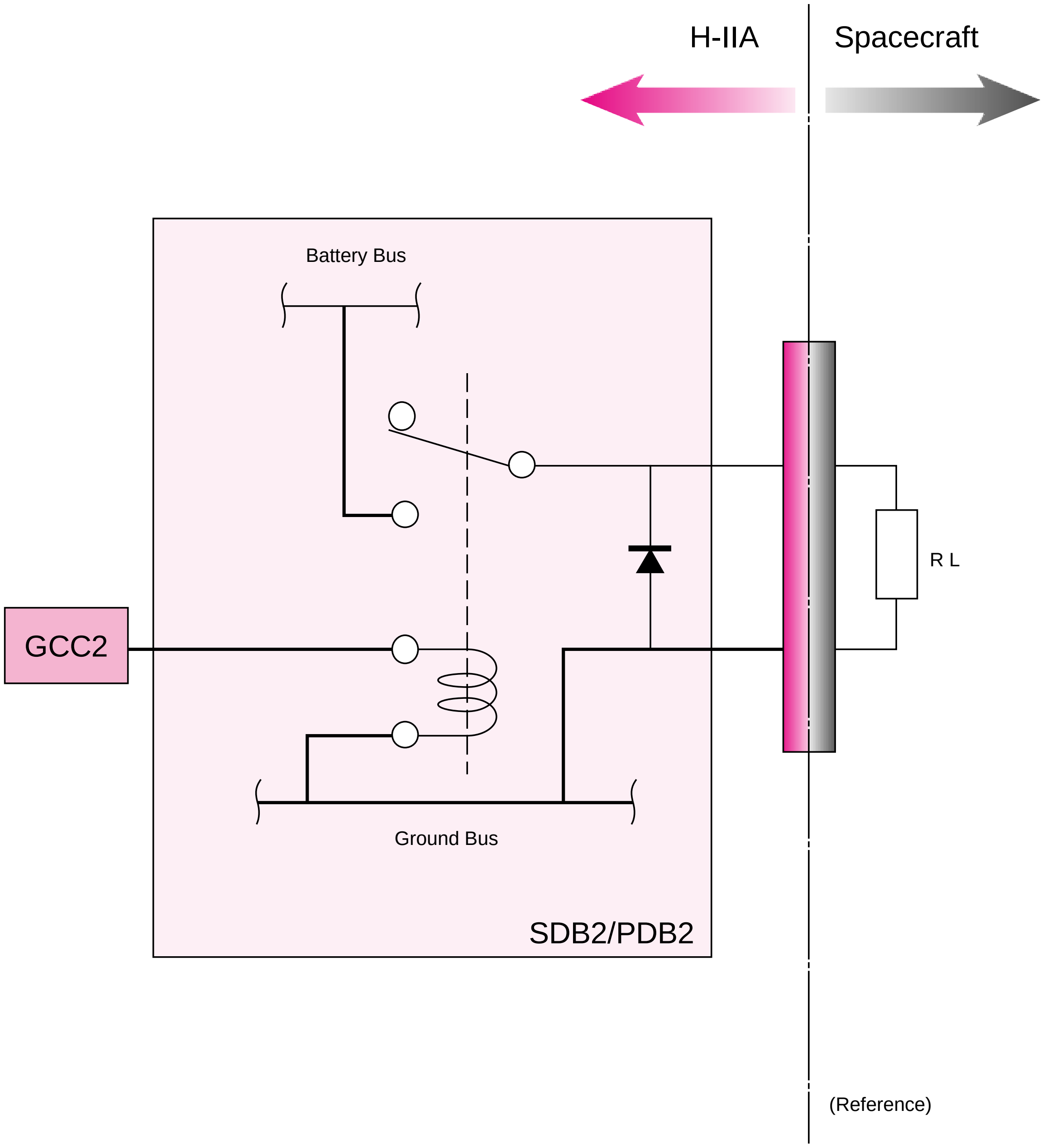


Figure 4.7-2 Electrical command wiring diagram

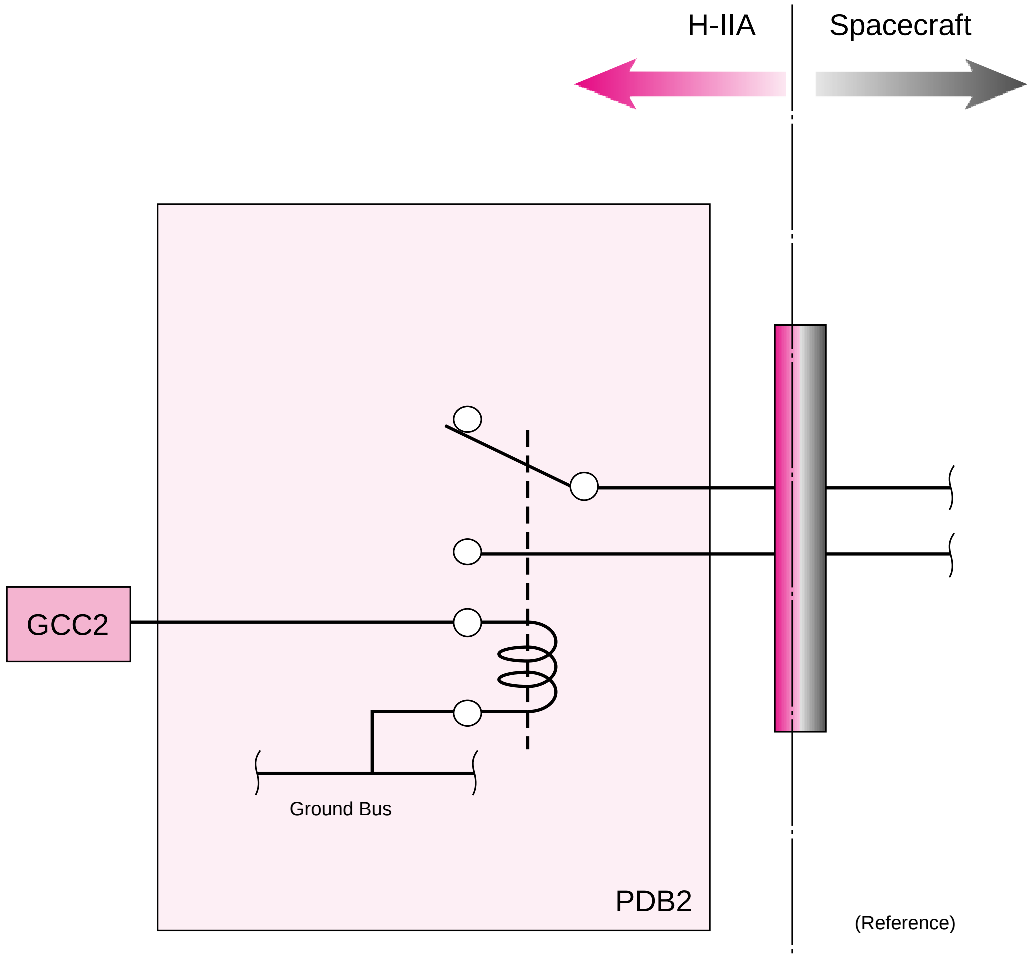
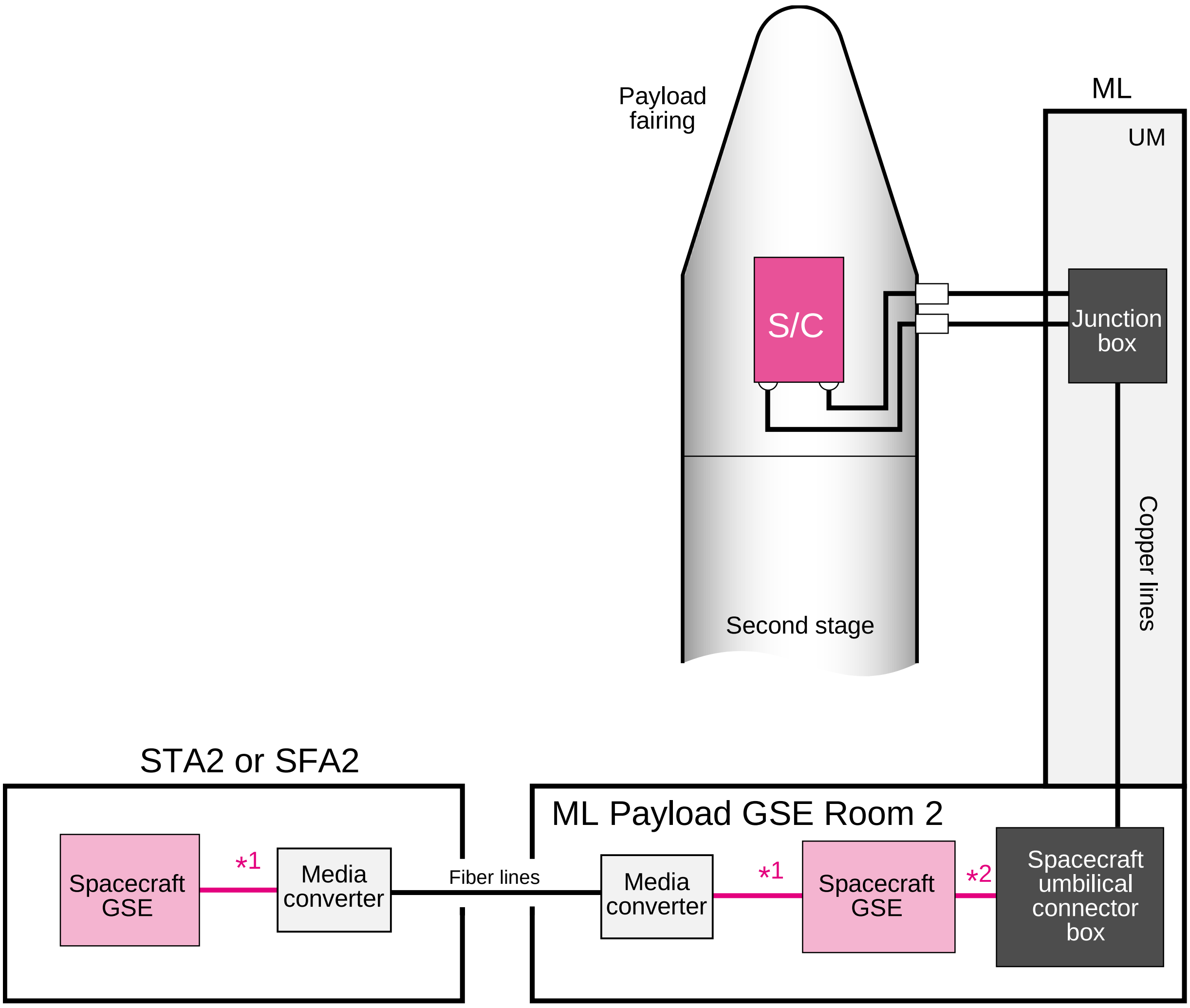
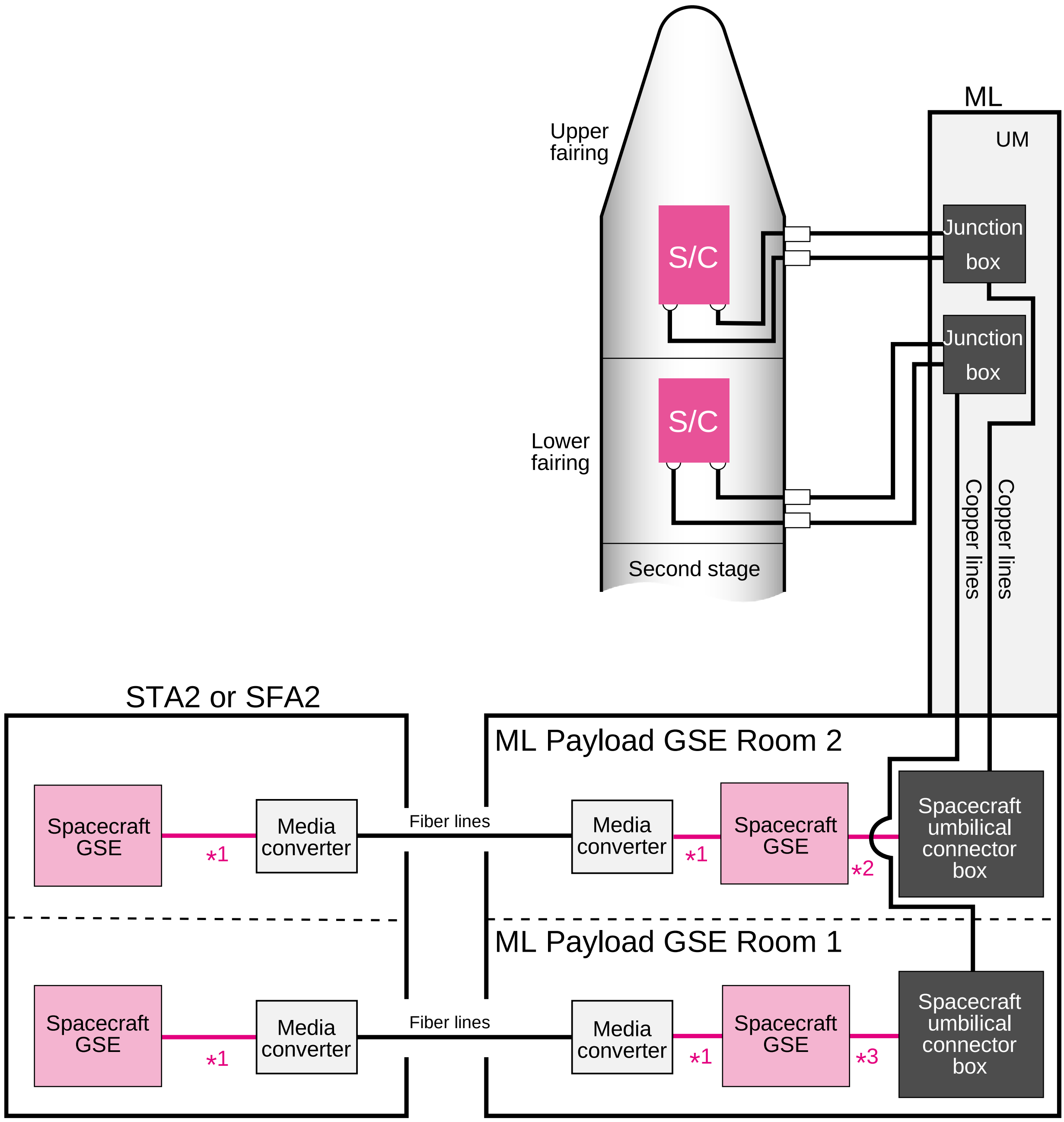


Figure 4.7-3 Dry loop command wiring diagram



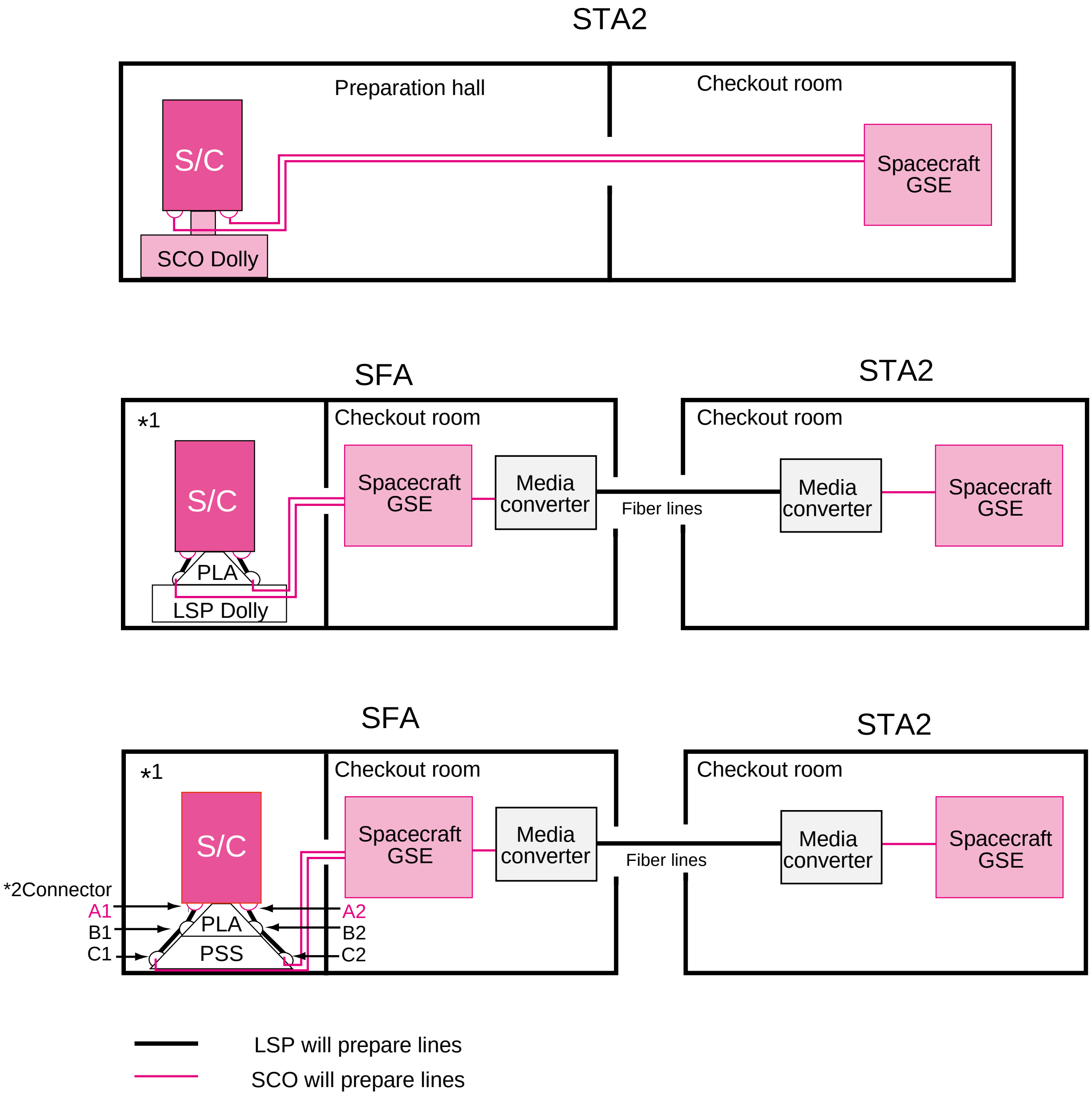
*1) SCO prepares 10 Base-T lines.
*2) SCO prepares parallel copper lines, Connector type (J2311-2314) is shown in Table 4.7-2.

Figure 4.7-4 Umbilical interface for dedicated launch



*1) SCO prepares 10 Base-T lines.
*2) SCO prepares parallel copper lines, Connector type (J2311-J2314) is shown in Table 4.7-2.
*3) SCO prepares parallel copper lines, Connector type (J1311-J1314) is shown in Table 4.7-2.

Figure 4.7-5 Umbilical interface for dual launch



*1) Spacecraft Fairing Assembly Hall.
Typically, SCO is allowed to check out the propellant loaded spacecraft from the remote building.
*2) Typically, A1,B1,C1 connectors (receptacles) specifications are the same.
Typically, A2,B2,C2 connectors (receptacles) specifications are the same.

Figure 4.7-6 SC validation test configuration

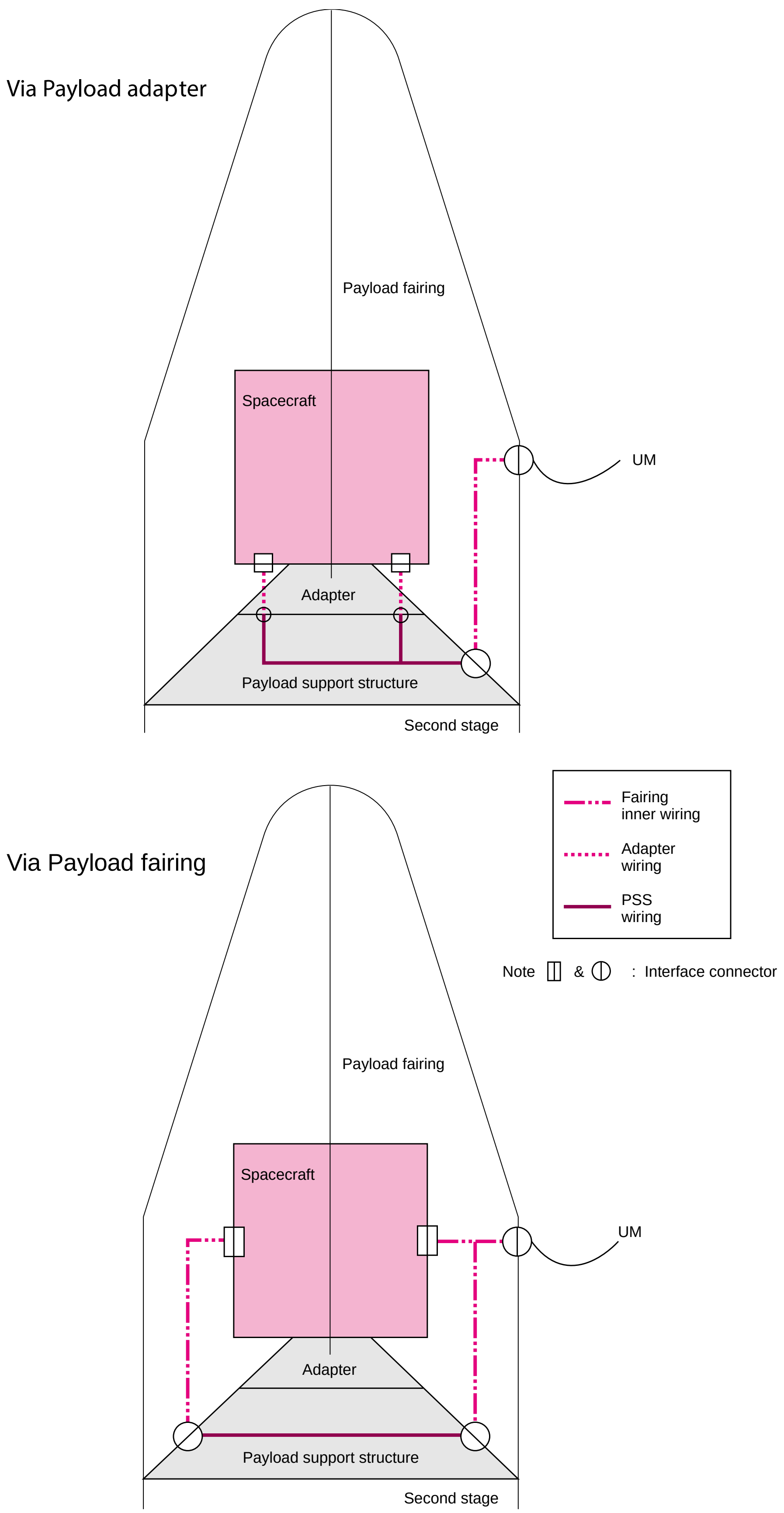


Figure 4.7-7 Interface connectors for dedicated launch

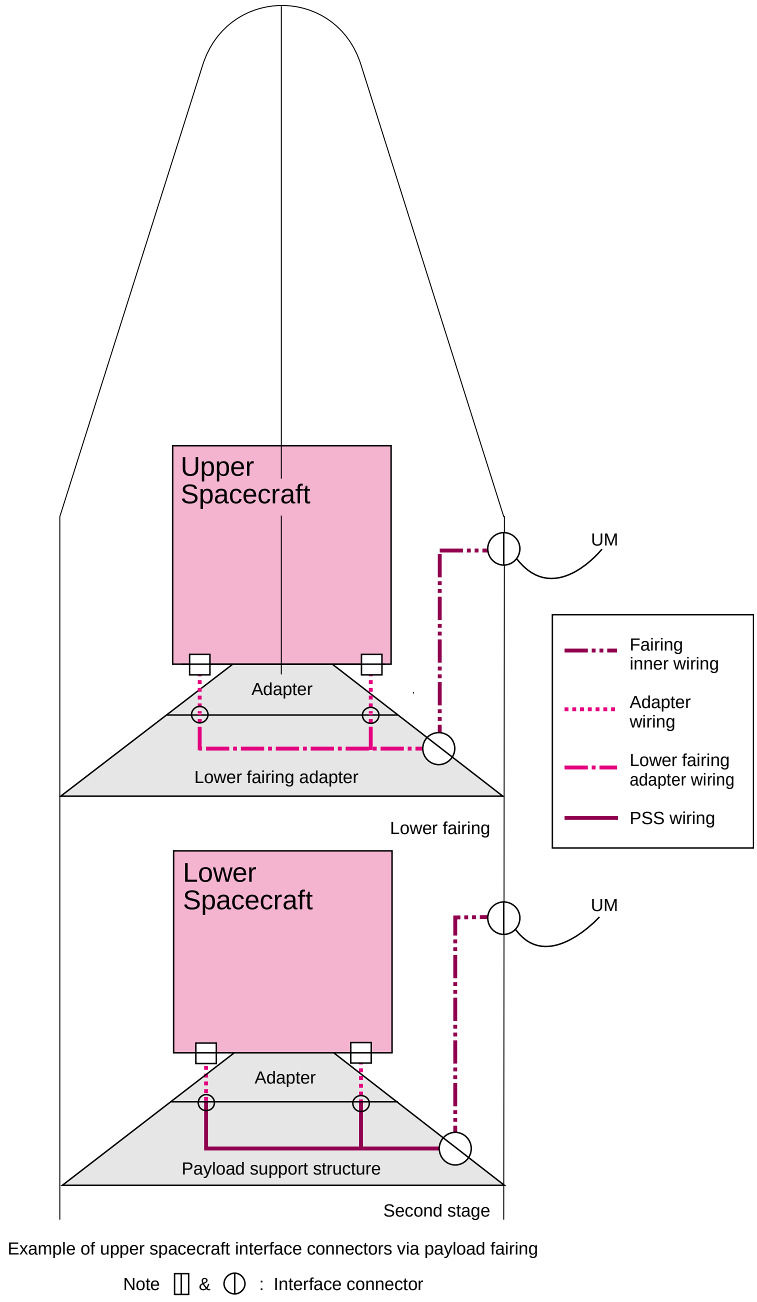
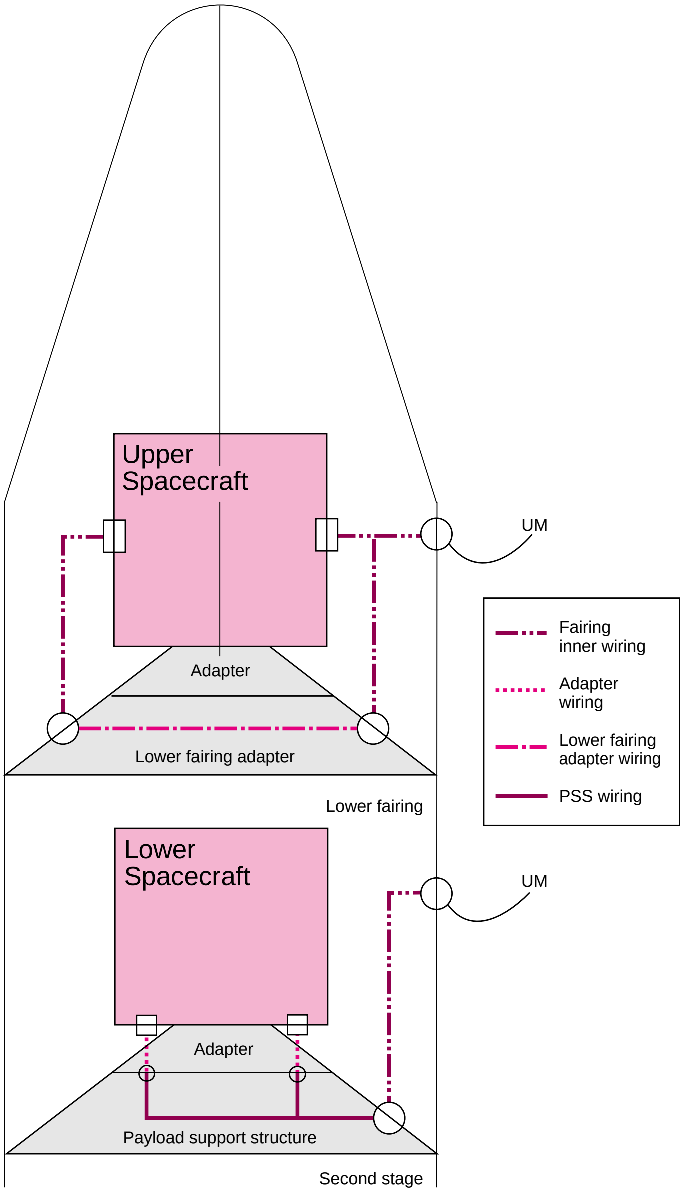


Figure 4.7-8 Interface connectors for dual launch (via payload adapter)

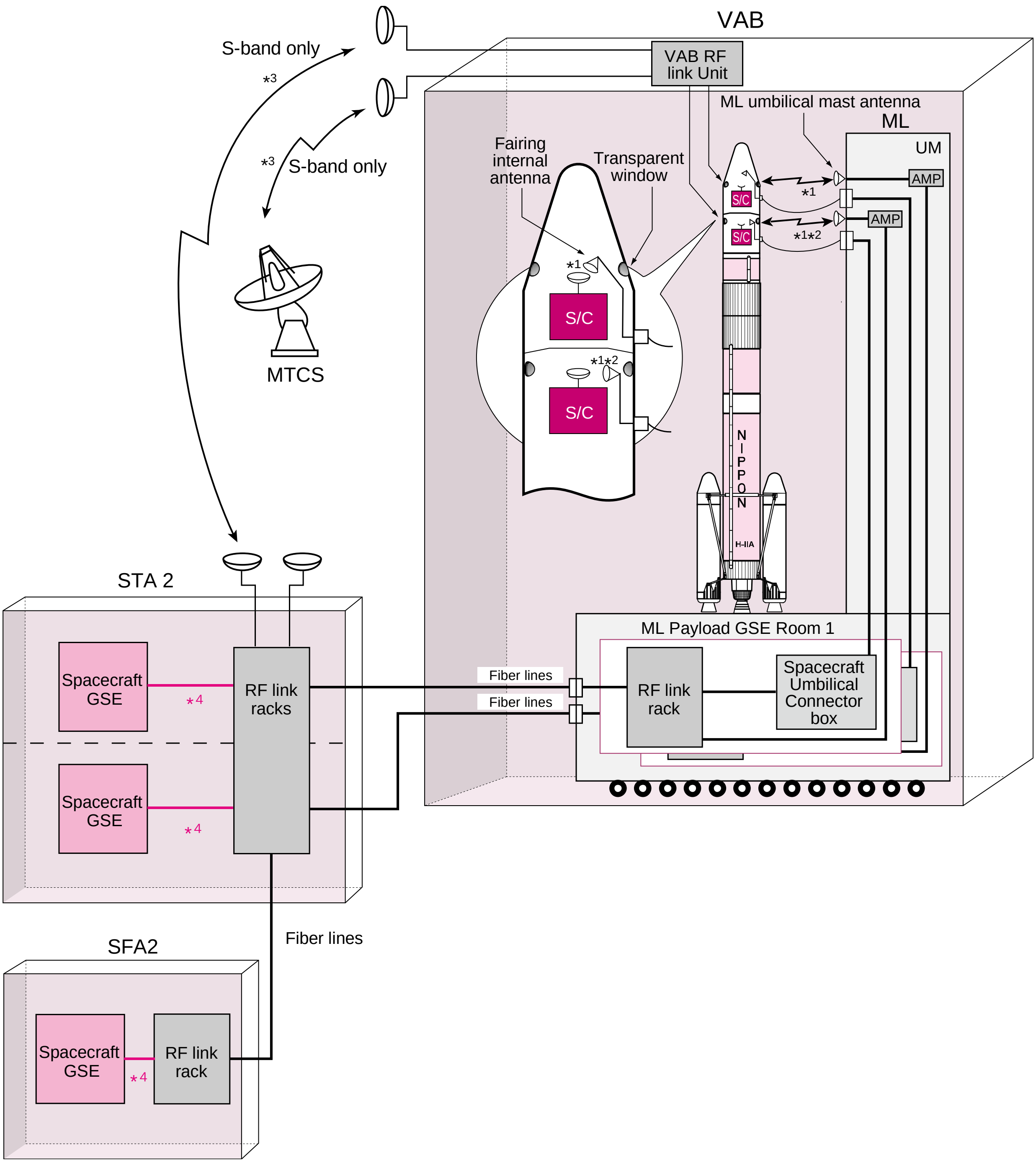


Example of upper spacecraft interface connectors via payload fairing

Note ◫ & ⦶ : Interface connector

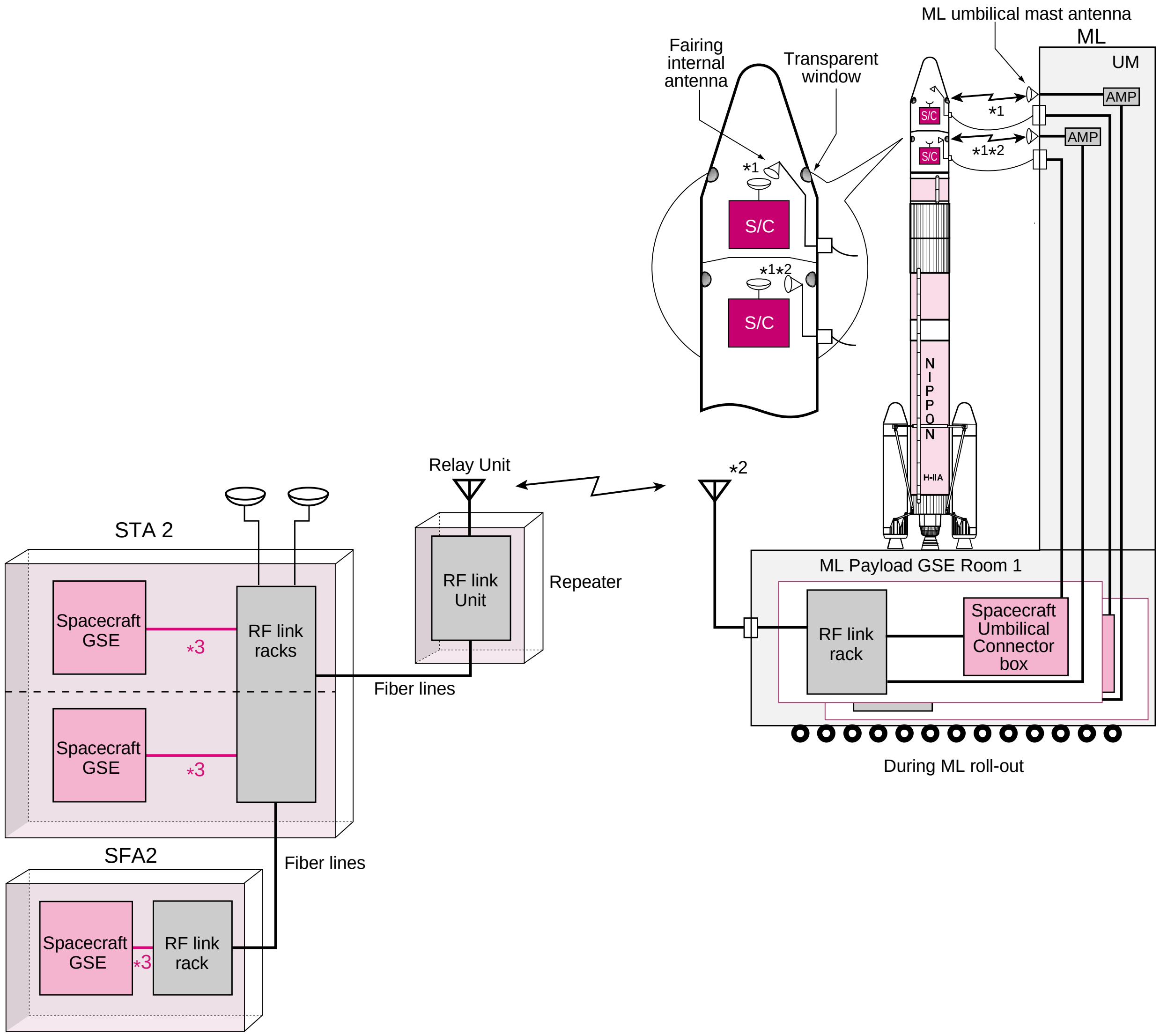
Note) Interface connectors via payload fairing are limited for upper spacecraft.

Figure 4.7-9 Interface connectors for dual launch (via payload fairing)



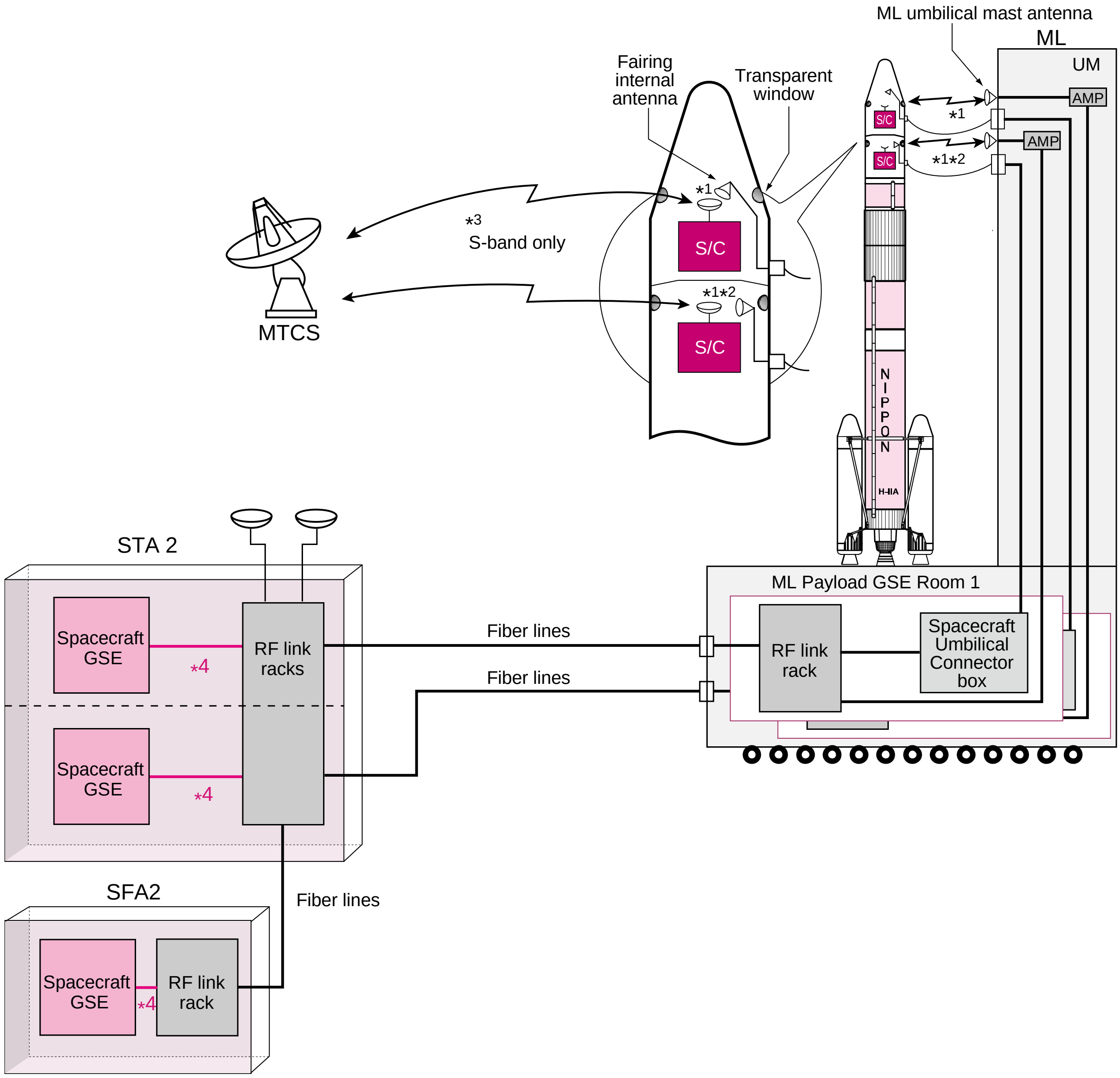
<Note>
*1 : Either ML umbilical mast antenna or fairing internal antenna is selectable in remote.
*2 : Single equipment for C-band, X-band, Ku band. Dual equipment for S-band.
*3 : S-band air link between VAB and STA2, MTCS is only for either one of the spacecraft.
*4 : SCO prepares co-axial line.

Figure 4.7-10 VAB RF link schematic



<Note>
*1 : Either ML umbilical mast antenna or fairing internal antenna is selectable in remote.
*2 : RF link is only for either one of the spacecraft.
*3 : SCO prepares co-axial line.

Figure 4.7-11 S-band RF link schematic during ML roll-out



*1 : Either ML umbilical mast antenna or fairing internal antenna is selectable in remote.
*2 : Single equipment for C-band, X-band, Ku band. Dual equipment for S-band.
*3 : S-band air link between ML and MTCS is only for either one of the spacecraft.
*4 : SCO prepares co-axial line.

Figure 4.7-12 Launch pad RF link schematic

4.8 Other Ground Equipment Interfaces

4.8.1 Power

TNSC has AC power generators. In launch campaign, AC power is provided by TNSC AC generators. Backup generators are in standby status for another generator malfunction. Spacecraft and launch vehicle processing buildings have CVCF (Constant Voltage Constant Frequency) system. Several types of electrical power are available in TNSC, e.g. (380V/220V/50Hz/60Hz for European), (220V/120V/60Hz/50Hz, for American), and (100V/60Hz, for Japanese).

4.8.2 Liquids and gases

All chemicals used will be in compliance with the requirements restricting ozone-depleting chemicals. Gaseous helium (GHe) and gaseous nitrogen (GN_2) are available at STA2, SFA2, SFA and VAB for spacecraft use. The gas quality is MIL-P-27407A Type 1 Grade A or equivalent for GHe, and MIL-P-27401C Type 1 Grade B or equivalent for GN_2, respectively.

For more information regarding available liquids and gases, please contact LSP.

4.8.3 Fluid cleanliness and characteristics Analysis

Cleanliness and characteristics of liquids such as propellant (fuels, oxidizers), water, solvents, and gases provided for spacecraft can be analyzed at Support Laboratory. For more information regarding the analysis, please contact LSP.

4.8.4 Hazardous processing support facilities

For spacecraft propellant loading and gas pressurization, a filling and assembly hall is available at SFA and SFA2. The filling and assembly hall is built in consideration of the safety of personnel who performs spacecraft hazardous processing. For hazardous processing (e.g. propellant loading), safety equipment such as protective suits (SCAPE-suits), breathing air system are available.

Chapter 5 LAUNCH OPERATIONS

5.1 General

5.1.1 Scope

This chapter provides SCO with information on typical launch operations at the launch site. SCO are required to meet the requirements specified in this chapter and specified in the “Launch Vehicle Payload Safety Standard” for spacecraft and in the “Facility Guide for H-IIA Payload Launch Campaign”, with respect to the safety management, the safety design and the launch site operations at Tanegashima Space Center (TNSC).

5.2 Outline of the launch-related organizations

Figure 1.2-1 shows the outline of LSP and its relationship with SCO. LSP will coordinate with SCO regarding all issues during the launch campaign. Even issues related to ground safety, flight safety, range safety, safety reviews, and launch facilities maintenance, LSP will coordinate with JAXA TNSC and provide best solution for SCO. LSP will assign the Mission Manager as a single point of contact with SCO for this purpose.

5.3 Launch Operations Requirements

The launch operations requirements associated with the safety requirements, operations interface requirements should be applied to the design and fabrication of spacecraft, the spacecraft processing, and the launch operations.

After launch contract is signed, the documents (refer to Chapter 6) containing detailed description of the interface will be prepared and updated in due course with the results at the technical interchange meetings.

5.3.1 Safety requirements

SCO shall meet the safety requirements specified in the “Launch Vehicle Payload Safety Standard” (by JAXA). (Refer to Section 6.4.)

5.3.2 Launch operations interface requirements

The launch operations interface requirements are specified in the interface documents established by LSP with agreement of SCO, such as Spacecraft / H-IIA Interface Control Document (ICD), Launch Operations Support Plan (LOSP), and Joint Operations Plan (JOP).

5.4 Responsibility and Organization

LSP will organize a launch operations team, MILSET (Mitsubishi Launch Site Service Team), at TNSC. MILSET is shall be responsible for the launch operations. During the launch campaign at TNSC, SCO is required to appoint a Spacecraft Mission Director. Spacecraft Mission Director shall be responsible for the spacecraft processing. LSP's Mission Manager, acting as a point of contact with the SCO, will coordinate to resolve the spacecraft processing issues. The spacecraft processing prior to SCO/LSP joint operations are performed by SCO. During the joint operations, all launch operations are conducted by LSP under the support of SCO.

Figure 5.4-1 shows the basic organization chart for launch operations. SCO is required to submit its organization chart before launch campaign starts at TNSC.

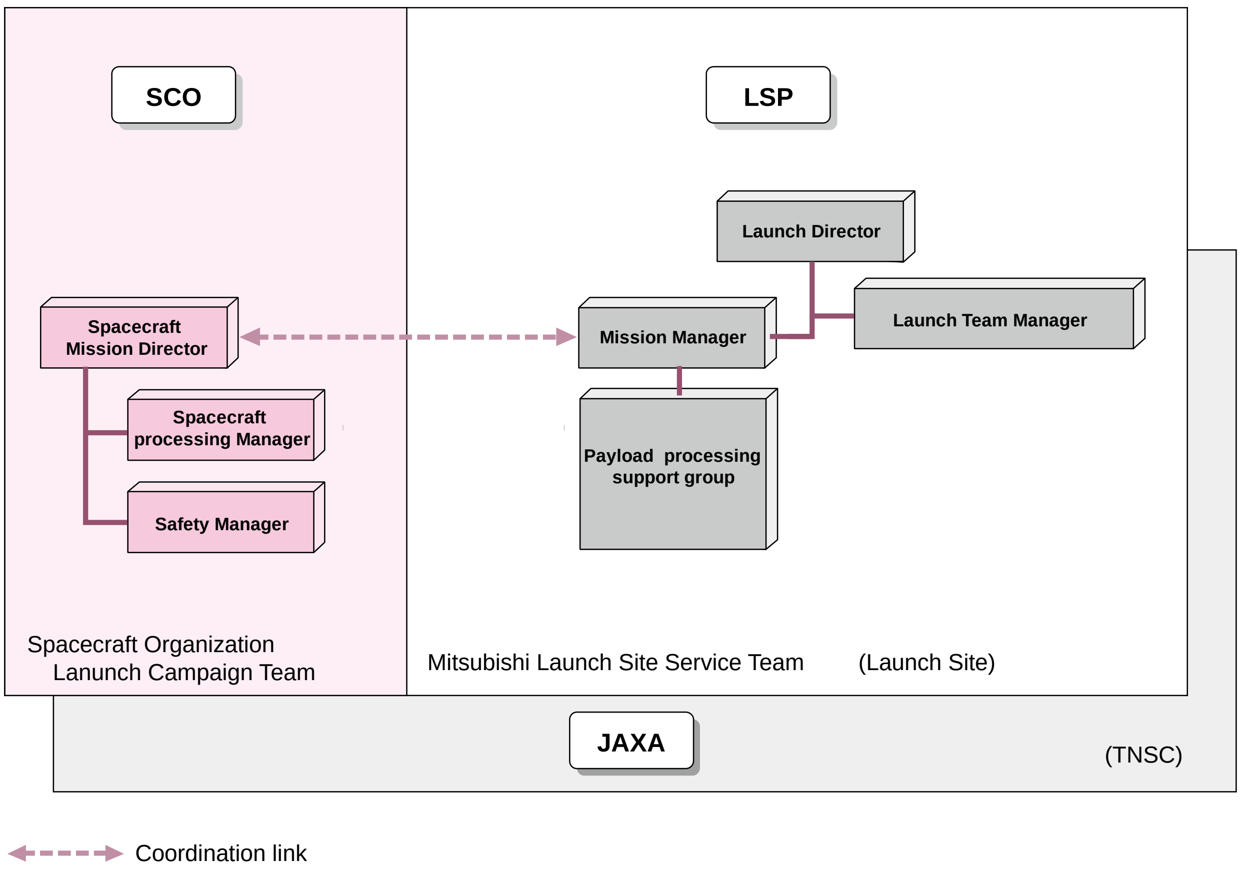


Figure 5.4-1 Launch Operations Team Organization

5.5 Restrictions

5.5.1 Restrictions on the ground

In order to ensure safety, some spacecraft processing, such as RF radiation and battery charge, will be required to operate by remote. Detailed restrictions will be coordinated and confirmed at the technical interchange meetings.

5.5.2 Restrictions on launch

This paragraph describes the launch window and launch related constraints. Details will be presented at the technical interchange meetings.

5.5.2.1 Launch window

(1) Launch Period and Launch Slot

Launch Period means a period equal to three (3) consecutive months which is preset at the time of the contract. Launch Slot means a period equal to thirty (30) consecutive calendar days which will be scheduled within the launch period. SCO and LSP will start coordination of the launch slot 1 year before launch.

(2) Launch Day

Launch Day means a calendar day established for launch during the Launch Slot within which the Launch Window is open. SCO and LSP will start coordination of Launch Day, including alternative date, 6 months before launch. Launch Day will be publicly announced 2 months before launch.

(3) Launch Window

Launch Window means a time period during the Launch Day within which the Launch may take place.

Launch Window will be determined by LSP considering restrictions and analyses such as sun and shade, sun angle in the trajectory or insertion orbit, collision avoidance analysis with manned spacecraft, and other relevant factors. To maximize launch opportunity, it is desirable that launch window is 45 minutes or more because of the launch vehicle recycle activities.

5.5.2.2 Launch requirement and constraint

Requirements and constraints for SCO before launch are shown below.

(1) Operations to access the spacecraft (e.g. detachment of Non-flight items, etc.) shall be finished before fairing door close on L-1 day.

(2) Prior to ML roll-out, SCO should turn off the spacecraft and EGSE in ML/GSE room.

(3) After ML roll-out to LP, SCO shall check the spacecraft status via EGSE in ML/GSE room by remote from STA2 or SFA2. After that, SCO shall evacuate from ML GSE room by X-9.5hr.

(4) The spacecraft electrical power supply shall be switched from external to internal,

typically by X-15min. And preventing sparking at liftoff, low current signals shall be allowed through the umbilical cable carries. Recommended voltage and current are 28 VDC, and less than 10 mA.

(5) Upon confirming that all spacecraft systems are ready for launch, SCO shall push the “Ready” button by X-300s.

(6) If SCO postpones the launch because of anomaly, SCO shall push the “Emergency Stop” button.

5.5.2.3 Launch criteria

Typical launch criteria are shown below.

(1) Launch Vehicle

(a) Ground wind

(i) During ML roll-out: peak wind speed shall be below 22.4 m/s

(ii) At lift-off: peak wind speed shall be below 20.9 m/s

(b) Rain

(i) During ML roll-out: rainfall shall be below 15mm/h

(ii) At liftoff: rainfall shall be below 8mm/h

(iii) No falling hails during ML roll-out

(c) Cloud

No cumulonimbus clouds exist in the flight trajectory.

(d) Lightning

No atmospheric discharge exists in the flight trajectory at liftoff. (This is based on detailed weather observation at liftoff phase.)

(i) No launch allowed if lightning is observed within 10 km of launch site

(ii) Launch suspended if lightning is observed within 20 km of the planned flight path

(iii) No launch allowed if lightning or cumulonimbus clouds* are observed in the planned flight path

*) Including the clouds with 1.8km vertical thickness of freeze layer

(e) Flight wind

No launch allowed if the launch vehicle performance is adversely affected by the flight wind profile as follows.

(i) Separated SRB-As are expected to fall out of the planned area.

(ii) Engine gimbal angle is expected to exceed the design limit.

(iii) Flight load is expected to exceed the design limit.

(2) Range Safety and Flight Safety

(a) Ground wind

Peak wind speed shall be below 20 m/s during tracking phase.

(b) Flight wind

No launch allowed if range safety is expected to be adversely affected by flight wind profile as follows.

(i) Destructed objects are expected to fall outside the warning area when the launch vehicle is destructed near the launch pad.

(c) Facilities

Range safety and flight safety systems shall be in operations.

(d) Collision Avoidance

Launch window shall be set taking into account that the launch vehicle and spacecraft will not collide with manned spacecraft.

(e) Ground safety

It shall be confirmed that no one stays within the specified evacuation area.

(f) Maritime safety

It shall be confirmed that specified warning areas on the ocean are cleared.

5.5.2.4 Launch postponement

In case that the launch vehicle cannot be launched within the launch window on the scheduled date, launch will be postponed one (1) day or more. One (1) day is the minimum postponement duration. In case that the postponement is determined after the launch vehicle propellant loading, next launch day will be set typically after three (3) days from the first launch day. Depending on the causes of the postponement, access to the spacecraft may be restricted. For each launch mission, typical cases of postponement plan will be coordinated and established through the meeting during the launch campaign.

5.6 TNSC Facilities and GSE Related to Spacecraft Processing

The following describes the major facilities and GSE which can be utilized by SCO at TNSC. The facilities and GSE actually utilized by SCO are specified in the Spacecraft / H-IIA ICD.

(a) No. 1 Spacecraft Test and Assembly Building (STA 1) and No.2 Spacecraft Test and Assembly Building (STA 2) (spacecraft functional test)
(b) Spacecraft and Fairing Assembly Building (SFA) (propellant loading, battery charging, encapsulating into the payload fairing, etc.)
(c) No. 2 Spacecraft and Fairing Assembly Building (SFA2) (spacecraft functional test, propellant loading, encapsulating into the payload fairing, etc.)
(d) Vehicle Assembly Building (VAB) (mounting on launch vehicle, battery charging, final checkout, arming, etc.)
(e) Movable Launcher (ML)
(f) Takesaki Range Control Center (RCC)
(g) Small satellite propellant loading building (SPLB)
(h) Fuel handling facility (FHF) , Oxidizer handling facility (OHF) (propellant handling)
(i) Pyrotechnics storage facility and solid propellant storage facility
(j) Nondestructive Test Facility (NDTF) (solid motor nondestructive inspection)
(k) Other specifically requested facilities and GSE

Figure 5.6-1 shows the location of spacecraft-related buildings in TNSC's Osaki Range.

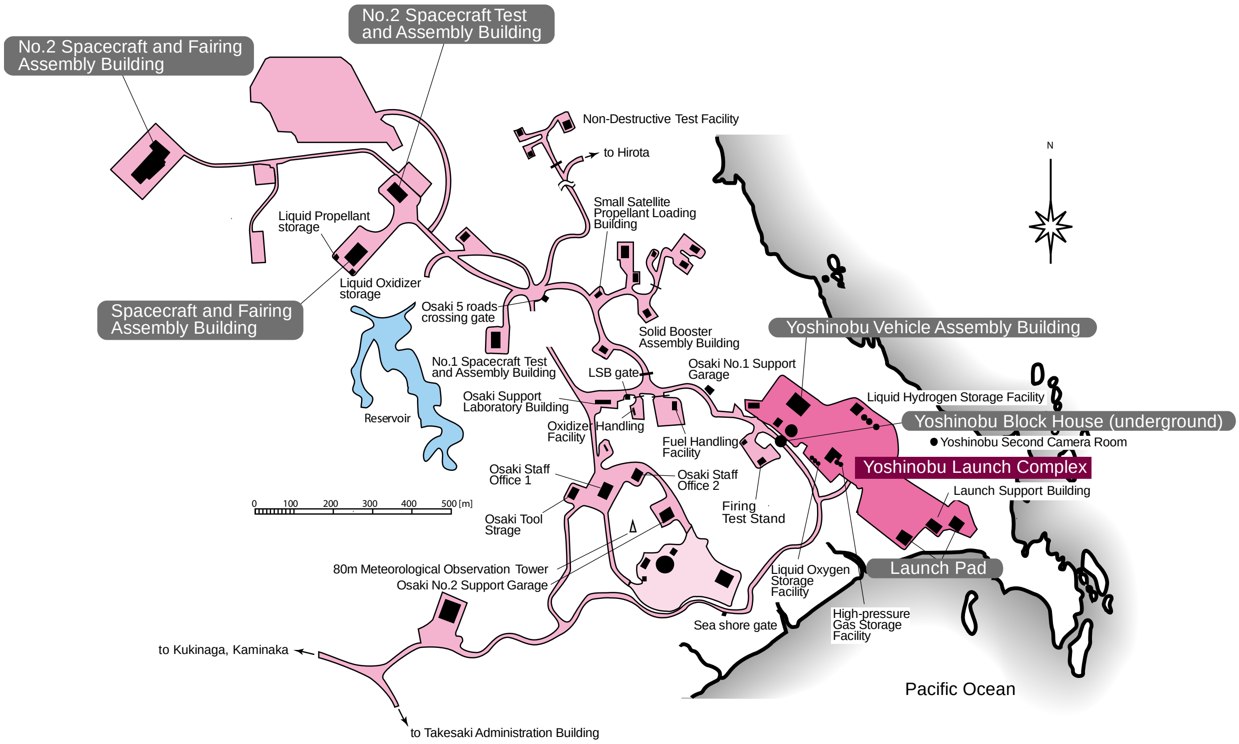


Figure 5.6-1 Location of spacecraft-related buildings in TNSC's Osaki Range

5.7 Launch Operations

This section describes the typical spacecraft processing and launch operations at TNSC.

5.7.1 Spacecraft processing

The spacecraft processing at TNSC are classified into three phases:

Phase 1 Spacecraft preparation and functional test
Phase 2 Spacecraft hazardous processing
Phase 3 SCO and LSP Joint operations (integrated procedure)

In Phase 1, operations are performed independently by SCO in STA1, STA2 or SFA2. In this phase, spacecraft processing activities are conducted under the responsibility of SCO with support of LSP.

In Phase 2, operations are performed independently by SCO in SFA, SFA2, and others. In this phase, spacecraft processing activities are conducted under the responsibility of SCO with support of LSP.

In Phase 3, operations are performed jointly by SCO and LSP in SFA or SFA2 and VAB. Operations from the spacecraft mating to payload adapter (PLA) and the payload support structure (PSS) through the encapsulated spacecraft integration with the launch vehicle are conducted by LSP with SCO support and witness.

Generally in the initial planning of spacecraft processing, SCO is expected to perform all operations from delivery of the GSE (checkout equipment) to TNSC to its removal from TNSC within 45 days (calendar days) (40 days before the launch, and five (5) days after the launch).
Figure 5.7-1 shows the typical spacecraft processing schedule.
Figure 5.7-2 shows the typical operations flow diagram for spacecraft preparation at TNSC.

5.7.2 Phase 1 (spacecraft preparation and functional test)

After arrival at Tanegashima Airport, Nishino-omote Seaport or Shimama Seaport, the spacecraft and GSE are transported to TNSC on the public roads. The spacecraft transported to STA1, STA2 or SFA2, are unpacked and installed by SCO. LSP will support these tasks. Hazardous materials, such as solid motors, pyrotechnics, propellant, and explosives, are transported to the specified place where they are to be stored. Consumable materials, propellant and high-pressure gas required for the spacecraft processing will be prepared by SCO or LSP, depending on the contract.

The spacecraft is transported to the preparation room (clean room), and GSE is set up in the checkout room adjacent to the test room. The spacecraft assembly and functional tests are performed in STA1, STA2 or SFA2. Hazardous processing, such as pyrotechnics installing and propellant loading, should not be performed in STA 1 and STA 2. However, inert high-pressure gas systems may be charged if sufficient safety is ensured (this operation requires permission by JAXA safety division).

After assembly and functional tests, the spacecraft are loaded into the container and transported to SFA or SFA2 by LSP.

Figure 5.7-3 shows the typical Phase 1 operations flow diagram.

5.7.3 Phase 2 (hazardous processing for spacecraft)

Hazardous processing such as installing solid motors and pyrotechnics, loading propellants, and charging high-pressure gas systems shall be performed in SFA or SFA2. Hazardous processing should be performed by the minimum required number (but more than two) of persons who have received safety instruction and training. Other persons give operations instructions or monitor the status from the monitor room. Only explosion-proof GSE can be set up in the room where the hazardous processing is to be performed.

Figure 5.7-4 shows the typical Phase 2 processing flow diagram.

5.7.3.1 Preparing and assembling pyrotechnics and solid motor

Before installation on the spacecraft, pyrotechnics and solid motor should be inspected in NDTF (pyrotechnics only).

After inspection and assembly, the pyrotechnics are transported into SFA or SFA2. All these operations, including transportation between buildings, shall be under the responsibility of SCO.

LSP will support operations requiring use of the TNSC equipment and materials.

5.7.3.2 Hazardous processing

(1) Transport (by LSP, non-hazardous operations)

The spacecraft is loaded in the container, transported from STA 1, STA 2, to SFA and unloaded from the trailer in the air lock. After the cleanliness in the air lock room has been established, the spacecraft is taken out of the container and is moved into the assembly room (clean room). Figure 5.7-5 shows typical transport operations.

(2) Loading the propellant and charging the high-pressure gas system

The propellant and the high-pressure gas for pressurization up to the flight level are loaded in SFA or SFA2 (or in VAB for special cases) by SCO. Also the operations such as depressurization, purging and flushing should be under the responsibility of SCO. Battery charge after propellant loading is allowed at SFA or SFA2 from remote building, under the condition that other operations are not conducting.

(3) Installing the pyrotechnics

The pyrotechnics and the solid motor can be installed by SCO in SFA or SFA2. However, pyrotechnics wire connection and arming should be conducted by SCO in VAB as part of the countdown operations.

5.7.3.3 Final spacecraft assembly

(1) Weight measurement

The spacecraft can be weighed in STA1, STA2, SFA or SFA2 under the responsibility of SCO. LSP will request SCO to weigh the spacecraft, concerning its effect on flight performance. Whether the weight is to be measured before or after loading the propellant and charging the high-pressure gas systems will be determined by coordination with SCO.

SCO can utilize the weighing equipment of TNSC.

(2) Final inspection

Electrical and mechanical inspection and solid propellant motor arming inspection must be completed before spacecraft encapsulation into the payload fairing.

After entering Phase 3 operations, direct access to the launch vehicle body (including the spacecraft) is allowed only when it is approved in the technical interchange meeting. The operations by telecommunications signals via the umbilical line or RF signals are allowed at SFA or SFA2 from remote building under the condition that other operations are not conducting.

5.7.4 Phase 3 (Joint operations by SCO and LSP)

Phase 3 consists of joint operations by SCO and LSP. Operations from mating the spacecraft to the payload adapter (PLA) / the payload support structure (PSS) through mounting the encapsulated spacecraft on the launch vehicle are conducted by LSP. SCO will support and witness.

Figure 5.7-6 shows the typical Phase 3 operations flow diagram for a dedicated launch.

Figure 5.7-7 shows the typical Phase 3 operations flow diagram for a dual launch.

The spacecraft and PLA mating can be performed in earlier phase.

5.7.4.1 Encapsulation into the payload fairing

The first operations in Phase 3 (joint operations by SCO and LSP) are to mate the spacecraft to the PLA, to mount them on the PSS and to encapsulate them into the payload fairing in SFA or SFA2. These operations are performed by LSP with the support of SCO. The spacecraft is usually handled together with PLA by means of the handling jig prepared by SCO. The additional weight of the adapter must be considered in the spacecraft and spacecraft handling jig design.

5.7.4.2 Encapsulation for dedicated launch

After the checkout of the spacecraft in SFA or SFA2, the spacecraft is encapsulated into the payload fairing. The major operations here are as follows:

(1) The spacecraft is mated to the PLA, and then mounted on the PSS. The pyrotechnics for the spacecraft separation is also installed.

(2) The payload fairing encapsulates the spacecraft from the top, and then the fairing bottom flange and forward connection flange of the PSS are bolted to each other. (In case of the Model 5S fairing, each half-fairing is mated with the PSS.)

Figure 5.7-8 shows the typical encapsulation sequence for dedicated launch.

5.7.4.3 Encapsulation for dual launch

After the checkout of the spacecraft in SFA or SFA2, two spacecraft are encapsulated into the payload fairing separately. The major operations here are as follows:

(1) The lower spacecraft is mated to the PLA, and then mounted on the PSS. The pyrotechnics for the spacecraft separation is also installed.
(2) The lower payload fairing for a dual launch (hereafter referred to as the lower fairing) encapsulates from top of the lower spacecraft and then the bottom flange of the lower fairing is bolted to the connection flange of the lower PSS.
(3) The upper spacecraft is mated to the PLA and the pyrotechnics for the spacecraft separation is installed.
(4) The upper spacecraft with the PLA is mounted on the upper PSS which is connected to the forward end of the lower fairing.
(5) The upper fairing encapsulates from top of the upper spacecraft and then the upper fairing is bolted to the upper PSS.

Figure 5.7-9 shows the typical encapsulation sequence for a dual launch.

5.7.4.4 Transportation of encapsulated spacecraft

The encapsulated spacecraft is loaded on the spacecraft/fairing trailer in SFA or SFA2 air lock room and transported to VAB. After arrival to VAB, the encapsulated spacecraft is unloaded from the trailer and hoisted.

Figure 5.7-9 shows the typical transportation sequence of the encapsulated spacecraft.

During transportation except for hoisting, LSP monitors the following conditions.

Temperature	: 5 ºC to 30 ºC (target value is selectable in Table 3.3-1)
Humidity	: less than 60 % RH
Vibration	: less than 17.7m/s$^2{}_{0\text{-P}}$ (1.8 G $_{0\text{-P}}$) (each axis)

5.7.4.5 Mounting on launch vehicle

After the encapsulated spacecraft is hoisted on the upper floor of VAB, it is directly mounted on the top of the second stage. After that, the air conditioning duct is connected to the fairing, and the payload fairing assembly transport jig is removed and then quick disconnect (QD) and fairing separation pyrotechnics are installed to their positions.

Figure 5.7-10 shows the typical sequence of mounting the encapsulated spacecraft on the launch vehicle.

5.7.4.6 Spacecraft inspection after installation

The spacecraft functional test can be conducted according to the joint operations schedule determined in advance. Details will be determined at the technical interchange meetings and the coordination meetings before joint operations. This also applies to the RF link test, leakage inspection, battery charging, and visual inspection. Spacecraft arming and disarming operations shall be inspected and witnessed by JAXA launch site safety division. During these operations, turning on the electric system or RF radiation system is prohibited. The electric or RF radiation systems of the spacecraft are prohibited to operate during the installation of pyrotechnics of the launch vehicle. While some propulsion system operations are being performed, switching may be prohibited. Details will be determined during the joint operations schedule coordination with LSP.

5.7.4.7 L-5 operations

The final preparation for the countdown configuration is performed during pre-countdown operations. Details will be determined at the technical interchange meetings and the coordination meeting before countdown.

5.7.4.8 L-4 ~ L-0 operations

SCO will perform the final functional test using the ground line and RF, and charge the battery within the specified time. Details will be determined at the technical interchange meeting and the coordination meeting before the launch countdown operations. Figure 5.7-12 shows the typical countdown schedule. The following gives some parts of restricted operations during the launch countdown operations.

(1) Hazardous operations

- (a) Leak check of the high pressure bottle of the launch vehicle (L-4)
  During this operation, access to the spacecraft is prohibited. (VAB entrance is controlled.)
- (b) Loading propellant for the second stage gas jet system of the launch vehicle (L-3)
  During this operation, all of spacecraft processing are prohibited. (VAB entrance is controlled.)
- (c) Pyrotechnics wire connection by LSP (L-3)
  During this operation, spacecraft processing such as switching or turning on the electric and RF radiation system are prohibited.
- (d) Vehicle body arming by LSP (L-2)
  During this operation, spacecraft processing such as switching or turning on the electric and RF radiation system are prohibited.

(2) Others

- (a) Final door close (L-1)

Until L-1 final door close, SCO shall finish accessing to the spacecraft. After this event, SCO cannot access the spacecraft directly.

(b) ML roll-out (L-0)

ML is rolled out from VAB and mounted on the LP. During this operation, SCO should turn off the SC and EGSE. After the ML roll-out to LP, SCO shall check the SC via EGSE at ML GSE room from STA2 or SFA2 by remote. After that operation, SCO shall evacuate from ML GSE room by X-9.5hr.

In case of a dual spacecraft launch, all restrictions from each spacecraft side will be imposed on each other. The problems will be coordinated at the technical interchange meeting and the coordination meeting before the launch countdown operations.

5.7.4.9 Terminal countdown

Figure 5.7-13 shows the typical terminal countdown sequence of the H-IIA launch vehicle on the launch date (L-0). During terminal countdown, SCO will perform the following operations:

(1) Spacecraft flight configuration

The final spacecraft flight configuration that includes the completion of switching power supply or arming pyrotechnic devices etc. should be expected to be established before X-15(generally) minutes.

Then, SCO will inform the status to the Launch Director via Mission Manager of LSP in order to make X-10 minutes Final Go/No-Go poll.

(2) Completion of spacecraft preparation

SCO shall conduct the final inspection and system check up to X-300 seconds and press the ready button that is one of the prerequisite for the automatic countdown sequence start.

(3) Automatic countdown sequence

LSP starts the automatic countdown sequence after X-270 seconds.

(4) Countdown recycle

If the recycle command is issued during the countdown, all countdown operations are reset to X-25 minutes.

5.7.4.10 Recycle operations (Launch postponement)

If the launch is postponed on the launch date, recycle operations are performed. The major recycle operations are as follows:

(a) Discharging the launch vehicle propellant and purging
(b) Transporting the ML to VAB
(c) Disarming spacecraft systems (if required)
(d) Disarming launch vehicle systems (if required)
(e) Operations required for repetition of L-1

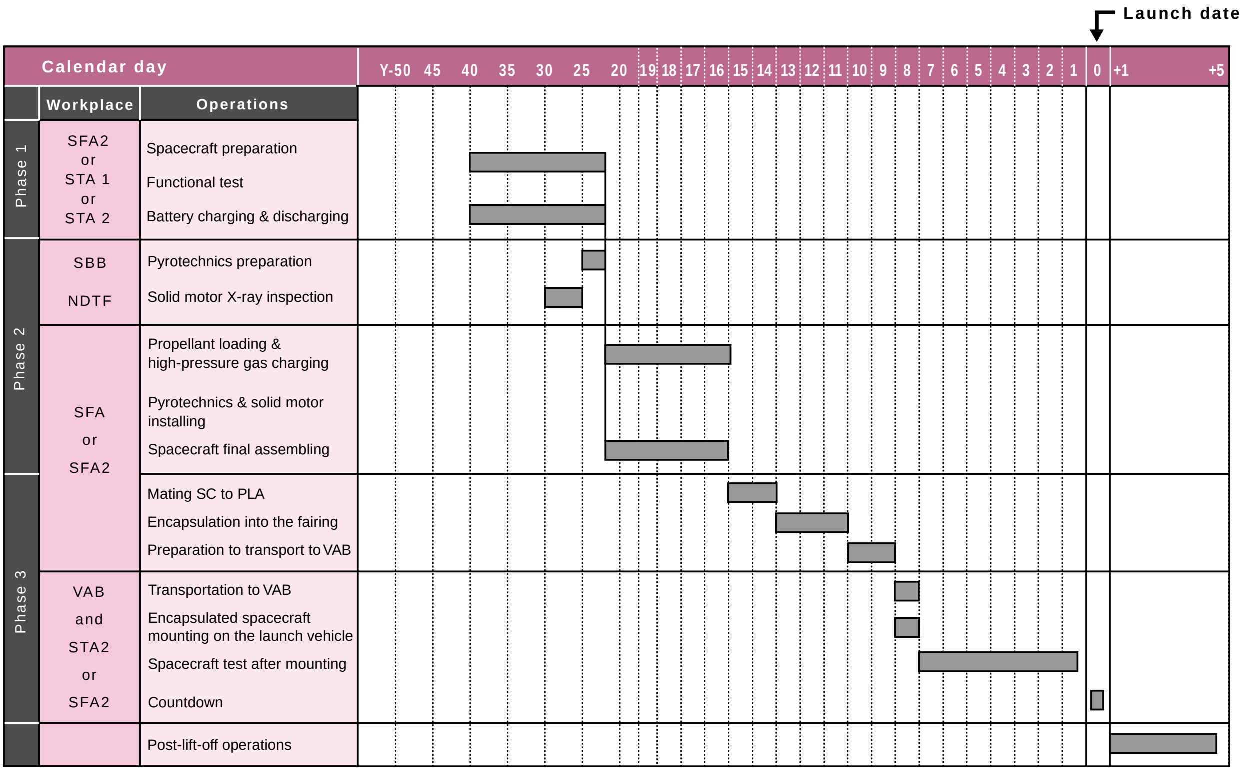


Figure 5.7-1 Typical spacecraft processing schedule

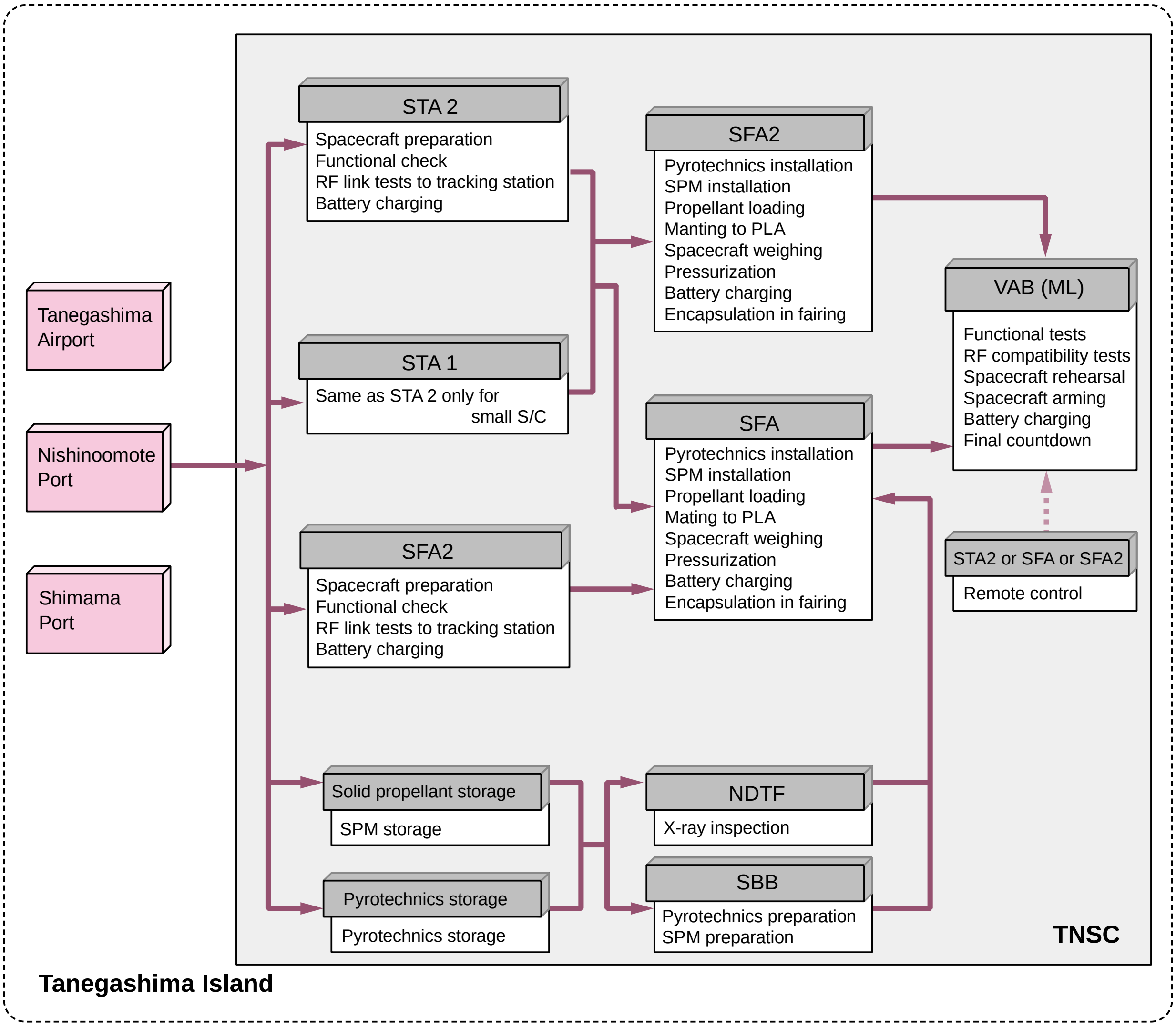


Figure 5.7-2 Typical spacecraft processing flow diagram at TNSC

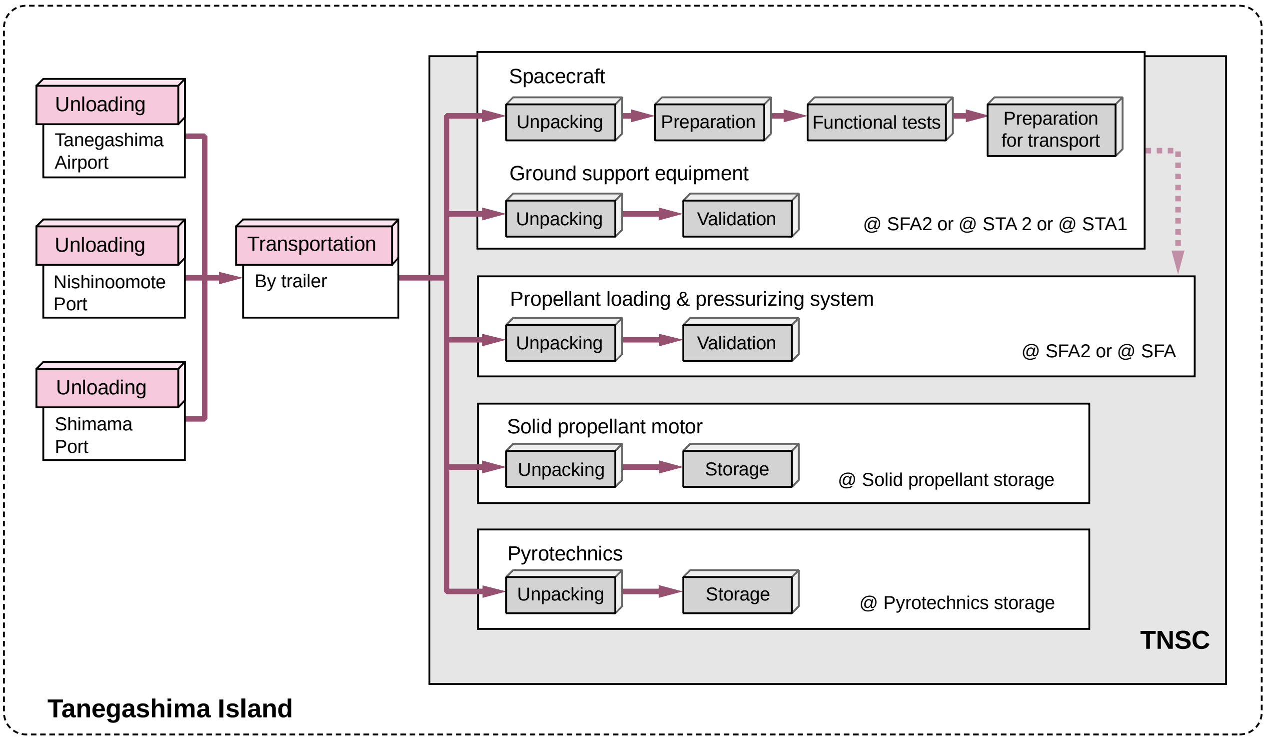


Figure 5.7-3 Typical phase 1 processing flow diagram

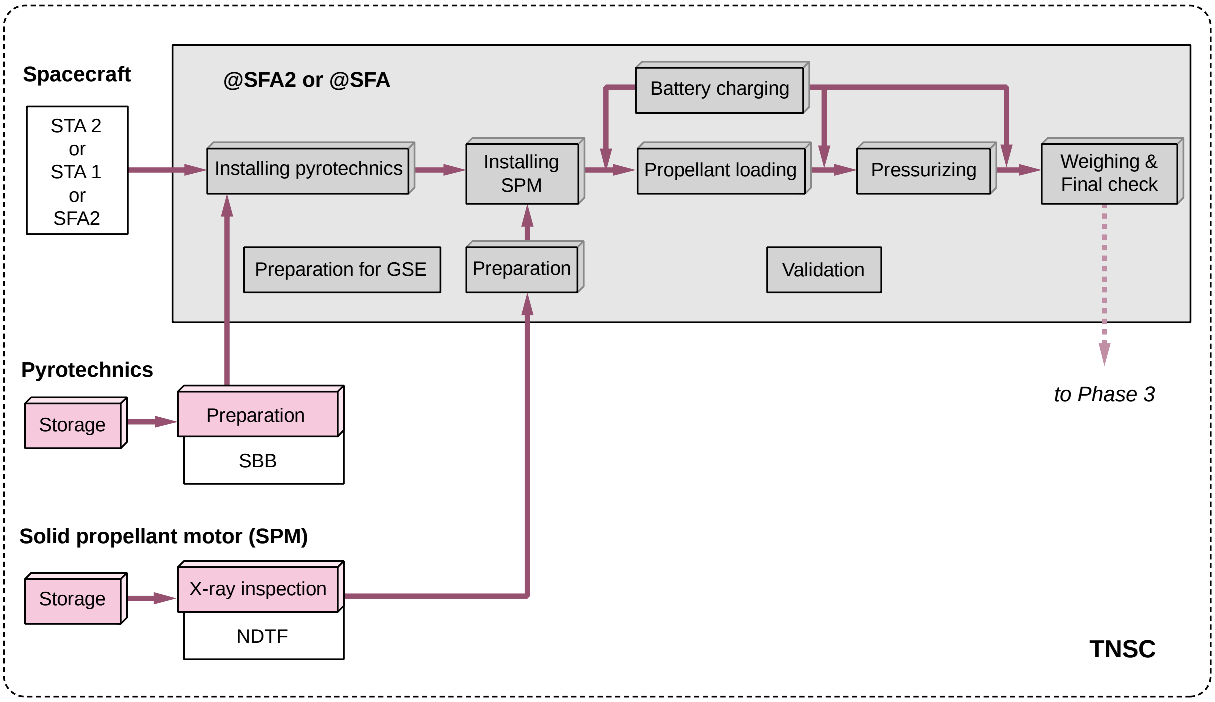


Figure 5.7-4 Typical phase 2 processing flow diagram

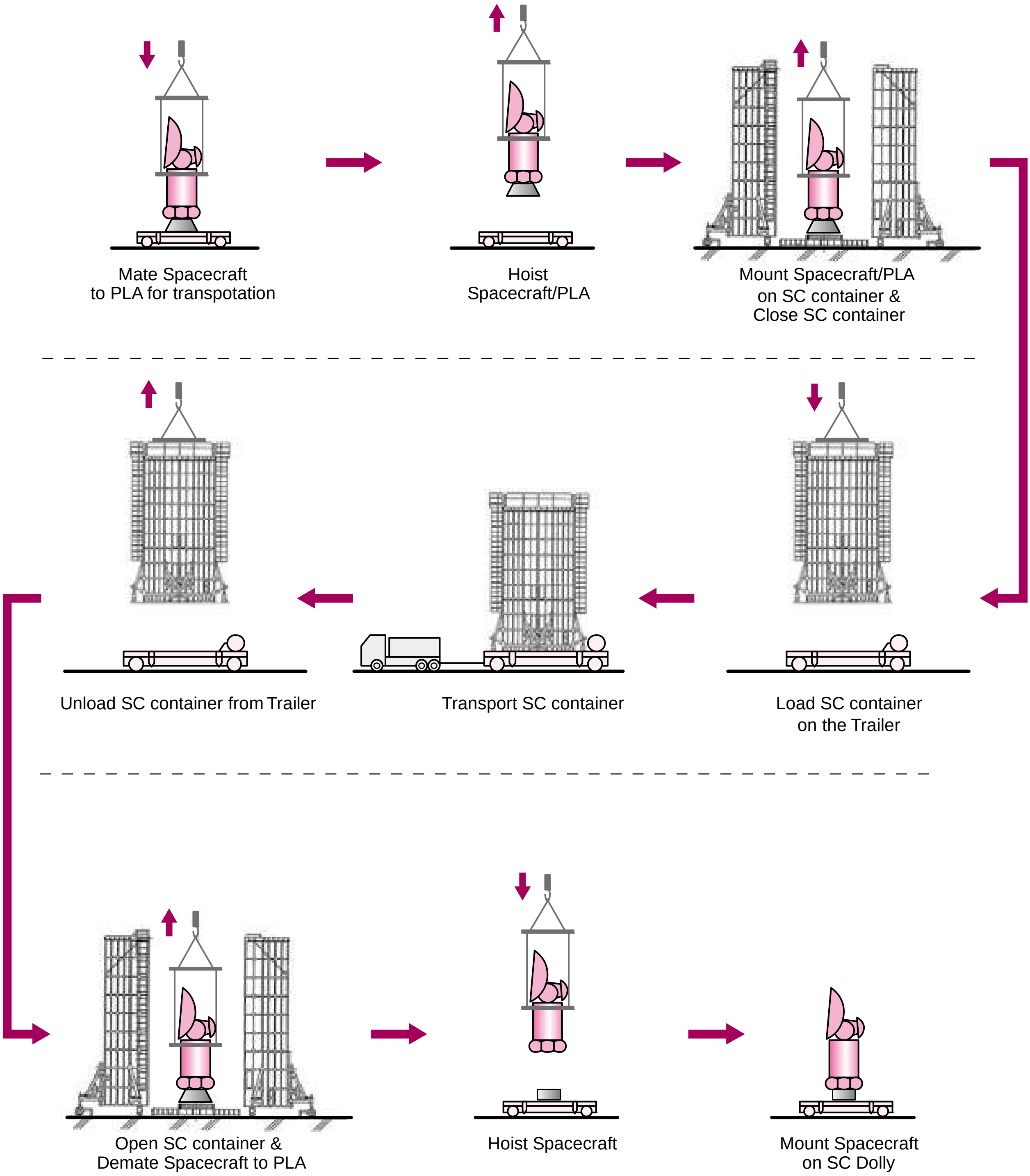


Figure 5.7-5 Typical Spacecraft transport operations from STA2 to SFA

L-15～L-14	L-13～L-11	L-10～L-9	L-8～L-6	L-5 ~ L-1	L- 0
Mating to payload adapter Mounting on payload support structure Preparation of encapsulation	Encapsulation in fairing	Preparation to transport to VAB	Transportation to VAB Hoisting of encapsulated spacecraft Mounting on launch vehicle	Spacecraft test after mounting All system (spacecraft / vehicle) rehearsal (dry run)	ML roll-out Countdown
@ SFA2 or @ SFA			**@ VAB**		**VAB → LP**

Figure 5.7-6 Typical phase 3 operations flow diagram for dedicated launch (for Model 4S fairing)

	L-18 ~ L-17	L-16	L-15 ~ L-14	L-13 ~ L-11	L-10 ~ L-9
Upper spacecraft		Preparation of connection to lower fairing	Mating to payload adapter Upper spacecraft moutuing on lower fairing lid Preparation of encapsulation	Encapsulation in Upper fairing	Preparation to transport to VAB
Lower spacecraft	Mating to payload adapter Mounting on payload support structure Preparation of encapsulation	Encapsulation in lower fairing			
@ SFA2 or @ SFA					

L-8 ~ L-6	L-5 ~ L-1	L - 0
Transportation to VAB Hoisting of encapsulated spacecraft Mounting on launch vehicle	Spacecraft test after mounting All system (spacecraft / vehicle) rehearsal (dry run)	ML roll-out Countdown
@ VAB		VAB → LP

Figure 5.7-7 Typical phase 3 operations flow diagram for dual launch

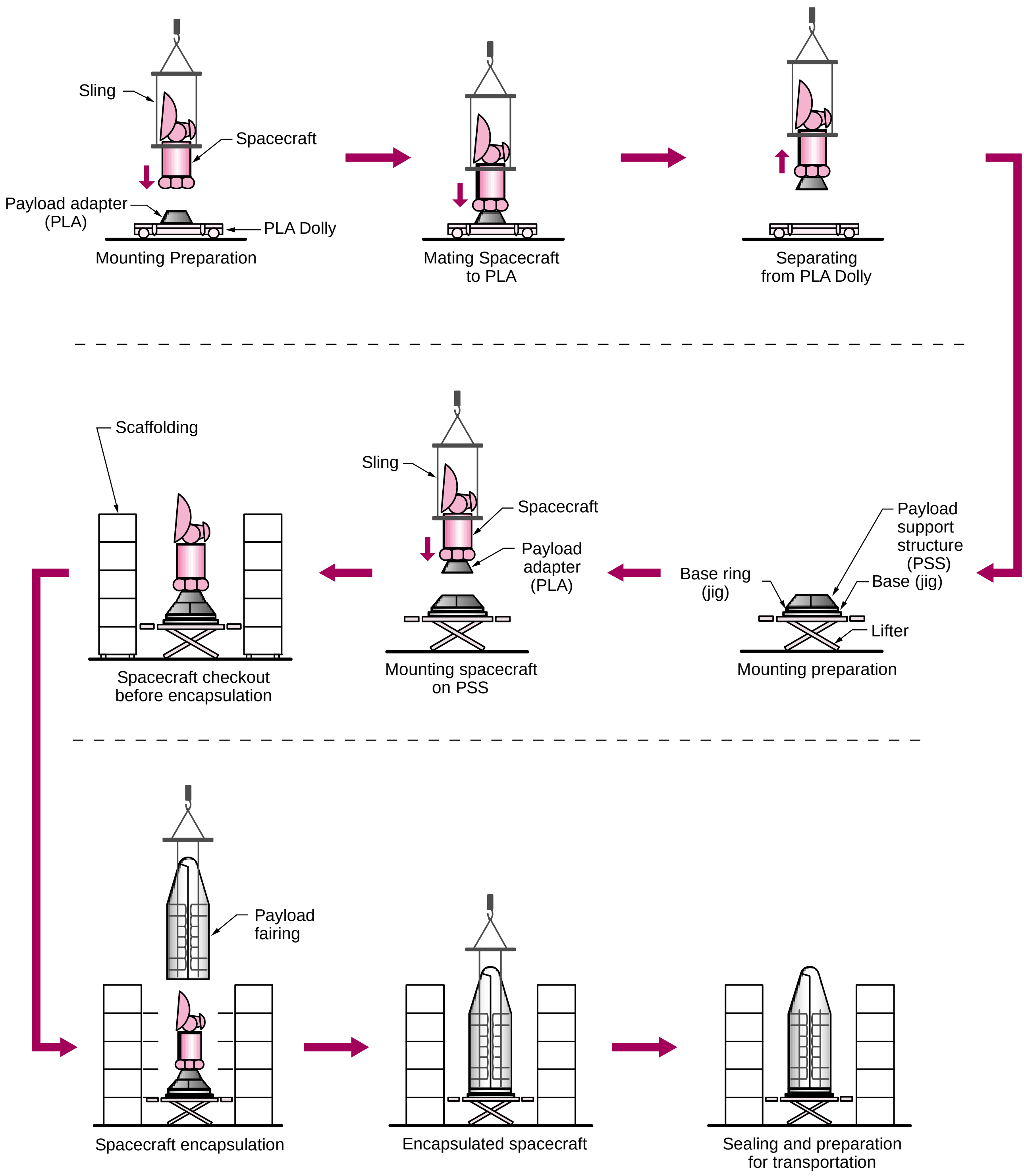


Figure 5.7-8 Typical encapsulation sequence for dedicated launch

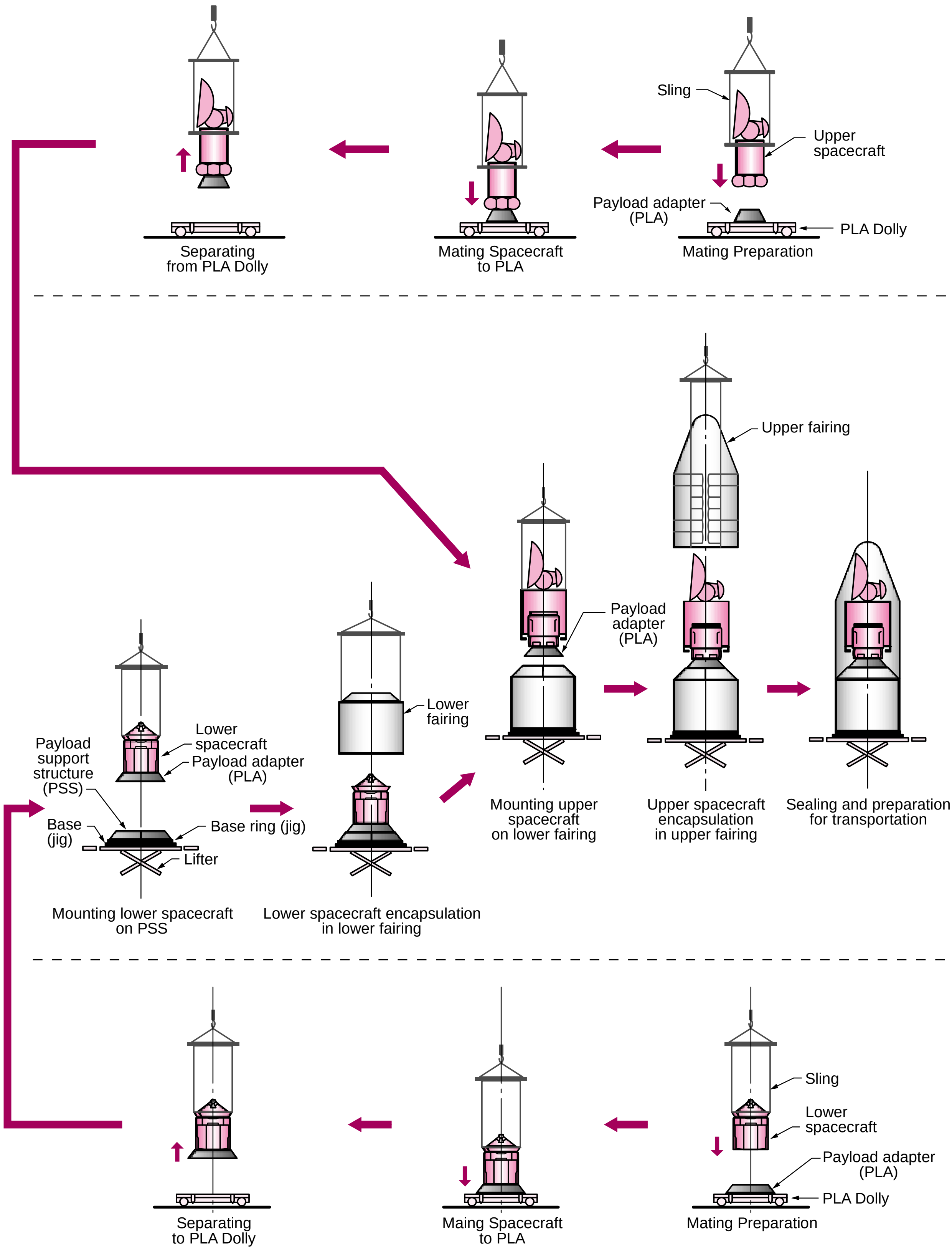


Figure 5.7-9 Typical encapsulation sequence for dual launch

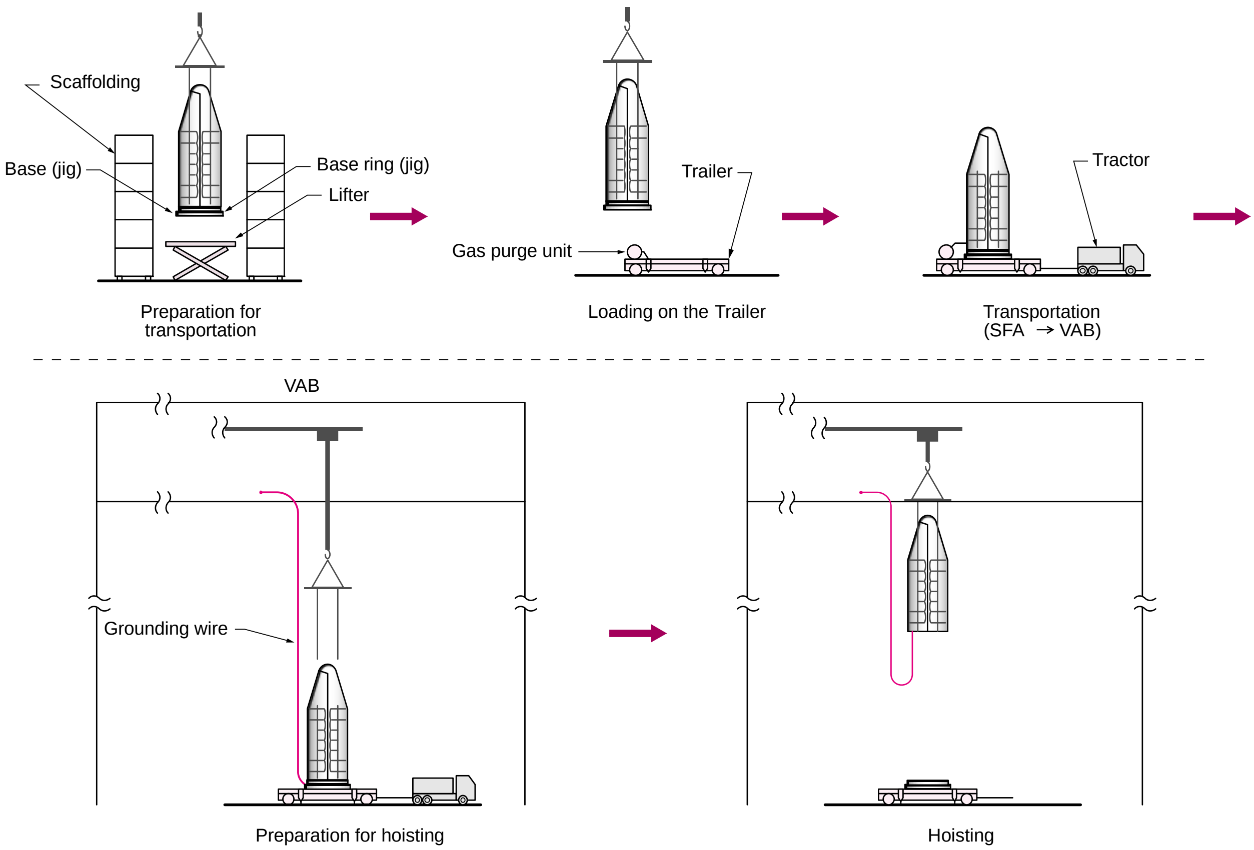


Figure 5.7-10 Transportation sequence of the encapsulated spacecraft

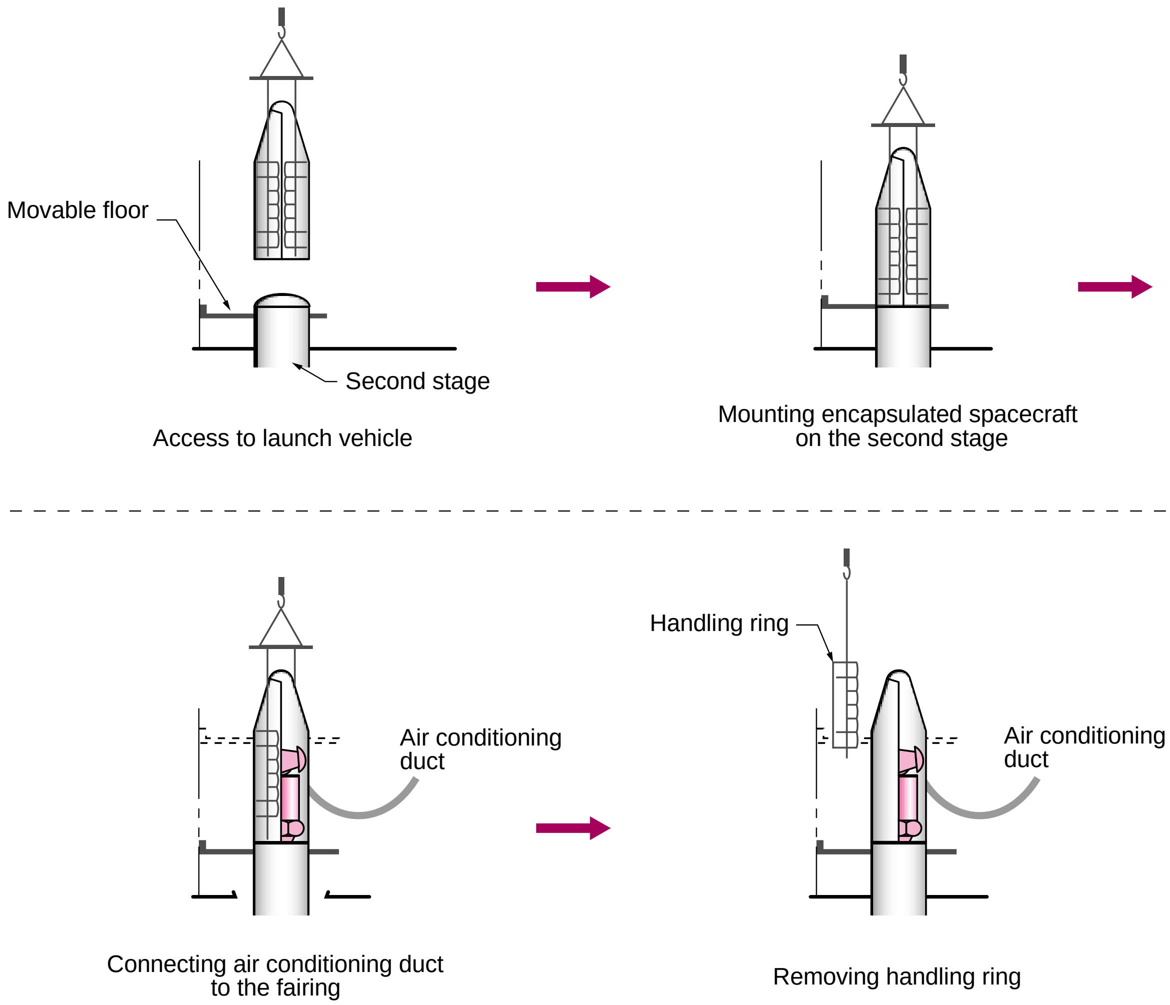
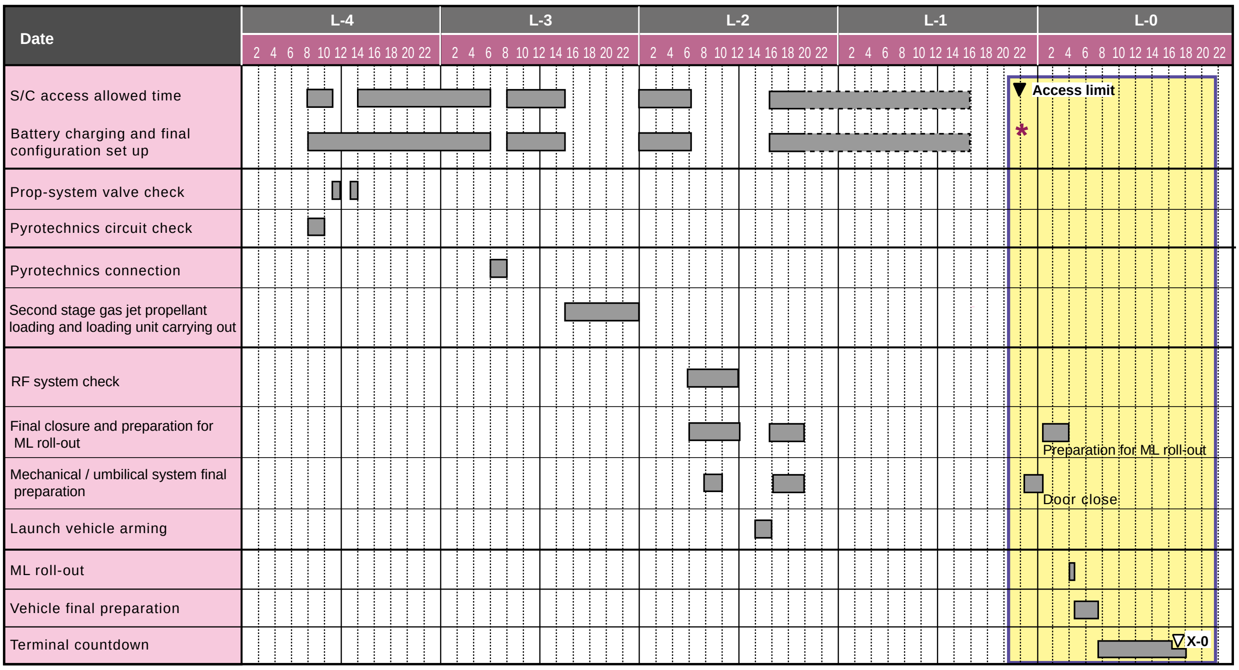


Figure 5.7-11 Encapsulated spacecraft mounting on launch vehicle



*) This is the typical schedule on the assumption that Lift-off time is 17 o'clock on L-0.
The start time of L-1 day is depending on the launch time.

Figure 5.7-12 Typical countdown schedule

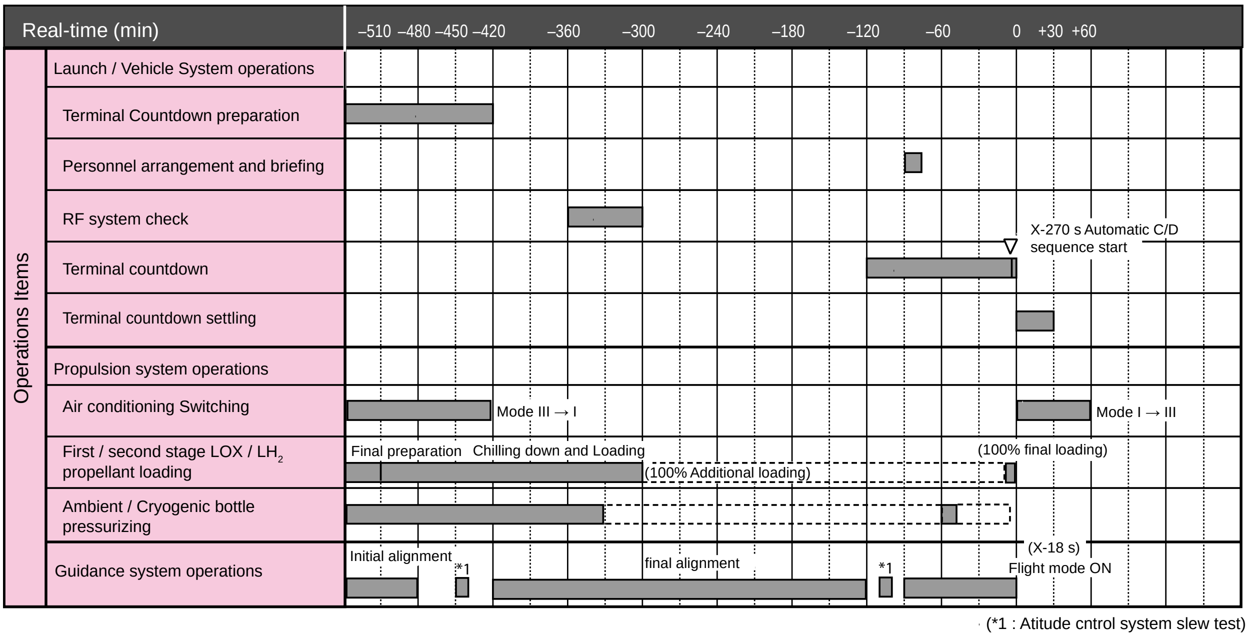


Figure 5.7-13　Typical launch vehicle system countdown schedule

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Chapter 6 INTERFACE MANAGEMENT

6.1 General

This chapter describes the spacecraft / H-IIA launch vehicle interface management.

6.1.1 Launch services point of contact

The contact point of LSP is described below.

(1) Contract

Space Systems Business Development Department is responsible for the development of a commercial launch services business and launch services contract management.

(2) Launch services

Space Systems Division is responsible for production of the H-IIA launch vehicle. After the contract, it becomes a single point of contact with SCO and responsible for the H-IIA Launch Services.

6.1.2 Interface management document

6.1.2.1 Document to be submitted by SCO

SCO is required to submit to LSP the documents shown in Table 6.1-1 which are required for coordinating the interface between the spacecraft and H-IIA launch vehicle System.

6.1.2.2 Document to be prepared by LSP

After the contract, LSP and SCO have technical interchange meetings (TIMs), and the results of the agreement between SCO and LSP are incorporated in the “Spacecraft / H-IIA Interface Control Document (ICD)” which specifies details of the interface items.

The ICD is maintained and managed by LSP until launch.

Table 6.1-1 Spacecraft/launch vehicle interface items

	Interface Items	Provider	Output Date (month)	Notes
(1)	Interface Requirements Document (IRD)	SCO	L-18	
(2)	Spacecraft processing requirements (PRD)	SCO	L-16	For launch operations support documents
(3)	Spacecraft drawing	SCO	L-18	For detailed interface definition
(4)	Spacecraft mass and orbit requirements --- initial	SCO	L-18	For mission trajectory
(5)	Spacecraft mass and orbit requirements --- updated (if any)	SCO	L-12	For mission trajectory
(6)	Spacecraft injection orbit accuracy requirements	SCO	L-16	For injection accuracy analysis
(7)	Spacecraft sun angle requirements	SCO	L-16	For sun angle analysis
(8)	Spacecraft separation condition requirements, mass property --- initial	SCO	L-16	For spacecraft separation analysis
(9)	Spacecraft separation condition requirements, mass property --- updated (if any)	SCO	L-6.5	For spacecraft separation analysis
(10)	Spacecraft contamination requirements	SCO	L-16	For CCAM analysis
(11)	Spacecraft dynamic model --- initial	SCO	L-18	For coupled loads analysis
(12)	Spacecraft dynamic model --- updated (if any)	SCO	L-9	For coupled loads analysis
(13)	Spacecraft thermal model	SCO	L-9	For integrated thermal analysis
(14)	Spacecraft EMC/RF characteristics	SCO	L-7	For radio frequency compatibility analysis
(15)	Spacecraft volume and sectional area information	SCO	L-7	For fairing venting analysis
(16)	Spacecraft volume, fill factor, and venting characteristics	SCO	L-7	For fairing venting analysis
(17)	Spacecraft sloshing characteristics	SCO	L-10	For launch vehicle stability analysis
(18)	Safety program plan	SCO	L-18	
(19)	Safety data package	SCO	L-16	
(20)	Documents required for legal procedures	SCO	L-12	
(21)	Customer's Logo	SCO	L-6	
(22)	Organization chart during launch campaign	SCO	L-3	
(23)	Interface control document --- initial	LSP	L-16	
(24)	Interface control document --- updated	LSP	L-3	
(25)	Mission trajectory result --- preliminary	LSP	L-16	
(26)	Mission trajectory result --- final	LSP	L-10	
(27)	Injection accuracy analysis result --- preliminary	LSP	L-15	
(28)	Injection accuracy analysis result --- final	LSP	L-6.5	
(29)	Sun angle analysis result --- preliminary	LSP	L-15	
(30)	Sun angle analysis result --- final	LSP	L-9	
(31)	Spacecraft separation analysis result --- preliminary	LSP	L-14	
(32)	Spacecraft separation analysis result --- final	LSP	L-5	
(33)	CCAM analysis result --- preliminary	LSP	L-14	
(34)	CCAM analysis result --- final	LSP	L-3.5	
(35)	Coupled loads analysis result --- preliminary	LSP	L-14	
(36)	Coupled loads analysis result --- final	LSP	L-5	
(37)	Radio frequency compatibility analysis result	LSP	L-4	
(38)	Integrated thermal analysis result (Option)	LSP	L-5	
(39)	Fairing venting analysis result (Option)	LSP	L-5	
(40)	Payload / Fairing clearance analysis result (Option)	LSP	L-4	
(41)	Launch operations support documents --- initial	LSP	L-11	
(42)	Launch operations support documents --- updated	LSP	L-3	

6.2 Interface Coordination with SCO

Spacecraft / H-IIA launch vehicle interface coordination is handled by LSP. The results of the interface coordination are incorporated in the ICD, whenever applicable.

6.2.1 Interface schedule / Interface items

The interface schedule is established to define interface tasks including analysis, match mate test and review meetings. It specifies the time of the tasks implementation and data exchange. The details will be coordinated at technical interchange meetings (TIMs) for each mission. A typical interface schedule and interface items for the standard orbit mission are shown in Figure 6.2-1 and Table 6.1-1, respectively.

Note that the schedule and items shown in Figure 6.2-1 and Table 6.1-1 are those for the standard missions (such as standard GTO or standard SSO) using already developed spacecraft bus. In case that special orbit injection is required or in case that the spacecraft has some first flight items, which additional analyses might be needed, then LSP can tailor the schedule based on the spacecraft requirements and coordination. Also, in case of missions using newly developed spacecraft bus, early phase of the spacecraft safety review (phase 0, 1, and 2) must be held separately and total integration period will be longer than that of Figure 6.2-1.

6.2.2 Mission analysis

6.2.2.1 General

LSP conducts following mission analyses to confirm that the spacecraft is injected into the required orbit with the required condition.

(1) Mission trajectory design
(2) Injection accuracy analysis
(3) Sun angle analysis
(4) Spacecraft separation analysis
(5) Contamination and collision avoidance maneuver analysis
(6) Coupled loads analysis
(7) Radio frequency compatibility analysis
(8) Integrated thermal analysis (Option)
(9) Fairing venting analysis (Option)
(10) Payload / Fairing clearance analysis (Option)

Typically, some of these analyses will be conducted two cycles, preliminary and final. The preliminary analyses are conducted based on the requirements and mathematical models provided from SCO just after the mission integration kick-off. The final analyses are conducted based on the updated requirements or condition, and the results are reviewed at the Mission Analysis Review (MAR).

6.2.2.2 Overview of each analysis

6.2.2.2.1 Mission trajectory design

The mission trajectory design is conducted to determine the trajectory, and to provide the major sequence of events and target orbit at spacecraft separation which meet the mission requirements and constraints of the spacecraft. In this design, the mission requirements from SCO (orbit requirement, mass properties, sun angle constraints, requirements at spacecraft separation, etc.), launch vehicle data, and design criteria are used.

6.2.2.2.2 Injection accuracy analysis

The injection accuracy analysis is conducted to evaluate the injection orbit error at spacecraft separation caused by error sources including the vehicle model and inertial sensors, and to provide the covariance matrix at spacecraft separation, trajectory dispersion, and attitude data for other analyses. This analysis is based on the mission trajectory.

6.2.2.2.3 Sun angle analysis

The sun angle analysis is conducted to provide information on the sun angle historical profile and eclipse duration in GTO, and to verify sun angle constraint of the spacecraft. This analysis is based on the mission trajectory.

6.2.2.2.4 Spacecraft separation analysis

The spacecraft separation analysis is conducted to analyze the spacecraft pointing error, angular rate error, and clearance between spacecraft and second stage at separation by analyzing separation motion of the spacecraft and second stage, based on the latest information on spacecraft mass properties provided from SCO.

6.2.2.2.5 Contamination and collision avoidance maneuver analysis (CCAM)

The contamination and collision avoidance maneuver analysis is conducted to analyze the relative position between the launch vehicle and the spacecraft after spacecraft separation. This analysis also supplies information of contamination to which the spacecraft is deposited by the launch vehicle collision avoidance maneuver after the spacecraft separation.

6.2.2.2.6 Coupled loads analysis (CLA)

The CLA is conducted to assess the load imposed on the spacecraft and the relative displacement from the payload fairing during launch. This analysis uses the spacecraft CLA model provided from SCO in combination with the vehicle body model. The analysis result will be supplied to SCO. Even in case that the spacecraft mechanical properties does not comply with the H-IIA interface condition by this manual description, interface requirements between the spacecraft and the H-IIA launch vehicle will be tailored by the coordination based on the CLA.

6.2.2.2.7 Radio frequency compatibility analysis

The radio frequency compatibility analysis is conducted to verify the RF compatibility based on the H-IIA launch vehicle / spacecraft spurious radiation shown in Figure 3.6-1 and 3.6-2.

6.2.2.2.8 Integrated thermal analysis (Option)

The integrated thermal analysis is conducted to verify that the spacecraft thermal environments are within the range specified in the ICD. This analysis uses the thermal mathematical model provided from SCO and the mission trajectory. This analysis covers thermal environments from the spacecraft encapsulation into the payload fairing to the spacecraft separation.

6.2.2.2.9 Fairing venting analysis (Option)

Fairing venting analysis is conducted to show time history of the air pressure in the payload fairing to confirm that the pressure does not cause adverse effect on the spacecraft. This analysis use spacecraft volume, fill factor, and venting characteristics information.

6.2.2.2.10 Payload / Fairing clearance analysis (Option)

Payload/Fairing clearance analysis is conducted to confirm that the clearance between the spacecraft and payload fairing are greater than zero during flight. This analysis is conducted when the usable volume is violated.

6.2.3 Interface tests

6.2.3.1 Match mate test between spacecraft and PLA

The match mate test of the spacecraft and payload adapter is conducted by LSP using the flight model payload adapter. The payload adapter or test jigs are prepared and transported by LSP. Typical timing of the match mate test is as shown in Figure 6.2-1. Details will be coordinated at TIM.

6.2.3.2 Umbilical connector disconnection test (Option)

A disconnection function between the spacecraft umbilical connector and payload fairing umbilical connector is checked to verify that normal connection and disconnection are possible. This test is conducted by LSP. A set of the payload fairing connectors (flight model or equivalent) is transported to SCO facilities, and is subjected to the connection and disconnection test.

6.2.3.3 Separation shock test support(Option)

The separation shock test (option) at spacecraft separation event is conducted by SCO using the flight model payload adapter or equivalent test model. This test is usually held together with the spacecraft / payload adapter match mate test.
Details will be coordinated at TIM.

6.2.4 Mission modifications

6.2.4.1 Payload fairing mission modification

The following items can be installed on the payload fairing as requested by SCO. Details are described in Section 4.5.4.

(1) Access doors
(2) Umbilical connectors
(3) Radio transparent window
(4) Internal antenna
(5) External antenna
(6) Internal and external antennas
(7) Acoustic blankets
(8) Independent air conditioner (air conditioned inlet, partition wall and relief valve)

Figure 6.2-1 illustrates the interface schedule for this mission modification.

6.2.4.2 Payload adapter mission modification

The following items can be installed on the payload adapter as requested by SCO.

(1) The number of separation springs
(2) Umbilical connectors

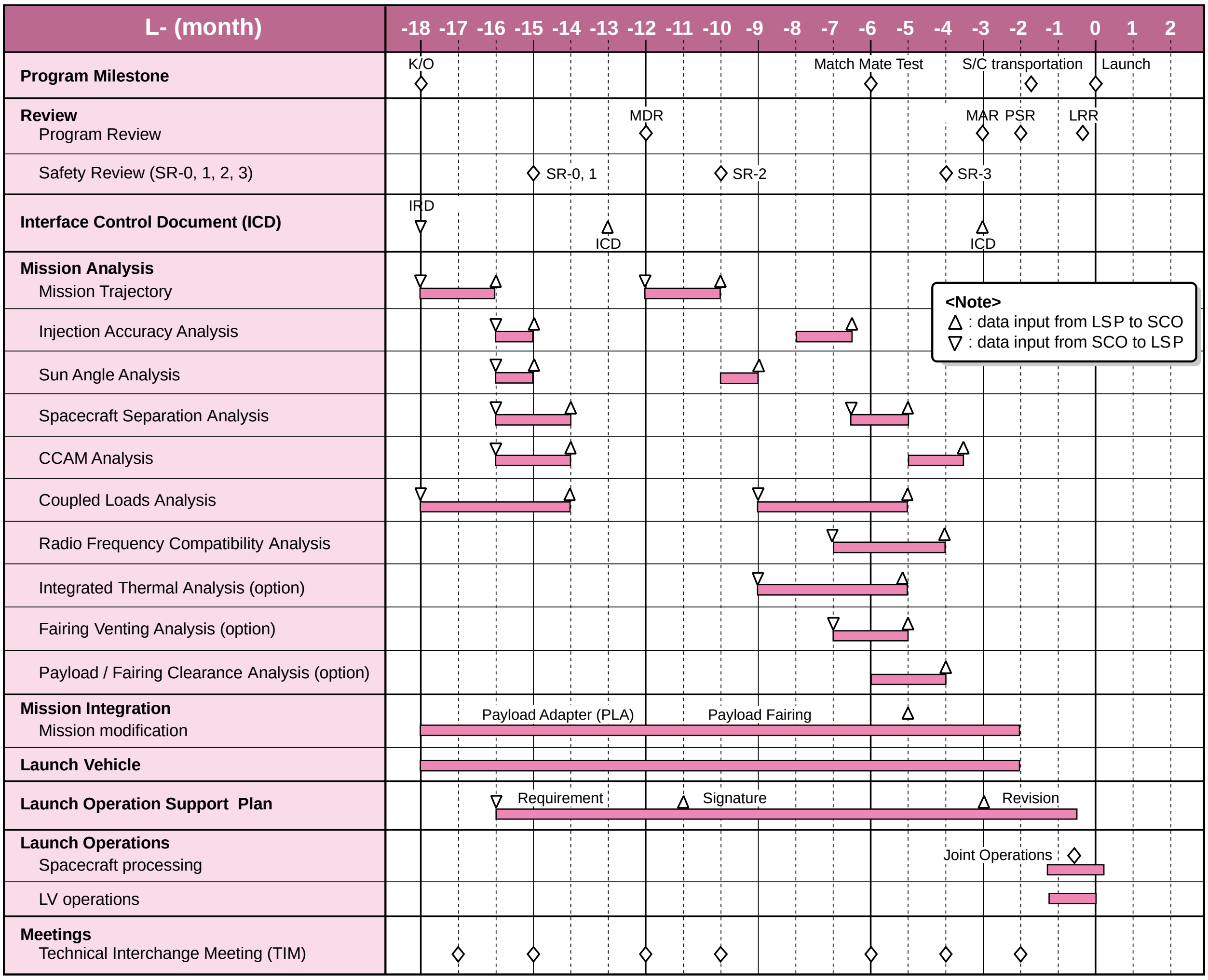

Figure 6.2-1 Typical spacecraft / H-IIA Launch Vehicle interface schedule for standard mission

6.3 Spacecraft / H-IIA Interface Control

6.3.1 Interface control document

The “Spacecraft / H-IIA Interface Control Document (ICD)” related to launch capability, interface restrictions and launch operations are created and maintained through TIMs held as required in the spacecraft design, fabrication and test phases, thereby clarifying the requirements and characteristics inherent to each mission.

Table 6.3-1 shows an example of contents of the ICD.

6.3.2 Coordination items and timing

The mission integration schedule is prepared by LSP according to results of coordination with SCO.

6.3.3 Launch Operations Interface Control

Focusing on spacecraft processing and SCO/LSP joint operations (phase 3 activities), launch operations support documents are created and maintained by LSP. Table 6.3-2 shows contents of the launch operations support documents.

Table 6.3-1 Standard spacecraft / H-IIA Interface Control Document

1. Scope

2. Applicable and reference documentation
- 2.1 Applicable documents
- 2.2 Reference documents
- 2.3 Safety submission sheets
- 2.4 Waivers

3. SC mission characteristics (reference)
- 3.1 Mission description
- 3.2 Spacecraft description

4. Mechanical Interfaces
- 4.1 Mechanical configuration
- 4.2 SC fundamental frequencies
- 4.3 Usable volume
- 4.4 Launch vehicle (LV) – Spacecraft adapter (PLA) interface
- 4.5 Umbilical connectors and micro switches
- 4.6 Purges and fluid connection interface
- 4.7 Encapsulated SC access

5. Electrical interfaces
- 5.1 Overall wiring diagram
- 5.2 Umbilical connectors
- 5.3 Umbilical wiring links
- 5.4 Electrical commands dedicated to spacecraft
- 5.5 Separation status transmission
- 5.6 In-flight telemetry
- 5.7 Power supply
- 5.8 Grounding

6. Radio frequency and electromagnetic interfaces
- 6.1 Characteristics of radio-electrical systems
- 6.2 RF telemetry and command link

7. LV and SC mission characteristics
- 7.1 SC input data for mission analyses
- 7.2 Trajectory and performance analysis
- 7.3 Launch windows
- 7.4 SC pointing and separation
- 7.5 Fairing separation

8. Verification analyses for induced environment
- 8.1 Mechanical environment
- 8.2 Thermal and humidity environment
- 8.3 Static pressure
- 8.4 Contamination and cleanliness
- 8.5 Radio and electromagnetic environments
- 8.6 Monitoring requirements

9. Verification tests
- 9.1 SC mechanical environment qualification and acceptance tests
- 9.2 LV-SC compatibility tests

10. Launch range operations: Facilities and support
- 10.1 Range capabilities
- 10.2 Range communication facilities
- 10.3 Umbilical lines and Ground lines
- 10.4 Overall data transmission
- 10.5 Operational constraints
- 10.6 Range services
- 10.7 General services
- 10.8 Procurement and Work responsibility

Chapter 6

Table 6.3-2 Launch operations support documents

Title	Contents
Launch Operations Support Plan (LOSP) during Launch Campaign	- Cover entire launch campaign period - Facility, equipment, services, consumable LSP will prepare - LSP activities to support SC launch preparation - Environment - GSE, propellant transportation plan internal launch site
Spacecraft/H-IIA Joint Operations Plan#1 -SC Transportation from STA2 to SFA- (JOP#1)	- Focused on SC transportation from STA2 to SFA - SCO/LSP Interface, allocation - Detailed Scenario and timeline
Spacecraft/H-IIA Joint Operations Plan (JOP#2)	- Focused on Joint operations after SC propellant loading - SC/adapter mating, fairing encapsulation, move SFA to VAB - Activity at VAB - SCO/LSP Interface, allocation - Detailed Scenario and timeline
Spacecraft/H-IIA Countdown Operations Procedure	- Focused on Countdown activities starting from 4 days before launch - Detailed interactive activities and communication between SCO and LSP - Detailed timeline for Go/No-go decision poll. - Specific constraint
Spacecraft Transportation Plan	- Transportation from entry port in Japan to launch site - Transportation from launch site to port to export from Japan. - Spacecraft, GSE, and propellant - Transportation route, means of transport, timeline, etc.

6.4 Safety Reviews

6.4.1 Payload safety requirements

The spacecraft design and spacecraft processing at the launch campaigns are required to meet requirements of“JMR-002: Launch Vehicle Payload Safety Standard” and “JERG-1-007: Safety Regulation for Launch Site Operation/Flight Control Operation”. SCO should enforce the safety program according to requirements of JMR-002. For more information regarding safety requirement document, please contact LSP.

6.4.2 Safety program plan

SCO is required to submit “Safety Program Plan“ to LSP according to JMR-002.

LSP establishes “Payload Safety Program Plan” based on the Safety Program Plan of SCO, and submits it to JAXA. LSP coordinates with JAXA to obtain the approval of JAXA safety authorities.

6.4.3 Safety review data package

SCO is required to submit the spacecraft safety data package to LSP according to JMR-002 for each safety review phase.

LSP pre-reviews the data package to ensure compliance with the safety requirements of JMR-002 and JERG-1-007 from the view point of integrated safety assessment between spacecraft and launch vehicle, and submits it to the JAXA safety review for approval.

JAXA holds safety review of this safety data package, and informs LSP and SCO of review results, and requires corrective actions if necessary.

6.5 Reviews and Other Meetings

This section describes the reviews and other meetings required to launch the spacecraft and launch vehicle.

6.5.1 Reviews and Meetings before launch campaign

6.5.1.1 Technical Interchange Meeting (TIM)

Technical interchange meeting is to coordinate technical interface matters. This meeting is held whenever required.

6.5.1.2 Program Management Meeting (PMM)

Program management meeting is to coordinate schedule and program promotion matters. This meeting is held whenever required.

6.5.1.3 Mission Modification Design Review (MDR)

Mission modification design reviews are conducted at major program milestones for both the generic launch vehicle and mission specific design/analysis, including the payload adapter and separation system. Mission specific PDRs and CDRs are conducted for each program, if necessary. SCO is expected to participate in these reviews to ensure that the design meets SCO requirements.

6.5.1.4 Mission Analyses Review (MAR)

Mission analyses review is to review the result of each mission analysis, defined in Section 6.2.2, shows the launch vehicle mission plan and performance meet the mission requirements defined in the ICD. Typically, this review is held at LSP facilities, and also launch vehicle preparation status is reported.

6.5.2 Reviews and Meetings on launch campaign

6.5.2.1 Launch Site Readiness Meeting (LSRM)

Launch site readiness meeting is to introduce the following status before the spacecraft is shipped to launch site. If SCO does not attend this meeting, the result is informed to SCO.

(1) Launch site readiness status for SCO activities
(2) Launch site readiness status for LSP activities and launch vehicle manufacturing status

6.5.2.2 Launch Site Daily Meeting (LSDM)

During the launch campaign, launch site daily meeting is held every morning before starting the spacecraft activities of the day in order to share SC/LV status such as:

(1) Planned support activities by LSP and any constraint for the spacecraft activities.
(2) SCO and LSP activities' status, especially for any unexpected event that might affect the planned activities and/or schedule.

6.5.2.3 Launch Vehicle Readiness Review (LVRR)

Launch vehicle readiness review is to ensure that launch vehicle and GSE preparations are ready for countdown operations. The review is held just before the LRR to ensure the verified results are reported in the LRR. If SCO does not attend this meeting, the result is informed to SCO.

6.5.2.4 Range Safety Readiness Review (RSRR)

Range safety readiness review is to ensure that the flight safety, ground safety, and range safety are established, and preparations for telemetry, tracking/communication are completed. This review is held by JAXA and typically attended by LSP personnel. The result is informed to SCO.

6.5.2.5 Spacecraft Readiness Review (SCRR)

Spacecraft readiness review is to ensure that spacecraft and GSE preparations are ready for countdown operations. This review is held by SCO and typically attended by LSP personnel. This review is held just before encapsulation of the spacecraft into the payload fairing to ensure that the verified results are reported in the LRR.

6.5.2.6 Launch Readiness Review (LRR)

Launch readiness review is to confirm that all organizations involved in the launch are ready to start countdown operations. Each organization reports the current progress of preparation and conclusion of the review meeting. Based on the launch director’s approval after this review, the countdown operations are started.

6.5.3 Safety review

The safety review is described in Section 6.4.

6.5.4 Post-flight Meeting (PFM)

Post-flight meeting is to evaluate the post-flight analysis data; injection data, orbital tracking operations report, launch evaluation report. Also, the “lessons learned” about the launch vehicle, spacecraft processing support during launch campaign are discussed, and appropriate actions will be identified for the next launch program.

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Appendix 1 HISTORY OF NASDA LAUNCH VEHICLES

A1.1 General

N-I, N-II, H-I and H-II launch vehicles had been developed since early seventies when NASDA, currently JAXA, started the space development. Many organizations related to H-IIA Launch Services had contributed to these launch vehicle's development. Abstracts of the H-I and H-II Launch Vehicles are described in the following sections.

Table A1-1 summarizes launch results of NASDA launch vehicles.
Figure A1-1 shows configuration of NASDA launch vehicles.

A1.2 Abstract of The H-I Launch Vehicle

NASDA had used the H-I Launch Vehicle as the main launch vehicle since 1986. It had Delta-based first stage (the MB-3 engine), which was basically same as the N-I and N-II rockets with strap-on boosters (SOBs), and a domestically developed LOX / LH_2 second stage propulsion system (the LE-5 engine). Up to 1992, nine payloads (geostationary meteorological satellites, communications satellites, and TV broadcasting satellites) had all been successfully launched by the H-I from Osaki Range of the Tanegashima Space Center (TNSC).

A1.3 Abstract of The H-II Launch Vehicle

The H-II Launch Vehicle was designed to serve as NASDA's main space transportation system in the 1990s to meet the demands for large satellites launching at lower cost yet maintaining a high degree of reliability. It was capable of sending a 4-ton class payload into geostationary transfer orbit.

The H-II was a two-stage launch vehicle equipped with two large scale solid rocket boosters (SRBs) for thrust augmentation. A new liquid propellant (liquid hydrogen / liquid oxygen) engine, called LE-7, was developed for the first stage. The LE-7 was a high performance engine adopting a high-pressure staged-combustion cycle. The LE-5A engine, which was the improved version of the LE-5 engine developed for the H-I, was used in the second stage. A strapped-down inertial guidance system utilizing ring laser gyros was employed for the guidance system. The standard payload fairing was 4 meters in diameter and 12 meters in length so that it could encapsulate a payload up to 3.7 meters in diameter and 10 meters in length.

Besides carrying satellites into low earth orbit and geostationary transfer orbit, the H-II was capable of launching planetary probes. The H-II was launched from Yoshinobu launch complex at the TNSC.

Table A1-1 Launch results summary of NASDA Launch Vehicles (from 1975 to 2000)

Launch Vehicle	Number of Vehicles Launched		Success Rate (%)	Mission					
				GTO		LEO		SSO	
	Success	Failure		Success	Failure	Success	Failure	Success	Failure
N-I	7	0	100	3	0	4	0	–	0
N-II	8	0	100	7	0	–	0	1	0
H-I	9	0	100	6	0	1	0	2	0
H-II	5	2	71	3	2	4	0	1	0
Total	29	2	93	19	2	7	0	4	0

*1 : Include three dual launch missions.

*2 : In the restart phase, the second stage's engine was cutoff before scheduled plan (F5).

: The first stage's main engine was cut off before scheduled plan (F8).

Figure A1-1 Configuration summary of NASDA Launch Vehicles

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Appendix 2　SPACECRAFT DATA SHEET

Spacecraft organization having interest to launch its spacecraft using the H-IIA launch vehicle is expected to inform of the spacecraft requirements, by completing the spacecraft data sheet in the following pages and return it to the address below.

The information from the spacecraft organization will be used for LSP's evaluation of compatibility between the spacecraft and H-IIA launch system, or be used to establish the basis of the spacecraft/ H-IIA interface agreement.

Please note that not all the items shown in the following data sheet and not all the answers from the spacecraft organization are within the scope of the standard MHI Launch Services.

LSP will treat the information as spacecraft organization's proprietary information, and will not disclose any part of them outside LSP without written approval.

Space Systems Business Development Department
Business Development Division
Integrated Defense & Space Systems

Mitsubishi Heavy Industries, Ltd.
16-5, KONAN, 2-CHOME, MINATO-KU, TOKYO, 108-8215 JAPAN
Telephone + 81-3-6716-4342
Fax + 81-3-6716-5869
http://h2a.mhi.co.jp/en/index.html

Requirements / Information	Expected type of information	Value/ Tolerance	Unit	Note
1. Spacecraft Physical Characteristics				
Dimensions				
Spacecraft frame definition	drawing		-----	
Spacecraft overall (Launch Configuration)	drawing		-----	
Spacecraft height above separation plane	value		mm	
Mass / Dynamic Property				
Spacecraft launch Mass	value	+/-	kg	
Center of gravity -- X axis	value	+/-	mm	
-- Y axis	value	+/-	mm	
-- Z axis	value	+/-	mm	
Moments of inertia – Ixx	value	+/-	kgm^2	
-- Iyy	value	+/-	kgm^2	
-- Izz	value	+/-	kgm^2	
Products of inertia – Ixy	value	+/-	kgm^2	
-- Iyz	value	+/-	kgm^2	
-- Izx	value	+/-	kgm^2	
Fundamental natural frequency -- lateral	value		Hz	
-- longitudinal	value		Hz	
Propellant Sloshing Characteristics				
Number of propellant tanks	value		-----	
For each tank, please fill following characteristics ---	-----	-----	-----	
Propellant type name	value		-----	
Density	value		Kg/m^3	
Tank maximum volume	value		L	
Fill fraction	value		%	
Propellant mass	value		kg	
Tank Center location – X axis	value		mm	
-- Y axis	value		mm	
-- Z axis	value		mm	
Sloshing model – Pendulum mass	value		kg	
– Pendulum attachment point	value		mm	
– Pendulum length	value		mm	
– Pendulum damping ratio	value		ND	
Primary sloshing frequency (1G-model)	value		Hz	

Requirements / Information	Expected type of information	Value/ Tolerance	Unit	Note
2. Keplerian Orbit Injection Requirements (Osculating Orbit at Spacecraft Separation)				
Apogee altitude	value	+/-	km	
Apogee cap	value	+/-	km	
Perigee altitude	value	+/-	km	
Inclination	value	+/-	deg	
Argument of perigee	value	+/-	deg	
Right ascension of ascending node	value	+/-	deg	
Others (if any)	any form		-----	
3. Trajectory / Separation Requirements				
Launch window requirements / constraints	any form		-----	
Maximum free-molecular heat flux	value		W / m^2	
Maximum fairing ascent depressurization rate	value		kPa / s	
Sun angle constraints	any form		-----	
Maneuver or orientation requirements	any form		-----	
In flight contamination requirements	any form		-----	
Separation velocity	value	to or +/-	m / s	
Angular rate at separation -- Roll axis (assumed to be LV thrust axis)	value	+/-	deg / s	
-- Pitch and Yaw axis	value	+/-	deg / s	
Maximum angular acceleration	value		deg / s^2	
Attitude at separation	any form		-----	
Allowable pointing error at separation	value		deg	
Other requirements or constraints	any form		-----	
4. Spacecraft Environmental Condition				
Maximum allowable acceleration (static + dynamic) -- lateral	value		m/s^2	
Maximum allowable acceleration (static + dynamic) -- longitudinal	value	-----	m/s^2	
Allowable sine vibration environment	chart or table		-----	
Allowable acoustics environment	chart or table		-----	
Allowable shock environment	chart or table		-----	
In flight temperature range	value	to	℃	
Prelaunch ground temperature range	value	to	℃	
Prelaunch relative humidity range	value	to	%	
Air cleanliness	value		class	
Battery dedicated ventilation required?	yes / no		-----	
If yes, please show required air source characteristics (duration, flow rate, temperature, humidity, etc.)	any form		-----	

Requirements / Information	Expected type of information	Value/ Tolerance	Unit	Note
5. Interface Requirements				
Fairing / Adapter interface				
Spacecraft launch configuration digital data	CAD data, etc.		-----	
Spacecraft launch configuration sectional drawing showing protrusions under separation plane	drawing		-----	
Spacecraft effective diameter including protrusions (Launch configuration)	value		mm	
Spacecraft/Launch vehicle adapter interface diameter	value		mm	
Maximum spacecraft cross-sectional area	value		mm^2	
Line load constraint at the separation plane	chart or table			
Fairing Mission Modification				
Number of fairing access door	value		-----	
For each access door, please fill following characteristics	-----	-----	-----	
Size	value		mm	
Location (target object)	value (X,Y,Z)		mm	
Purpose	any form		-----	
Fairing RF transparent window requirement?	yes / no		-----	
If yes, please show antenna location	value (X,Y,Z)		mm	
Interstage LOGO Requirement	yes / no or drawing		-----	
Electrical Interface				
Number of umbilical connectors	value		-----	
Umbilical connector pins number or part number	any form		-----	
Umbilical connector location (for each connector)	value (X,Y,Z)		mm	Fairing umbilical is option
Grounding Characteristics	value or spec			
Electrical power from LV (number, specification, etc.)	any form		-----	Option
Dry loop command required?	yes / no		-----	Option
If yes, please show characteristics and operational plan (open/close timing, etc.), and constraints	any form		-----	
Electrical command signal from LV (number, specification, operational plan, etc.)	any form		-----	Option
Electrical signal to be monitored by LV (number, specification, etc.)	any form		-----	Option
Spacecraft radiation emission Level	chart or table		-----	
Acceptable spacecraft radiation susceptibility level	chart or table		-----	
RF command frequency	value		MHz	
RF telemetry frequency	value		MHz	
RF operations plan(timing of RF operations for each frequency from the beginning of spacecraft processing to the spacecraft separation)	Any FORM		-----	

Requirements / Information	Expected type of information	Value/ Tolerance	Unit	Note
6. Ground Operations Requirements				
Operational Plan				
Spacecraft preparation period (until beginning of joint operations)	value		day	
Electrical and RF interface schematics between spacecraft, check-out terminal equipment, and overall checkout equipment for each spacecraft processing phase, and assent phase.	schematic diagram		-----	
Electrical GSE (Please fill following for each GSE)				
Dimensions	value (L x W x H)	x x	mm	
Weight	value		kg	
Electrical power (voltage, current, frequency, power, phase, wire, etc.)	value			
Temperature	value	to	℃	
Humidity	value	to	%	
Cleanliness	value		class	
Allowable acoustic environment, in case it is located at ML GSE Room? (Operated near the spacecraft until lift-off)	chart or table		-----	
Other requirements / constraints	any form		-----	
Mechanical GSE (Please fill following for each GSE)				
Dimensions	value (L x W x H)	x x	mm	
Weight	value		kg	
Maximum floor loading	value		Pa	
Electrical power (voltage, current, frequency, power, phase, wire, etc.)	value			
Temperature	value	to	℃	
Humidity	value	to	%	
Cleanliness	value		class	
Operated location (or Applicable phase)	any form		-----	
Other requirements / constraints	any form		-----	
Propellants Handling (Option)				
Expected to be provided by LV side?	yes / no		-----	
If yes, please fill following for each	-----	-----	-----	
Type	name		-----	
Quantity	value			
Propellant analysis required at launch site?	yes / no		-----	

Requirements / Information	Expected type of information	Value/ Tolerance	Unit	Note
Fluid / Gas Handling (Option)				
Expected to be provided by LV side?	yes / no		-----	
If yes, please fill following for each	-----	-----	-----	
Type	name		-----	
Quality	any form		-----	
Quantity	yes / no			
Transportation Requirements				
Spacecraft transportation (Option)				
Spacecraft delivered location (Airport, TNSC, etc.)	location name		-----	
Spacecraft Container Dimensions	value (L x W x H)	x	mm	
Container weight with spacecraft	value		kg	
Constraint or Resources (power, air, etc.) for Spacecraft Transportation	any form		-----	
Spacecraft transportation (Option)				
Propellant delivered location (Airport, TNSC, etc.)	location name		-----	
Number of containers	value		-----	
Propellant container dimensions (maximum)	value (L x W x H)	x x	mm	
Container weight with propellant (maximum)	value		kg	
Constraint for propellant transportation	any form		-----	
Storage requirements/constraints before filling and after Launch	any form		-----	
GSE Transportation (Option)				
GSE delivered location (Airport, TNSC, etc.)	location name		-----	
Number of GSE containers	value		-----	
GSE container dimensions (maximum)	value (L x W x H)	x x	mm	
Container weight with GSE (maximum)	value		kg	
Constraint for GSE transportation	any form		-----	
Storage requirement after Launch	any form		-----	
Communication Services (Option)				
Dedicated communication with outer launch site area is need?	yes / no		-----	
If yes, please fill following for each	-----	-----	-----	
Communication method (phone, data, etc.)	any form		-----	
Location to communicate with	name		-----	
Miscellaneous Services (Option)				
Number of launch site visit	value		-----	
Working language	name		-----	
Real time launch video requirement	yes / no		-----	
X-Ray inspection requirement	yes / no		-----	
Secretary support	yes / no		-----	
Other services	any form		-----	

Requirements / Information	Expected type of information	Value/ Tolerance	Unit	Note
7. Interface Management				
Interface Test				
Match mate test	yes / no		-----	
Match mate test location	location name		-----	
Match mate test available timing	value		months	months prior to launch
Separation shock test	yes / no		-----	
Separation shock test location	location name		-----	
Separation shock test available timing	value		months	months prior to launch
Other interface test required to be conducted by LV side. (Purpose, configuration, timing, etc.)	any form		-----	
Spacecraft Data / Documentation Availability				
Spacecraft safety data as defined in JMR-002.	any form		-----	
Spacecraft CLA mathematical model	yes / no		-----	
Available Timing (preliminary)	value		months	months prior to launch
Available Timing (final)	value		months	months prior to launch
Spacecraft thermal mathematical model	yes / no		-----	
Available Timing (preliminary)	value		months	months prior to launch
Available Timing (final)	value		months	months prior to launch
Spacecraft test plan with qualification policy	any form		-----	
Launch Vehicle Documentation Requirements				
Required documents (Chapter 5 and 6 and Appendix-3 of this manual shows standard, and please identify if additional document/analysis are required.)	any form		-----	
Review / Meeting Support				
Availability to support review / meeting described in chapter 6.5 of this manual.	yes / no		-----	
Additional review / meeting requirements. (Purpose, timing, etc.)	any form		-----	

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Appendix 3 STANDARD SERVICES & OPTIONS

A3.1 General

This chapter describes standard and optional services. Appendix 3.2 to Appendix 3.6 describes standard services and Appendix 3.7 describes optional services.

A3.2 Mission Management

A3.2.1 Mission integration (General Process)

LSP will manage overall activities as follows:

(1) Management of mission integration (schedule control, documentation, obtaining permissions from domestic authorities.)
(2) Management of Spacecraft/Launch Vehicle integration such as, but not limited to interface control, mission analyses, launch vehicle production, develop documents, etc.
(3) Management of launch site support service at TNSC (preparation/operations of the launch site facilities, safety/security management, communication device arrangements, launch operations assist, logistics support)

The integration of the spacecraft and launch vehicle will be in accordance with documents as described in Appendix 3.2.2 and technical meeting and review as described in Appendix 3.3.5.

A3.2.2 Documentation

A3.2.2.1 Interface Documents

LSP will develop and maintain following document and provide to SCO.

(1) Mission Integration Schedule (MIS)
(2) Interface Control Document(ICD)
(3) Mission Analysis Report
(4) Safety Submission Responses
(5) Launch site Operation Support Plan (LOSP)
(6) Joint Operation Plan(JOP)
(7) Count down Plan
(8) Injection Data
(9) Post Flight Analysis Report
(10) Transportation Logistics Plan from first port of entry to/from launch site (for overseas

customer)

A3.2.2.2 Status Report

LSP will develop following status report periodically to SCO.

(1) Quarterly Report

Quarterly report includes program status, issues such as concern to the launch schedule, major non-conformances, and flight anomalies that apply design changes.

A3.2.3 Licenses /Permissions

LSP will support customer to obtain approval and permission from Japanese government so that SCO ships the spacecraft and associated equipment to Japan, performs launch preparation at the launch site.

(1) Obtain license for RF use (for both domestic and overseas customers)

In case that SCO plans radio frequency emission at the launch site, approval from the Japanese government is required. LSP will obtain the approval for SCO.

(2) Obtain certificate for JAXA High-pressure Gas Standard for spacecraft flight system (for overseas customer)

Spacecraft flight subsystems which accommodate high pressure gas (more than 1MPa) shall comply with JAXA standard, JERG-0-001, and need to obtain certificate from JAXA. LSP will obtain the certificate using data package provided by SCO.

(3) High-pressure gas ground support equipment (for overseas customer)

Any ground support equipment handling high pressure gas, more than 1MPa, such as propellant tank pressurization GSE, shall comply with specifications defined by Japanese law. LSP will support SCO to comply with the law, or LSP will provide equipment which has already been ensured to comply.

(4) Pyrotechnics (for overseas customer)

In case that the spacecraft installs pyrotechnics, or SCO plans to handle pyrotechnics at the launch site, approval from Japanese government is required. LSP will obtain the approval for SCO.

(5) Customs clearance and transport permission (for overseas customer)

LSP will provide support for customs clearance at entry to Japan and reshipment from Japan. LSP will also obtain approval from Japanese local government to transport spacecraft and associated equipment internal Japan.

In case that SCO brings hazardous goods, SCO is required to ensure that any hazardous shipment and package comply with international safety standard such as IATA(International Air Transport Association) and IMDG(International Maritime Dangerous Goods).

A3.3 Spacecraft/Launch Vehicle Integration

A3.3.1 Launch Vehicle Production

Launch vehicle configuration is as follows:

(1) Launch Vehicle hardware : H2A20x configuration with following accommodation;

(2) One (1) Payload Fairing (PLF) with

- (a) Two(2) 600 mm dia. access doors
- (b) One(1) RF transparent window
- (c) One(1) RF internal antenna

(3) One (1) flight Payload Adapter with two (2) SC/LV interface connectors and their brackets, including separation system

(4) One (1) mission logo at LV inter-stage section

A3.3.2 Mission Analyses

See Section6.2.2.

A3.3.3 Launch Operations

LSP will perform launch site activities as following;

(1) Launch vehicle operations

- (a) Assemble the launch vehicle and conduct system check out
- (b) Perform spacecraft / launch vehicle integration, joint operations
  - (i) Mate spacecraft to payload adapter and install separation system
  - (ii) Encapsulate payload fairing
  - (iii) Transport the encapsulated spacecraft to the launch vehicle assembly building (VAB)
  - (iv) Mount the encapsulated spacecraft on the launch vehicle second stage at VAB
- (c) Countdown rehearsal
- (d) Countdown operations and launch vehicle flight data monitor.

(2) Support for spacecraft processing

LSP will support SCO for its spacecraft processing. See Appendix 3.4 and Appendix 3.5 for more detail.

A3.3.4 Program Assurance

LSP will conduct launch vehicle product assurance activities in accordance with H-IIA's own certified process. LSP will report the status and result to SCO. In the event of any launch vehicle production issues that affect the spacecraft launch schedule, LSP will inform SCO immediately.

A3.3.5 Review and Meeting

See Section6.5.

A3.3.6 Support for the Spacecraft Verification

A3.3.6.1 Spacecraft Safety Reviews

See Section 6.4.

A3.3.6.2 Spacecraft Verification Test Support

(1) Sinusoidal Vibration Test

SCO is supposed to perform sinusoidal vibration test as its verification.

LSP will support the sinusoidal vibration test, such as notching condition. Details will be discussed through the TIM.

(2) Match-mate Test

In the payload adapter manufacturing process, LSP will conduct match-mate test, which includes,

(a) Prepare the flight model payload adapter with interface connectors, their brackets and associated harnesses.

(b) Mate spacecraft onto the payload adapter and install separation system under SCO assistance.

A3.4 Launch Site Support Services

A3.4.1 Support Period

Support duration is 45 calendar days with following conditions. (Support duration may vary depending on the contract.)

(1) Saturdays and Sundays are generally off. (except for pre-agreed activities and except after spacecraft/fairing encapsulation)
(2) Standard support hours: 8 o'clock to 17 o'clock (8 hours: one shift)
(3) LSP can support SCO extra hours or Saturdays /Sundays with advance notice.

A3.4.2 Launch Site Survey

LSP will arrange two (2) opportunities for SCO to survey launch site facilities before launch campaign.

A3.4.3 Coordination of Launch Site Activities

LSP will provide SCO with following daily coordination.

(1) Daily meetings as described in Section 6.5.
(2) Weather information twice per day.
(3) Security management
    (a) Obtain ID cards for SCO personal.
(4) Safety management
    (a) Provide the general safety training
    (b) Provide the safety training for hazardous processing
    (c) Arrange witness of JAXA safety officer for hazardous processing
(5) Countdown operations support

A3.4.4 Launch Site Preparation for Spacecraft Activities

LSP will have completed following preparation prior to the spacecraft activities.

(1) Configure RF link system.
(2) Configure network system.
(3) Prepare clean room.
(4) Prepare clean room garments.
(5) Prepare personal handy phones (PHS).
(6) Prepare safety equipment for propellant loading activity. (SCAPE suites, toxic gas detector, scrubber, breath air, liquid waste disposal, etc.)
(7) Locate the high pressure blast shield.
(8) Calibrate the spacecraft weight measurement equipment.
(9) Configure launch site visual monitor (ITV) and their intra-site distribution.
(10) Connect power supply terminal.
(11) Prepare various equipment.(thermostat bath, alignment equipment, spacecraft purge

support equipment, Hyrda Set®, high pressure gas supply equipment (for air pad), various toxic gas monitors (built in SFA, portable(SPM, CM4), palmtop(Dräger Pac III)), O_2 sensor, man lift, cherry picker, fork lift, GSE carrying car, SC Fairing trailer, tractor, truck)

A3.4.5 Support for Spacecraft Processing

LSP will support spacecraft processing as follows.

(1) Provide instruction on the STA2, SFA, VAB and RCC facilities.
(2) Maintain clean room environment.
(3) Monitor and record temperature, humidity and particle, Non Volatile Residue (NVR) for spacecraft processing room(s).
(4) Operate safety equipment for propellant loading activity.
(5) Analyze fluid and gas (particle count, chemical compositions)
(6) Operate the launch site RF system.
(7) Adjust scaffold platform so that customer can access spacecraft just before fairing encapsulation.
(8) Adjust scaffold platform so that customer can access spacecraft through fairing access door after fairing encapsulation.
(9) Provide qualified operator for crane and forklift. (for overseas customer)

A3.4.6 General Consumables

LSP will provide following general consumables for launch facilities.

(1) Deionized water
(2) IPA
(3) Disposable face masks and gloves for clean room

A3.4.7 General Assistant Service

LSP will provide following services for SCO;

(1) Offices and meeting rooms
  (a) Office room(s) for SCO's personnel which can accommodate approx. 30 people.
  (b) Meeting room(s) which has a speaker phone that SCO can use for telecon between TNSC and SCO site.
  (c) Copier and printer.

(2) Preparation of network

Network interface port (Ethernet hub) in above office room(s) which enables SCO to access internet

(3) Photographic recording per SCO request location (during joint operations)
(4) Provide pre-launch and lift-off video by internet streaming

A3.4.8 Transport Services

(1) Inter-Site Transport

LSP will provide spacecraft and equipment transportation between the facilities in TNSC inter-site.

(a) Spacecraft

LSP will transport the spacecraft between the facilities using a transporter and a dedicated container or fairing.

(b) GSE and related equipment

LSP will transport the equipment (GSE) between the facilities.

(2) Transport to launch site (for overseas customer)

LSP will transport SC, GSE, and other equipment from Japanese first entry port (sea port or airport) to the launch site.

SCO is required to transport them from SCO site to Japanese first entry port.

LSP will support SCO to clear Japanese customs.

(3) Transport from launch site (for overseas customer)

After launch, LSP will transport spacecraft container, GSEs, other equipment from launch site to a Japanese port (sea port or airport).

SCO is required to transport them from the port to SCO site.

A3.5 Spacecraft processing Facilities

LSP will provide following facilities for spacecraft processing.

A3.5.1 No.2 Spacecraft Test and Assembly Building (STA2)

(1) One (1) preparation hall in STA2

For the spacecraft integration, functional test and check out.

(2) One (1) checkout room in STA2

For controlling the spacecraft, including real-time monitor of the telemetry.

(3) Office room

Office rooms for SCO personnel.

[Note] There is another processing facility, No.2 Spacecraft and Fairing Assembly Building(SFA2), where SCO is allowed performing both non-hazardous and hazardous processing.

A3.5.2 Spacecraft and Fairing Assembly Building (SFA)

(1) Filling and assembly hall and the checkout room in SFA

For spacecraft propellant loading and fairing encapsulation.

A3.5.3 Vehicle Assembly Building (VAB) and Movable Launcher (ML)

(1) One(1) ML-GSE room for spacecraft GSE.

(2) One(1) GSE room is available at the 9th or 10th floor level at VAB for spacecraft preparation.

A3.5.4 Takesaki Range Control Center (RCC)

(1) Console for a customer representative in Takesaki Range Control Center (RCC) during final countdown where SCO is requested to state final determination of Go/No-Go for launch from SCO perspective.

(2) One (1) office room in RCC during final countdown phase.

A3.5.5 Other Facilities

(1) Storage facilities for propellant tank.

(2) TNSC cafeteria

A3.6 Communication System

LSP will provide the following communication systems.

Table A3.6-1 Available communication system

No	Item	Type	Note
1	RF-link	S/C/X/Ku band (via single-mode fiber lines)	STA2 checkout room ⇄ LV or ML mast STA2 checkout room ⇄ SFA See Section 4.7.7.
2	Umbilical-link	Copper line	GSE ⇄ Spacecraft See Section 4.7.5. 2X61 pins DBAS line 2XCoaxial DBAS line
3	GSE link	10/100base-T (via 100base-F, single-mode fiber lines)	Console / Monitor ⇄ GSE See Section 4.7.5. STA2 checkout room ⇄ ML GSE room STA2 checkout room ⇄ SFA
4	Dedicated LAN	10base-T	STA2 Office room ⇄ STA2 Checkout room
5	Internet	Wired(JAXA-net), or Wireless(Internet Service Provider)	Communication throughput speed is approximately 5Mbps.
6	UTC time indication equipment	IRIG-B	Cannon Connector XLR5-31
7	GPS signal	Coaxial	For Network Time Protocol Server
8	ITV	Closed network TV	For processing monitor on LAN
9	OIS	15 channel	Operational Inter-communication System For SCO, 1 channel assigned.
10	Paging System	Wireless	For propellant loading activity
11	Cell Phone	Inside TNSC	Extension telephone system
12	Explosion Proof Cell Phone	Inside TNSC	
13	Wired telephone	Inside TNSC	
14	International Telephone		Up to pre-defined amount by contract
15	Video Conference system	ISDN(NTT*)	Not operated in IP on internet.
16	Tele-conference system	Public telephone line	
17	Facsimile	Public telephone line	Option

*) NTT (Nippon Telegraph and Telephone Corporation) is one of telecommunications company in Japan.

Appendix 3

A3.7 Options

A3.7.1 Optional Services List

Table 3.7-1 shows overview of optional services.

A3.7.2 Major Optional Activities

A3.7.2.1 Spacecraft Propellant Loading (for chemical propulsion system)

(1) Procure propellant
(2) Prepare propellant loading equipment
(3) Fill spacecraft tank with propellant
(4) Pressurize flight tank
(5) Prepare contingency off-loading equipment
(6) Support propulsion system checkout

A3.7.2.2 Separation Shock Test

SCO is required to verify that the spacecraft withstands the shock generated by the separation.

As support of the shock test, LSP will ;

(1) Prepare payload adapter with separation system and ship to/from SCO test site.
(2) Final mate of the spacecraft to the payload adapter
(3) Install separation system
(4) Initiate separation.
(5) Review the test report generated by SCO.

Table A3.7-1 Optional Launch Services

No	Items	Note
1	Hard ware	
(1)	Additional access door(s)	Standard: 2 doors. See Section 4.5.4.1.
(2)	Additional RF transparent window	Standard: 1 transparent window. See Section 4.5.4.3.
(3)	Additional antenna	Standard: 1 internal antenna. See Section 4.5.4.4.
(4)	Command	Pyrotechnic command, electrical command, dry loop command. See Section 4.7.4.
(5)	Power Supply to spacecraft	See Section 4.7.4.
(6)	In-Flight telemetry via H-IIA	See Section 4.7.4.
2	Test Support	
(1)	Separation/shock test	See Appendix 3.7.2.2 for details.
(2)	Match mate test in TNSC	
3	Additional mission analysis	
4	Support in launch campaign	
(1)	Spacecraft propellant loading	See Appendix 3.7.2.1 for details.
(2)	Launch campaign extension	
(3)	General assistant/translator	Translator, hotel/airline reservation, medical support services, etc.
(4)	Commuter bus service between hotel area and launch site	
(5)	Clean room garments laundry (for SCO's own garments)	
(6)	Additional shipment of S/C GSE from Japan first entry port to the launch site, and return.	
(7)	Long distance phone charge	Up to pre-defined amount by contract
(8)	Facsimile	

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Appendix 4 PAYLOAD ADAPTERS AND PAYLOAD FAIRINGS

Several types of payload adapter are available as follows.

(1) Model 1194M shown in Appendix 4.1
(2) Model 1666MA shown in Appendix 4.2

Except 3 points shown below, low shock payload adapter Model 1194LS-H interface condition is same as Model 1194M, and low shock payload adapter Model 1666LS-H interface condition is same as Model 1666MA. Contact LSP for details.

i) Separation shock environment*
ii) Clamp bands maximum tension
iii) Stay-out zone around the PLA
iv) Separation system keyway position

*) See Section 3.2, Figure 3.2-4 for separation shock environment of low shock payload adapter.

The information regarding H-IIA payload fairings are shown in Appendix 4.3.

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Appendix 4.1 1194M ADAPTER

The main characteristics are as follows.

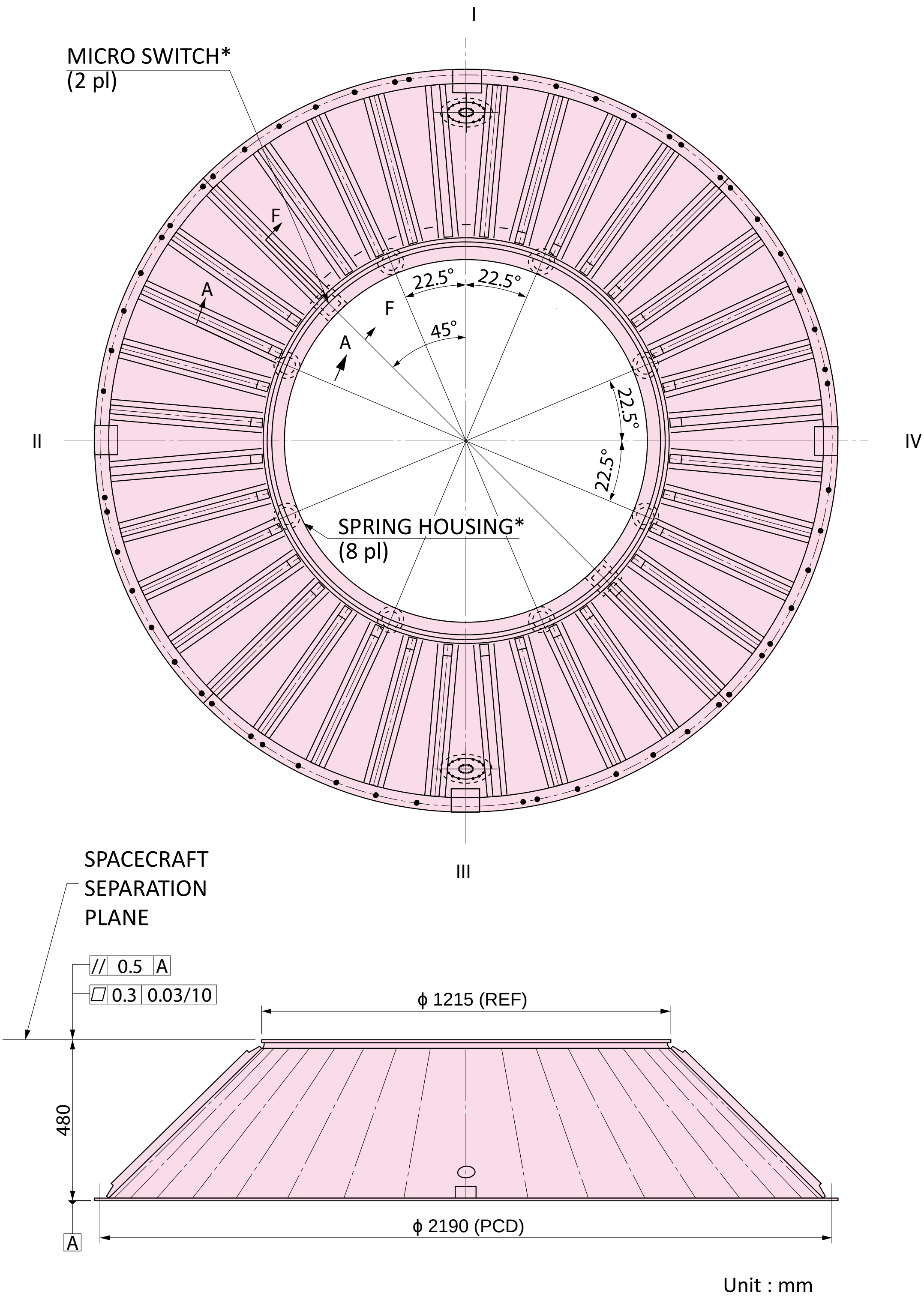
(1) Interface diameter	: 1,215 mm
(2) Height	: 480 mm
(3) Material	: Aluminum Semi-monocoque
(4) Attached system	: Clamp bands
(5) Separation springs	: 4 - 8 springs
(6) Clamp band Maximum tension	: 36.8 kN (32.0kN for Model 1194LS-H)
(7) Maximum load per spring	: 1,670 N
(8) Adapter mass	: 100 kg

This adapter has two (2) vent holes of 45 mm ø (1590.4 mm^2) assuming that the internal volume of the spacecraft is less than 2 m^3.

When interface connectors are installed on the separation plane, connectors are located 789.125 mm from the center of the vehicle axis.

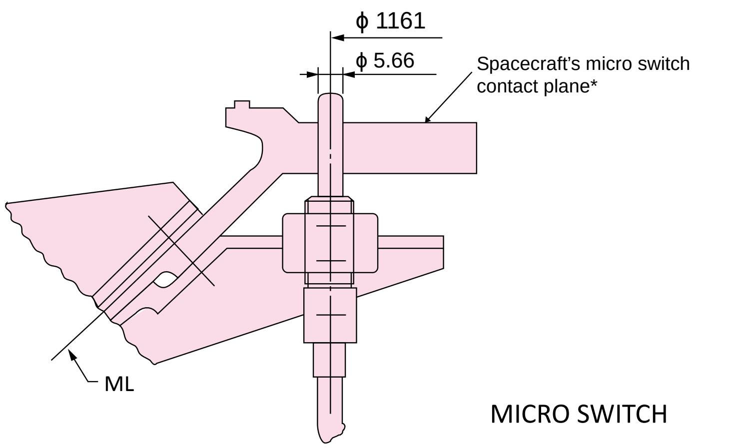
Figure A4.1-1 shows the photograph of the 1194M adapter.
Figure A4.1-2 shows a general view of the 1194M adapter.
Figure A4.1-3 to Figure A4.1-5 show details of the 1194M adapter.
Figure A4.1-6 to Figure A4.1-7 show the stay-out zone around the 1194M adapter.
Figure A4.1-8 shows the usable volume below 1194M adapter spacecraft separation plane.
Figure A4.1-9 shows the limit load of the 1194M adapter.
Figure A4.1-10 shows the spacecraft separation shock spectrum of the 1194M adapter.
Figure A4.1-11 shows the limit load at separation plane of the 1194M adapter.

Figure A4.1-1 Photograph of the 1194M adapter



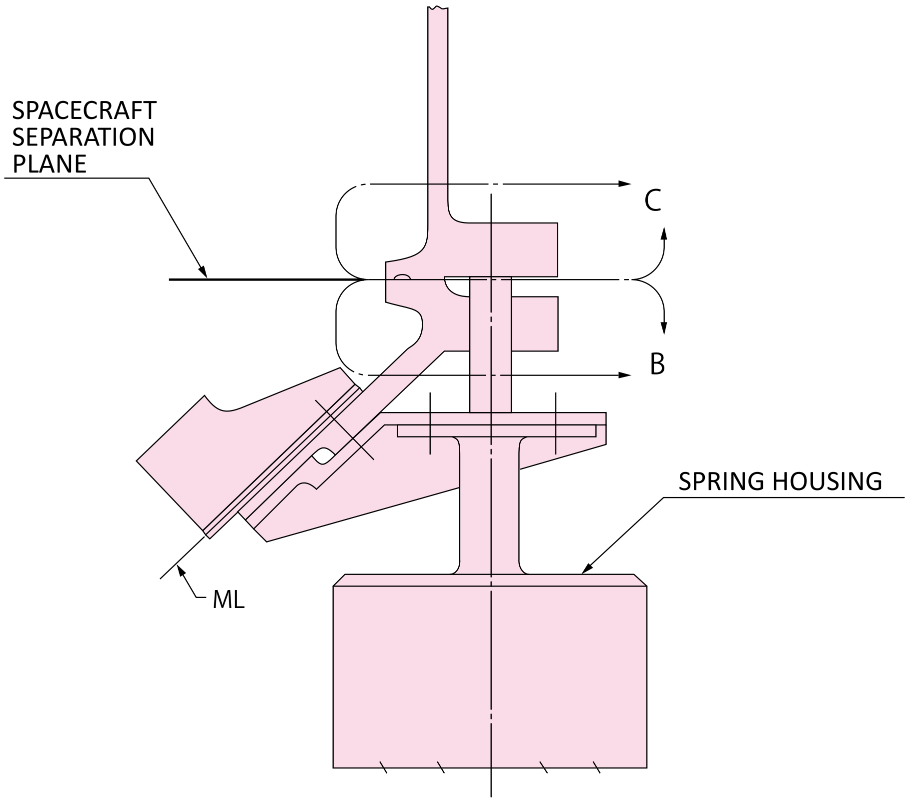
*) Spacecraft's micro switches can be installed in the area expect spring housing(8pl) center ±50mm and launch vehicle's micro switch(2pl) center ±50mm .

Figure A4.1-2 General view of 1194M adapter



*) Spacecraft's micro switches can be installed in the area expect spring housing(8pl) center ±50mm and launch vehicle's micro switch(2pl) center ±50mm .

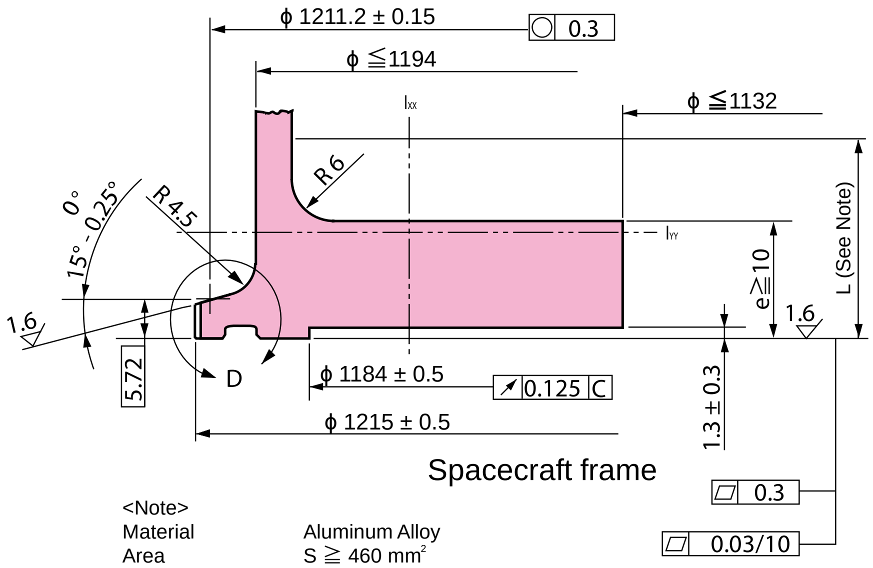
SECTION F–F



SECTION A–A

Unit : mm

Figure A4.1-3 Details of 1194M adapter #1 (Details of separation plane)

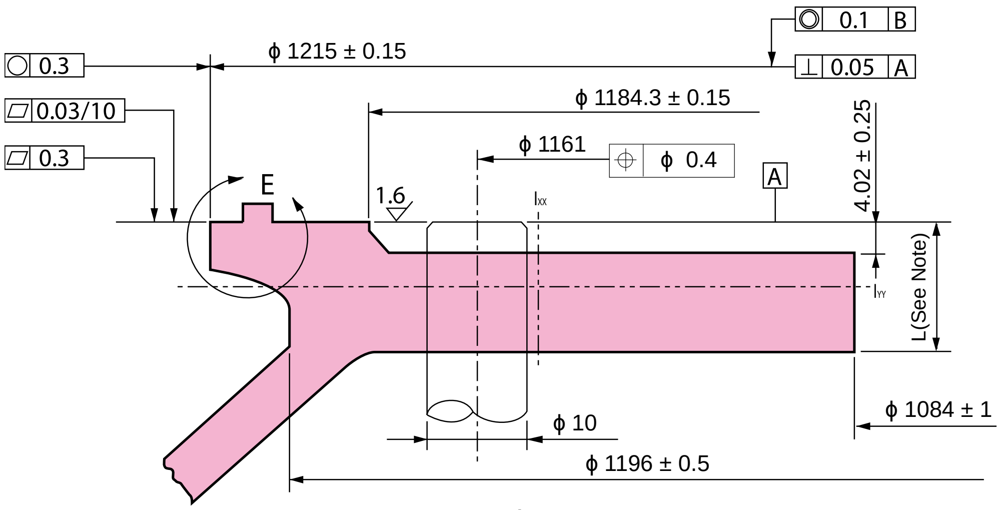


Spacecraft frame

<Note>

Material	Aluminum Alloy
Area	S ≧ 460 mm^2
Inertia	I_{XX} ≧ 5.1 x 10^4 mm^4
	I_{YY} ≧ 1.2 x 10^4 mm^4
Applicable Length	L = 25 mm
	No MLI around applicable length region

Detail C



Adapter

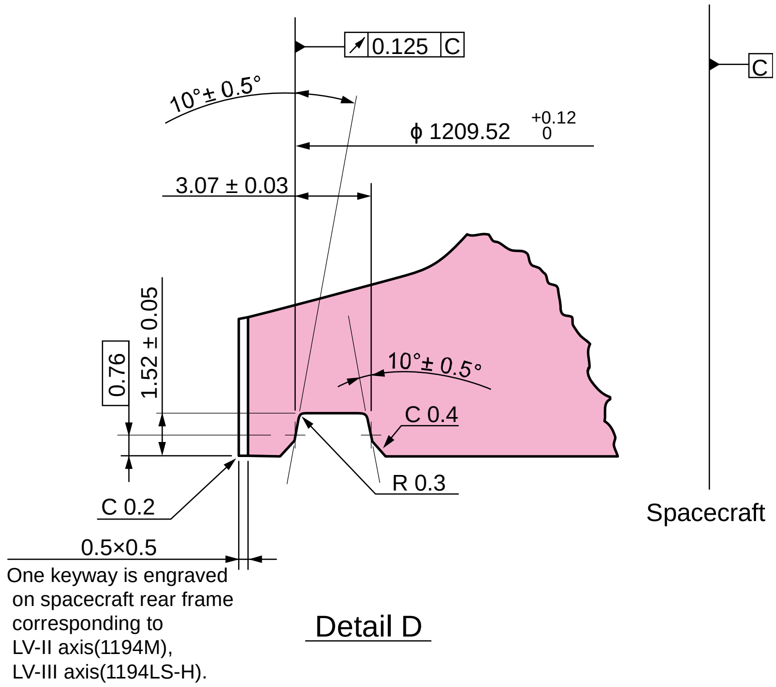
<Note>

Material	Aluminum Alloy
Area	S = 770 mm^2
Inertia	I_{XX} ≧ 2.0x10^5 mm^4
	I_{YY} ≧ 1.3x10^4 mm^4
Applicable Length	L = 18mm

Detail B

Unit : mm

Figure A4.1-4 Details of 1194M adapter #2 (Cross section of frames)



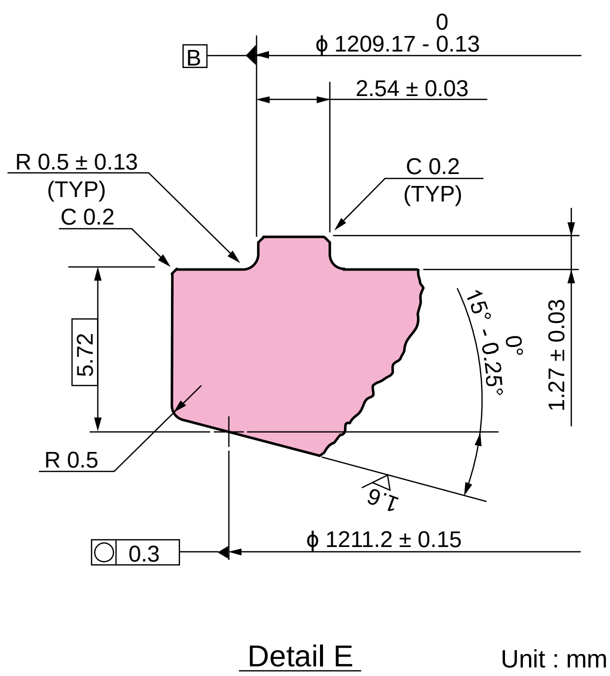
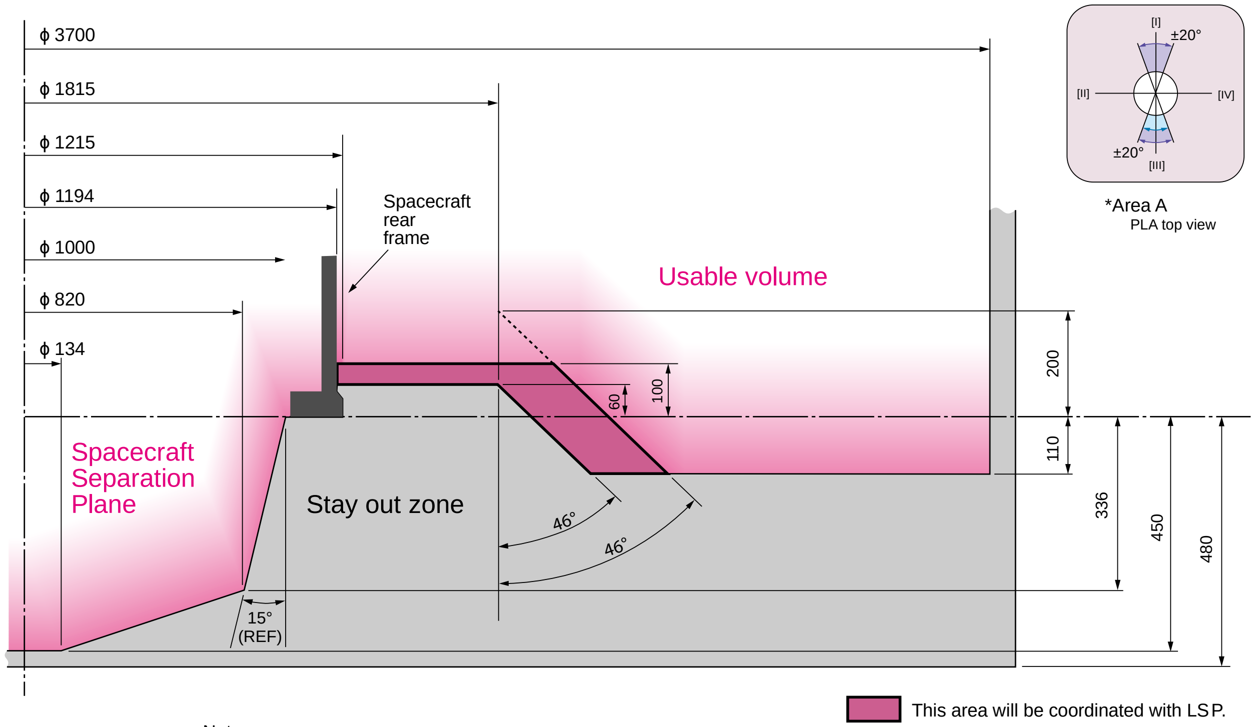
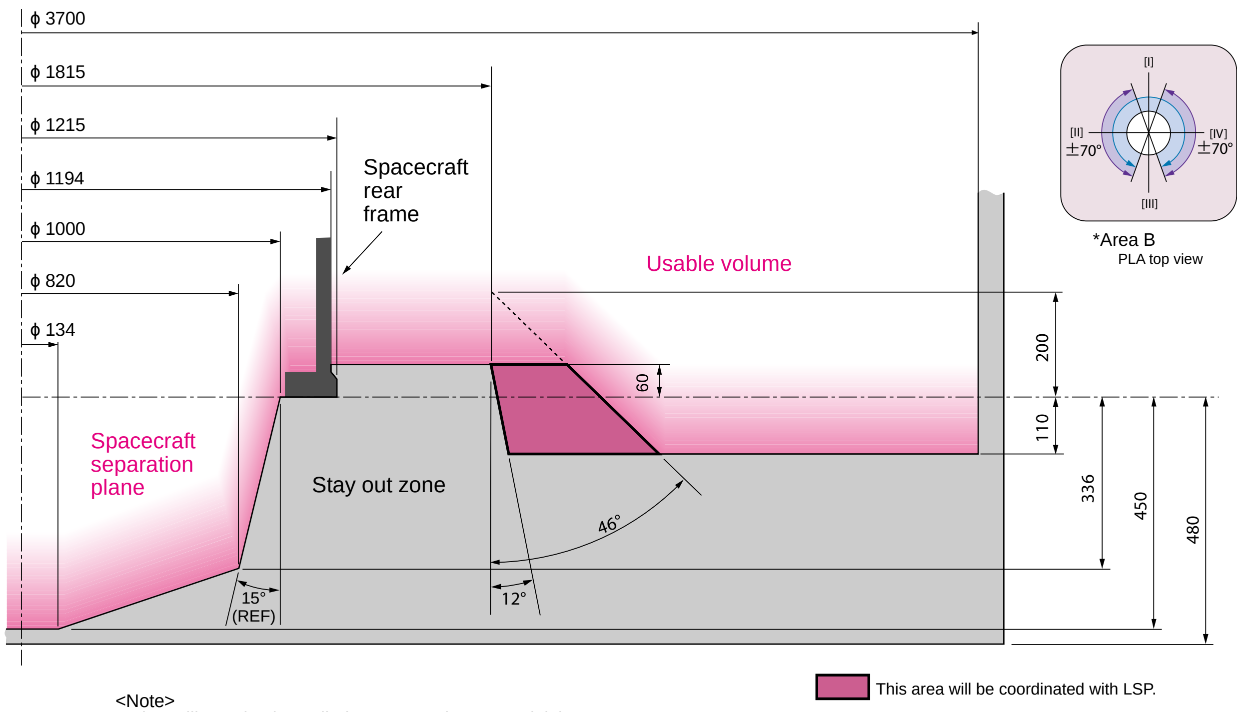


Figure A4.1-5 Details of 1194M adapter #3 (Details of frames)



<Note>
Above illustration is applied to “Area A”(* see top right).
For Model 1194M : “Area A” are III-axis ±20°area and I-axis ±20°area(shown in purple) .
For Model 1194LS-H : “Area A” is III-axis ±20°area(shown in blue).

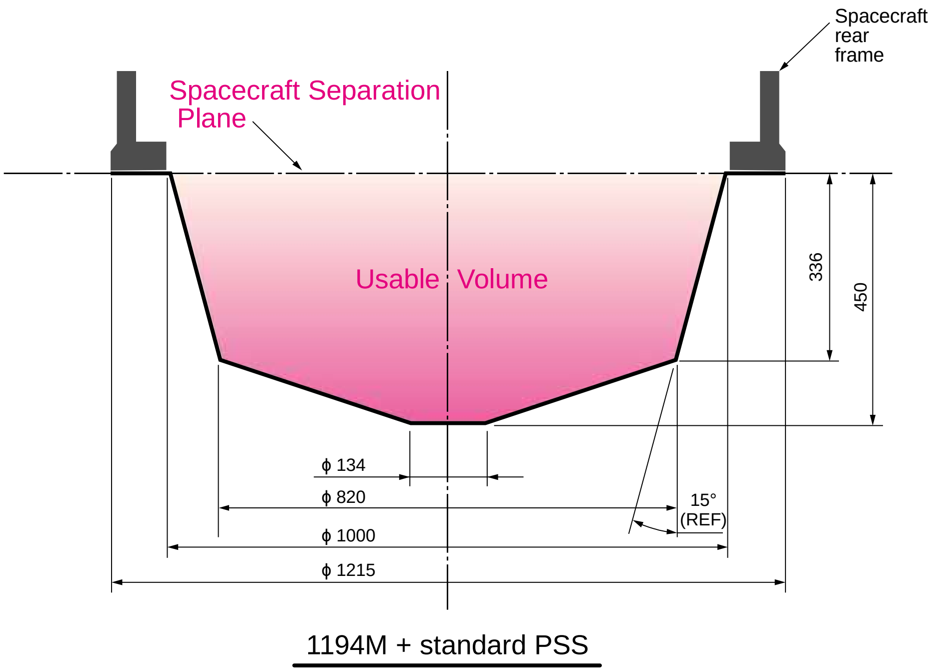
Figure A4.1-6 Stay-out zone around the 1194M adapter (I/III)



<Note>
Above illustration is applied to “Area B”(* see top right).
For Model 1194M : “Area B” are II-axis ±70°area and I V-axis ±70°area(shown in purple) .
For Model 1194LS-H : “Area B” is I-axis ±160°area(shown in blue).

Unit (mm)

Figure A4.1-7 Stay-out zone around the 1194M adapter (II/IV)



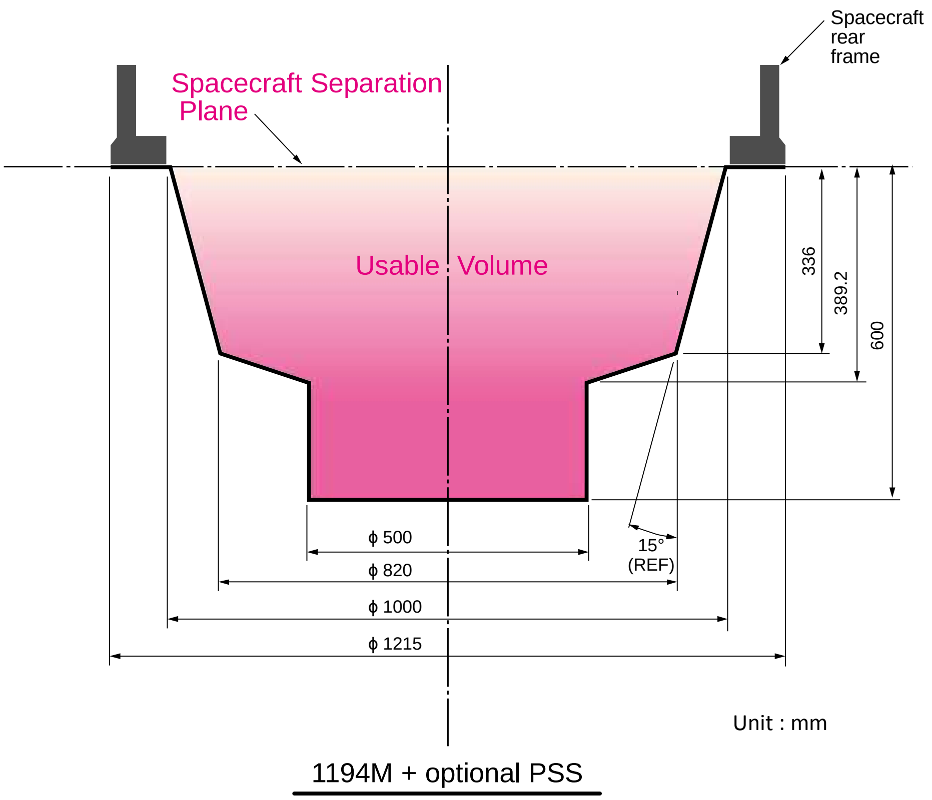
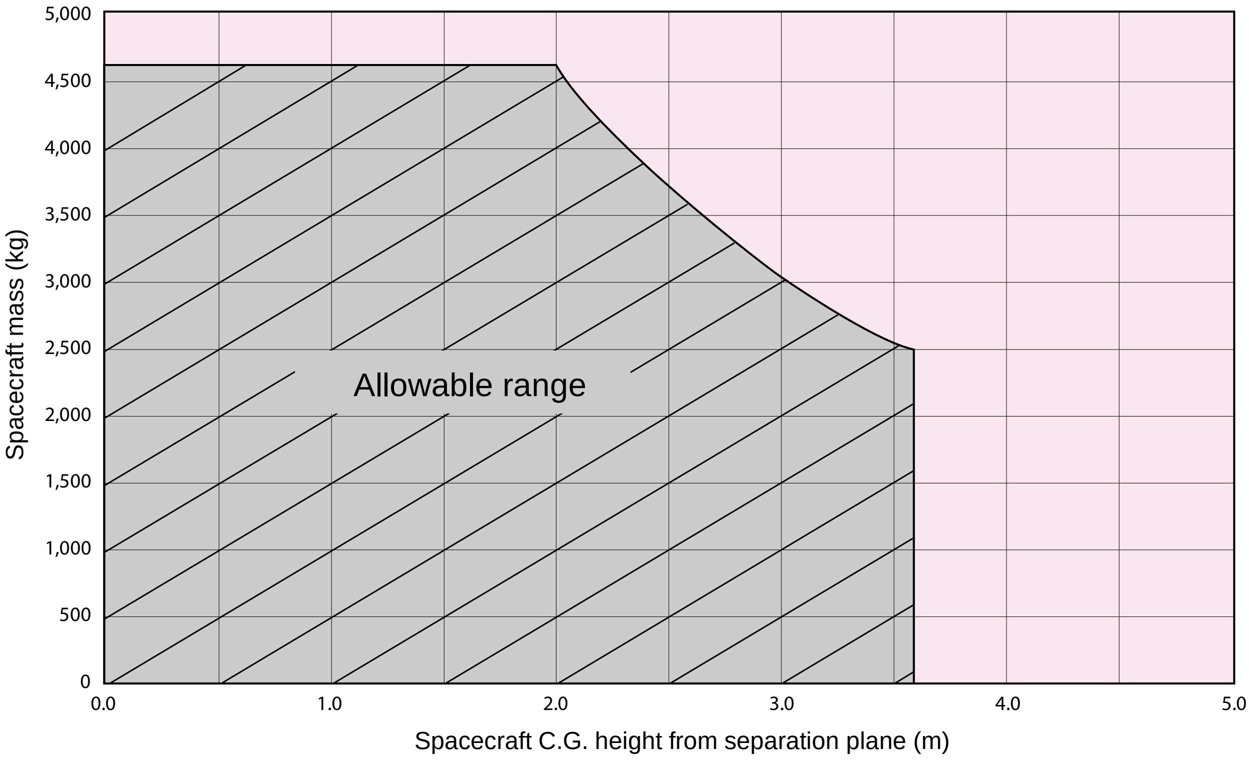
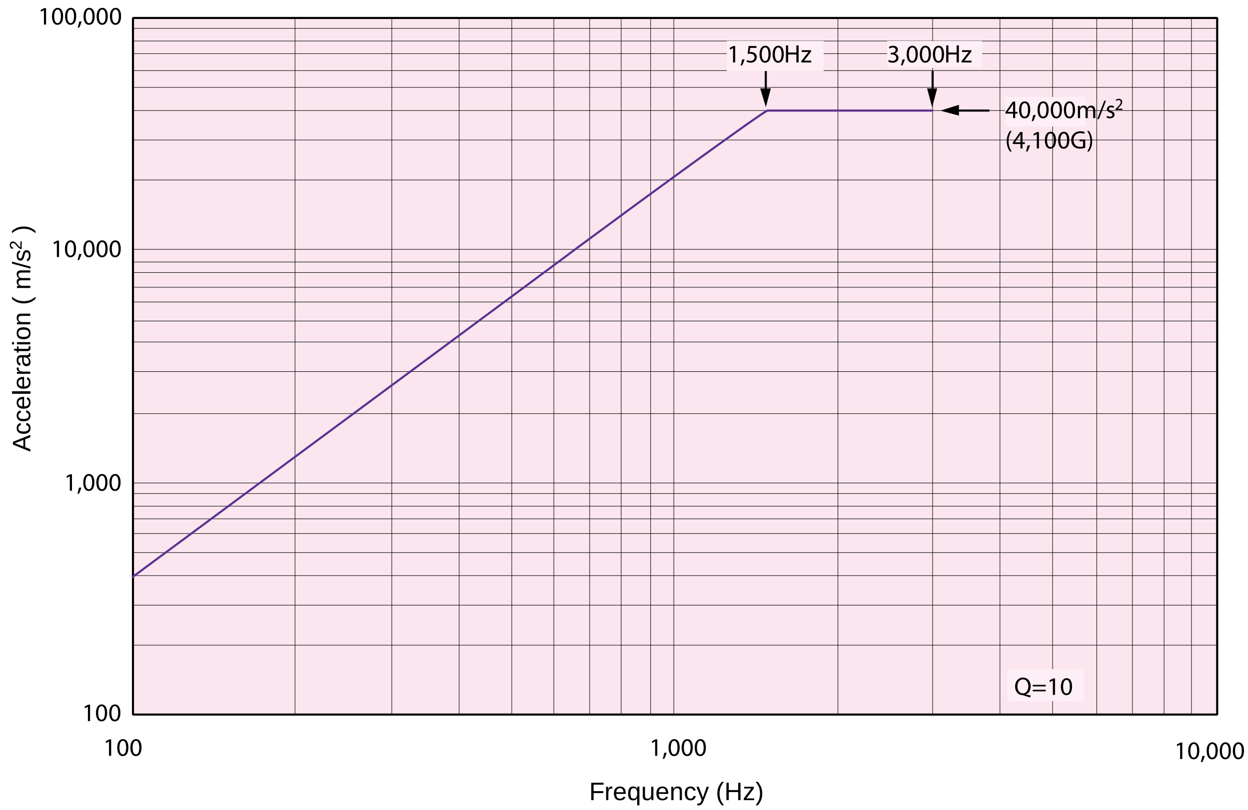


Figure A4.1-8　Usable volume inside 1194M adapter



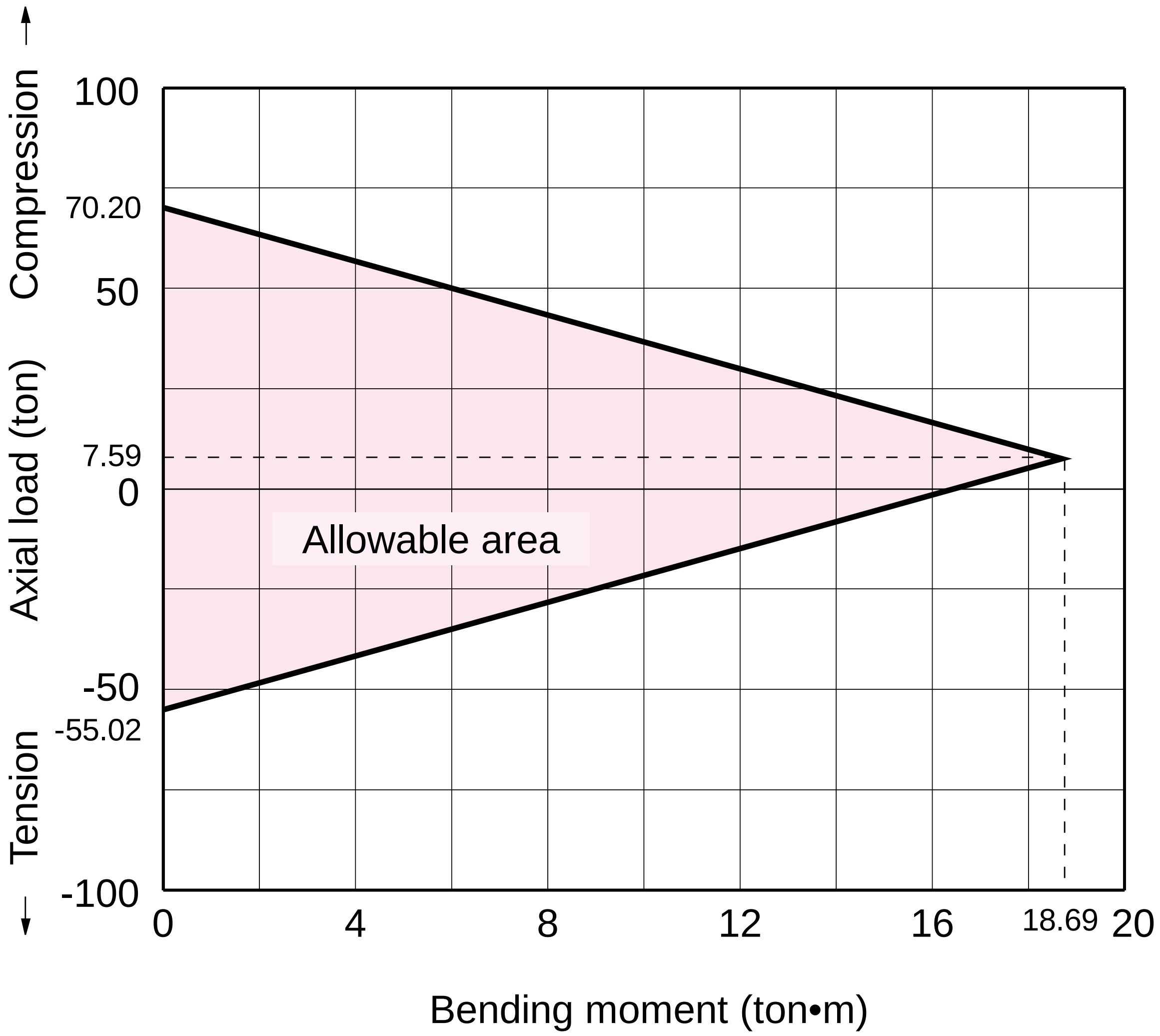
If the spacecraft mass is over 4600kg, please contact LSP for detail.
This view graph shows non- C.G. offset case in the radial direction.
It is required to be less than 10% variation in the load distribution in the circumferential direction of the satellite.
Detailed data will be shown in TIM.

Figure A4.1-9 Limit load of the 1194M adapter



* Separation shock spectrum of 1194M is measured 50mm above separation plane.

Figure A4.1-10 Spacecraft separation shock spectrum of the 1194M adapter



This view graph shows non- C.G. offset case in the radial direction.
It is required to be less than 10% variation in the load distribution in the circumferential direction of the satellite.
Detailed data will be shown in TIM.

Figure A4.1-11 Limit load at separation plane of the 1194M adapter

Appendix 4.2 1666MA ADAPTER

The main characteristics are as follows.

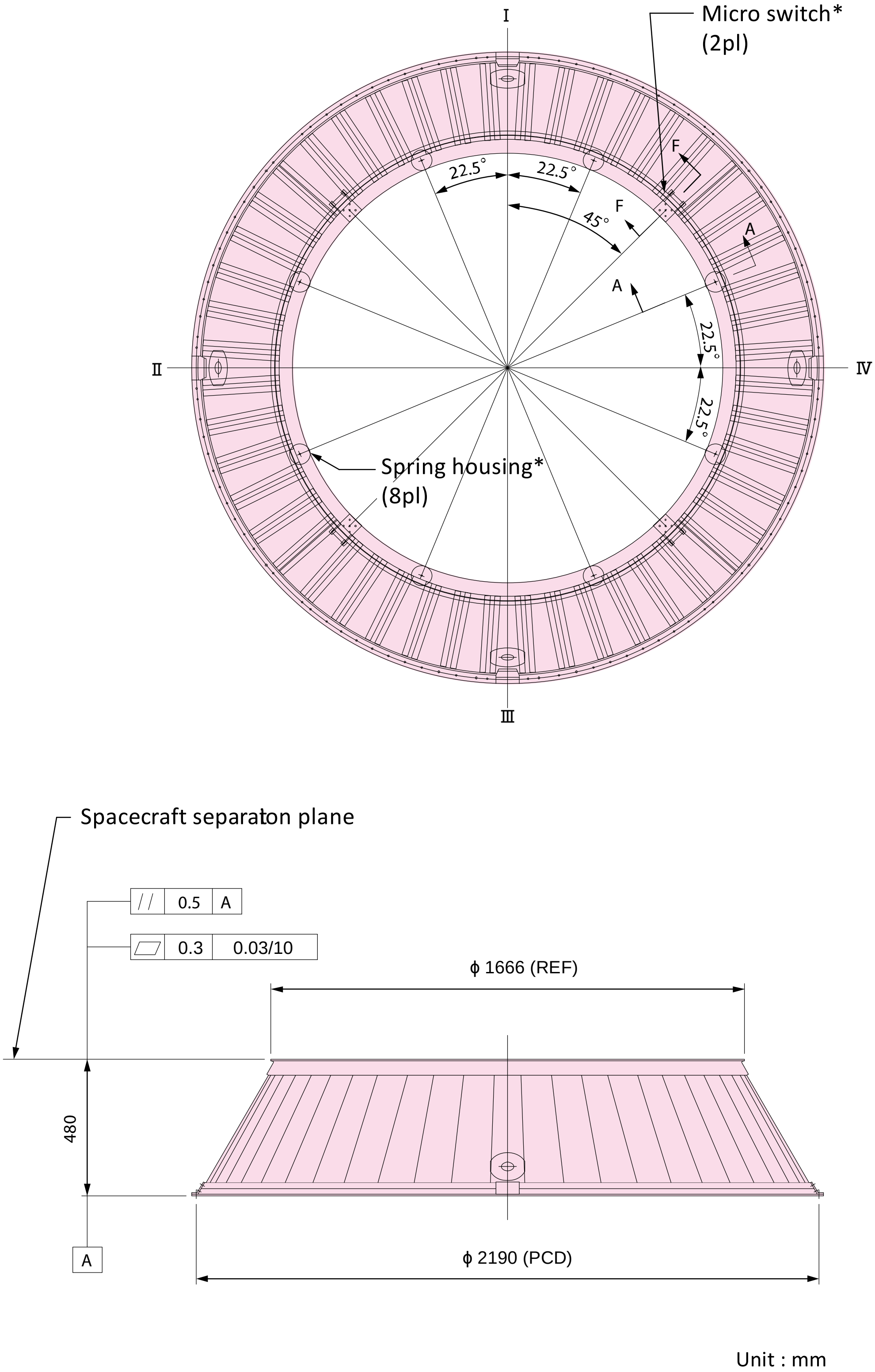
(1) Interface diameter	: 1,666 mm
(2) Height	: 480 mm
(3) Material	: Aluminum Semi-monocoque
(4) Attached system	: Clamp bands
(5) Separation springs	: 4 - 8 springs
(6) Clamp bands	
Maximum tension	: 38.9 kN (32.0kN for Model 1666LS-H)
(7) Maximum load per spring	: 1,670 N
(8) Adapter mass	: 100 kg

This adapter has four (4) vent holes of 45 mm ø (1590.4 mm^2) assuming that the internal volume of the spacecraft is less than 1.5 m^3.

When interface connectors are installed on the separation plane, connectors are located 942.5 mm from the center of the vehicle axis.

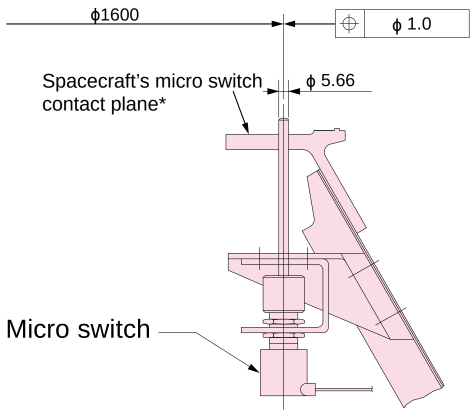
Figure A4.2-1 shows the photograph of the 1666MA adapter.
Figure A4.2-2 shows a general view of the 1666MA adapter.
Figure A4.2-3 to Figure A4.2-5 show details of the 1666MA adapter.
Figure A4.2-6 to Figure A4.2-7 show the stay-out zone around the 1666MA adapter.
Figure A4.2-8 shows the usable volume below 1666MA adapter spacecraft separation plane.
Figure A4.2-9 shows the limit load of the 1666MA adapter.
Figure A4.2-10 shows the spacecraft separation shock spectrum of the 1666MA adapter.
Figure A4.2-11 shows the limit load at separation plane of the 1666MA adapter.

Figure A4.2-1 Photograph of the 1666MA adapter



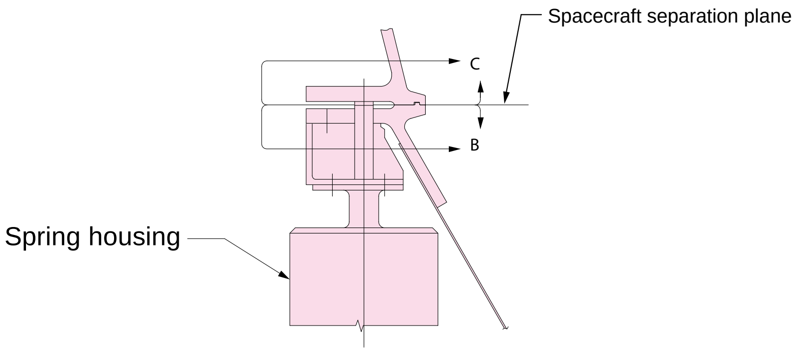
*) Spacecraft's micro switches can be installed in the area expect spring housing(8pl) center ±50mm and launch vehicle's micro switch(2pl) center ±50mm .

Figure A4.2-2 General view of the 1666MA adapter



*) Spacecraft's micro switches can be installed in the area expect spring housing(8pl) center ±50mm and launch vehicle's micro switch(2pl) center ±50mm .

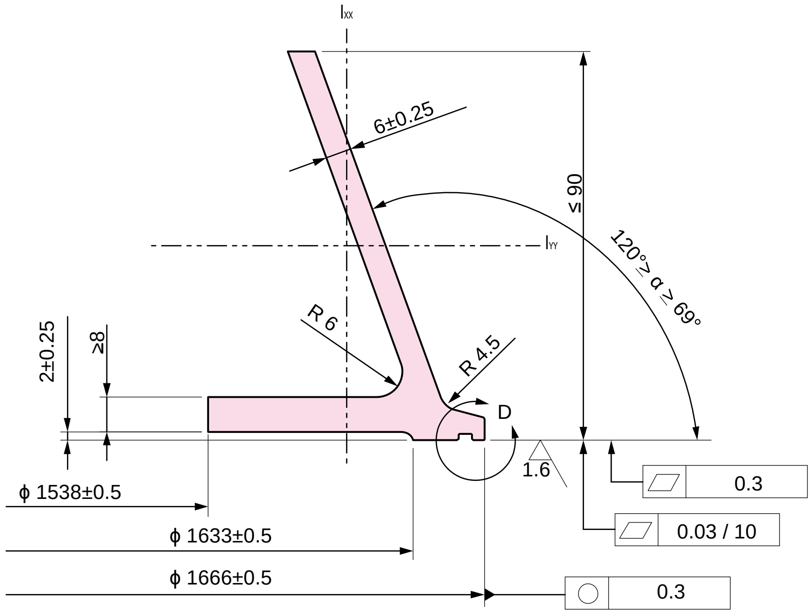
Section F-F



Section A-A

Unit : mm

Figure A4.2-3 Details of the 1666MA adapter #1

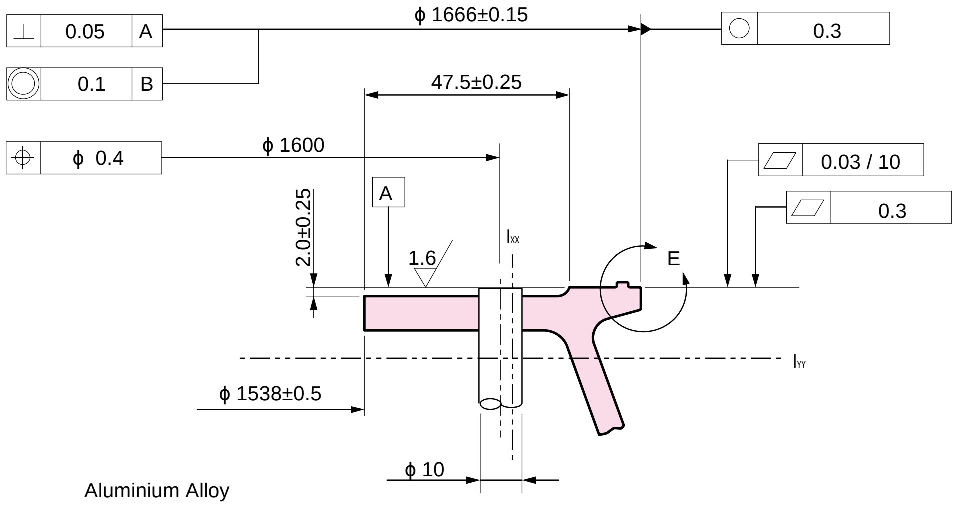


<Note>

Material	Aluminium Alloy
Area	S ≧ 804 mm^2
Inertia	Ixx ≧ 652,590 mm^4
	Iyy ≧ 64,570 mm^4
Applicable length	L ≦ 90 mm
	No MLI around applicable length region

Spacecraft frame

Detail C



<Note>

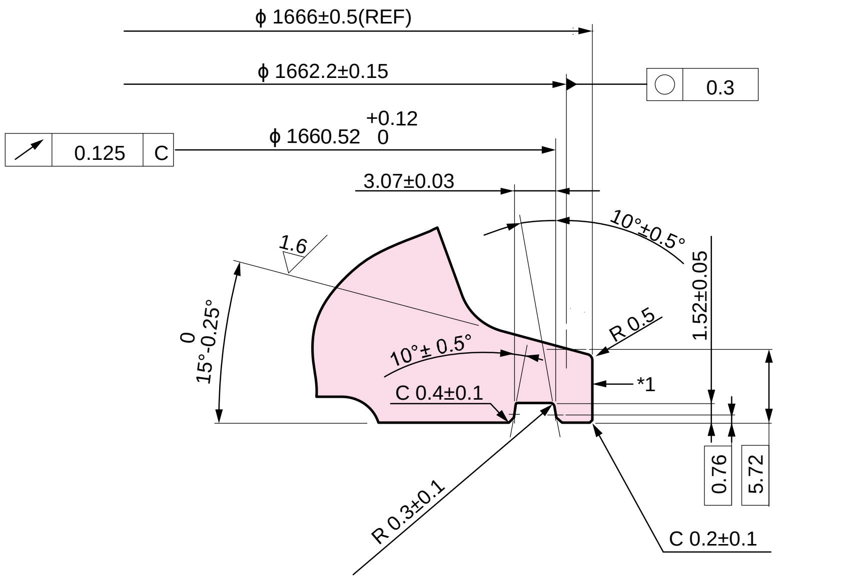
Material	Aluminium Alloy
Area	S = 741mm^2
Inertia	Ixx = 1.80×10^5 mm^4
	Iyy = 3.39×10^5 mm^4
Applicable length	L = 55 mm

Adapter

Detail B

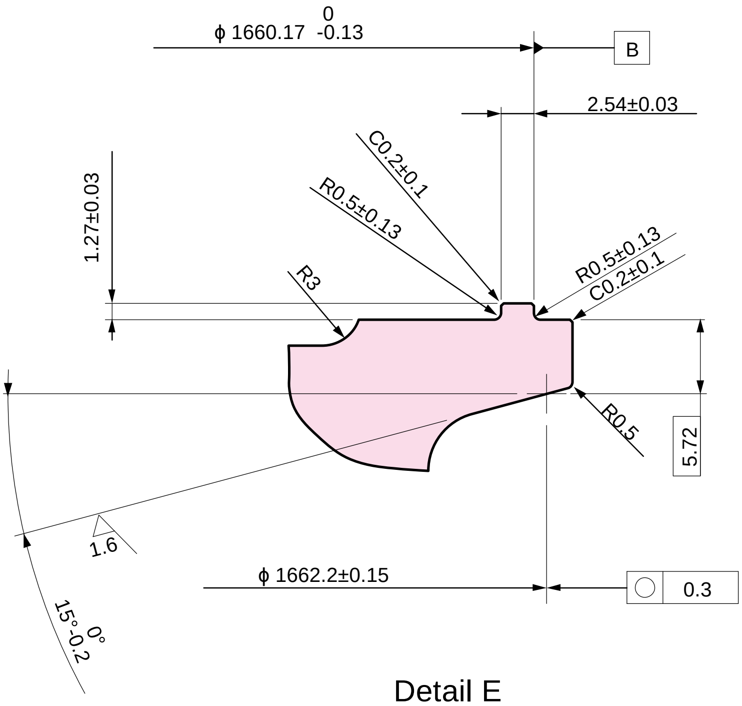
Unit : mm

Figure A4.2-4 Details of the 1666MA adapter #2



*1) One keyway is engraved on spacecraft rear frame corresponding to LV-II axis(1666MA), LV-III axis(1666LS-H).

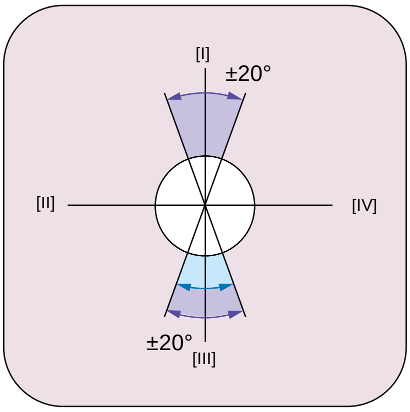
Detail D



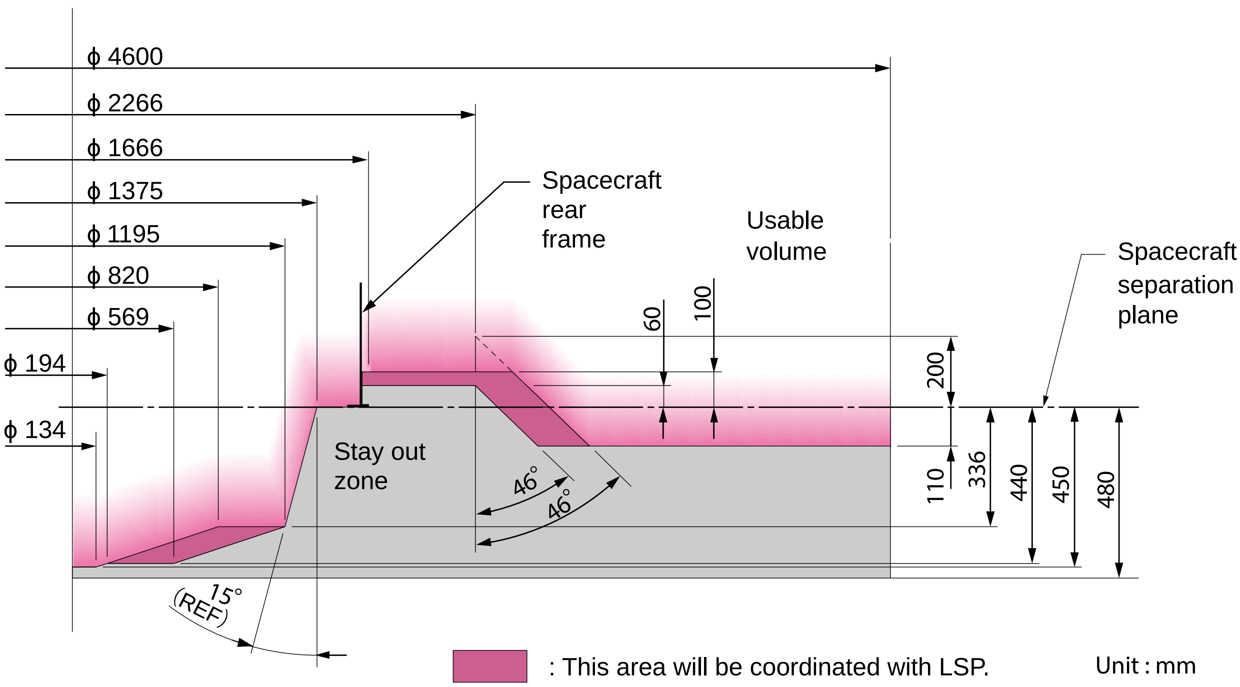
Detail E

Unit : mm

Figure A4.2-5 Details of the 1666MA adapter #3

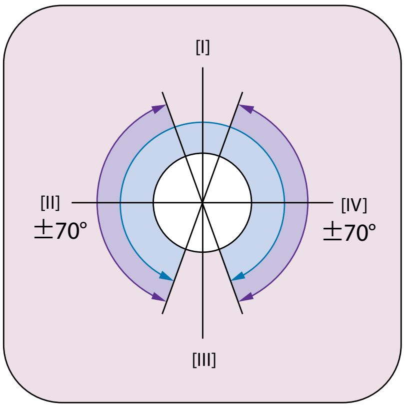


*Area A
PLA top view

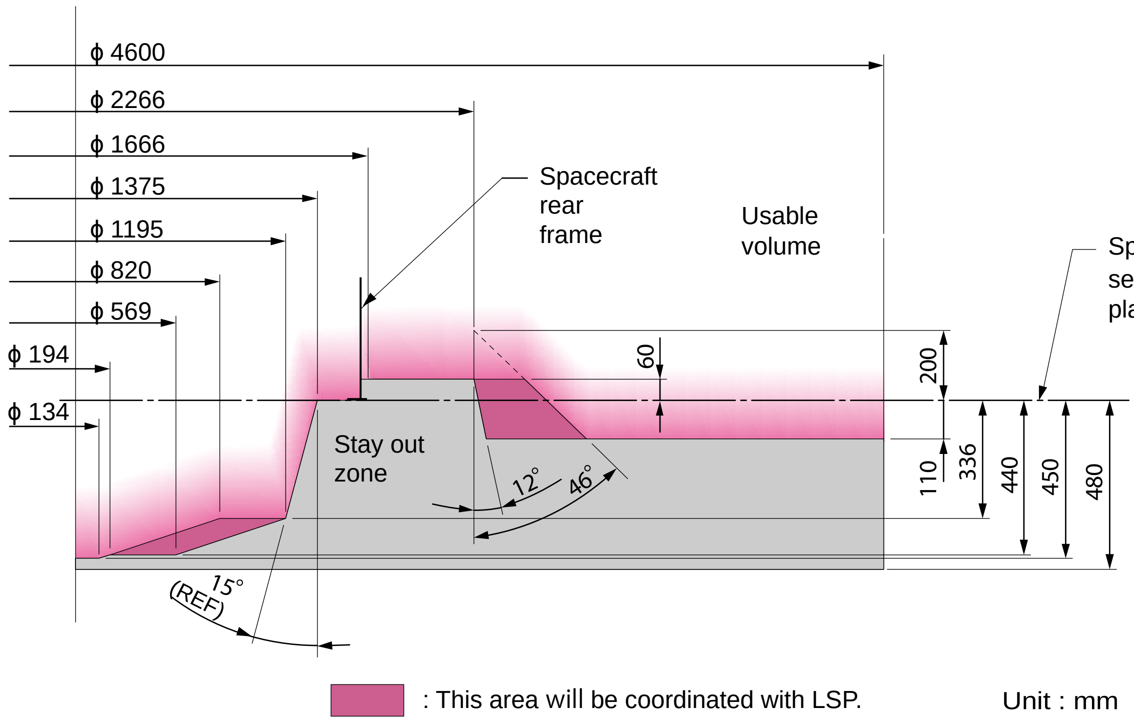


<Note>
Above illustration is applied to “Area A”(* See top).
For Model 1666MA : “Area A” are III-axis ±20°area and I-axis ±20°area(shown in purple) .
For Model 1666LS-H : “Area A” is III-axis ±20°area(shown in blue).

Figure A4.2-6 Stay-out zone around the 1666MA adapter (Area A)

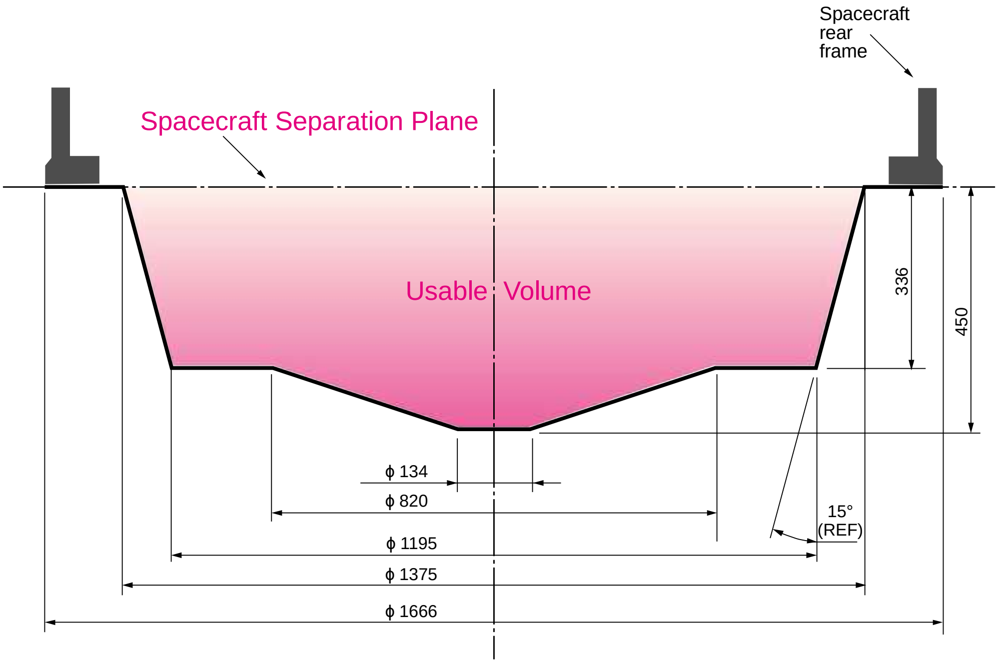


*Area B
PLA top view

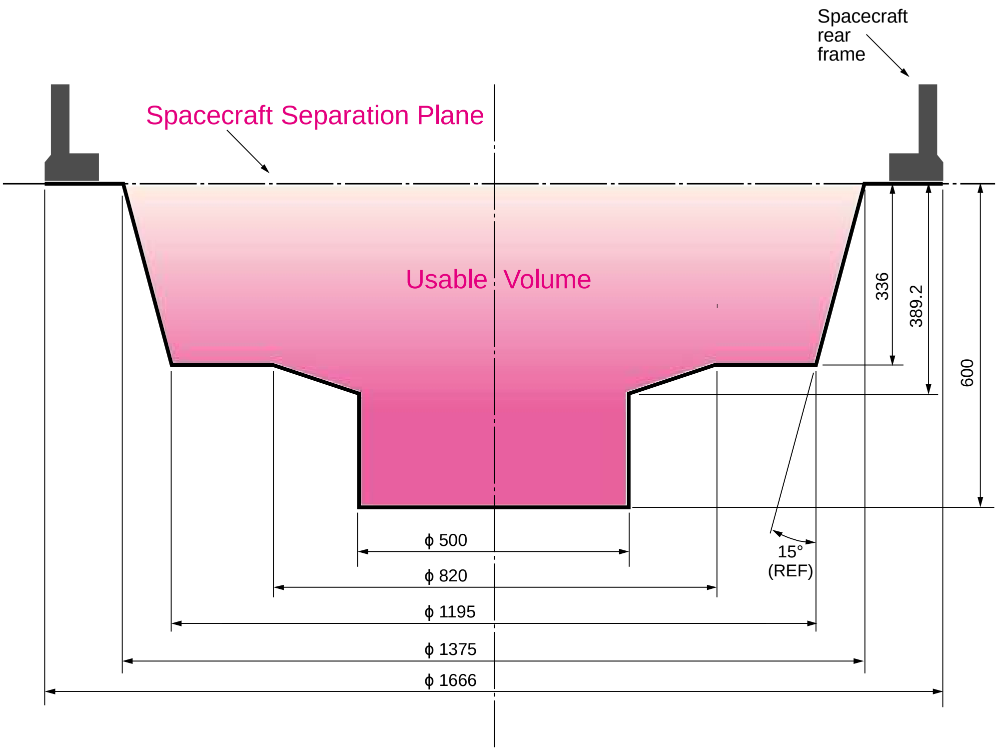


<Note>
Above illustration is applied to “Area B”(* see top).
For Model 1666M : “Area B” are II-axis ±70°area and IV-axis ±70°area(shown in purple) .
For Model 1666LS-H : “Area B” is I-axis ±160°area(shown in blue).

Figure A4.2-7 Stay-out zone around the 1666MA adapter (Area B)



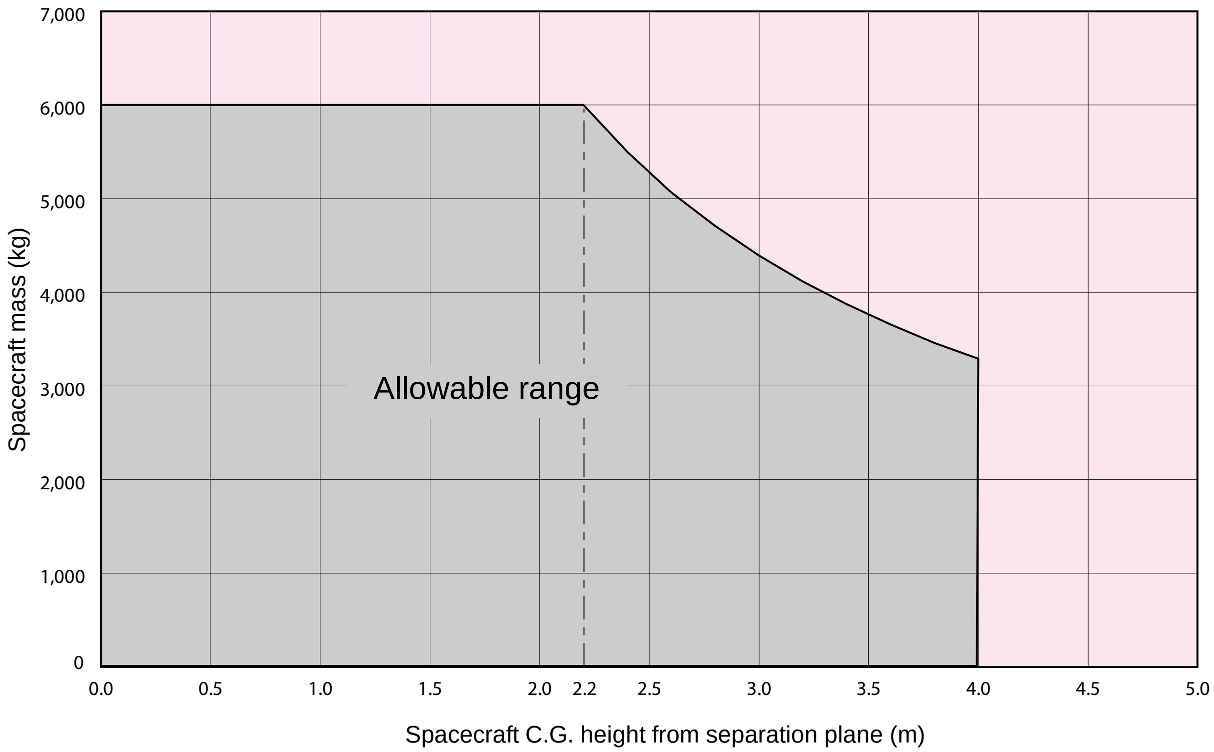
1666MA + standard PSS



Unit : mm

1666MA + optional PSS

Figure A4.2-8 Usable volume inside 1666MA adapter



This view graph shows non- C.G. offset case in the radial direction.
It is required to be less than 10% variation in the load distribution in the circumferential direction of the satellite.
Detailed data will be shown in TIM.

Figure A4.2-9 Limit load of the 1666MA adapter

* Separation shock spectrum of 1666MA is measured 125mm above separation plane.

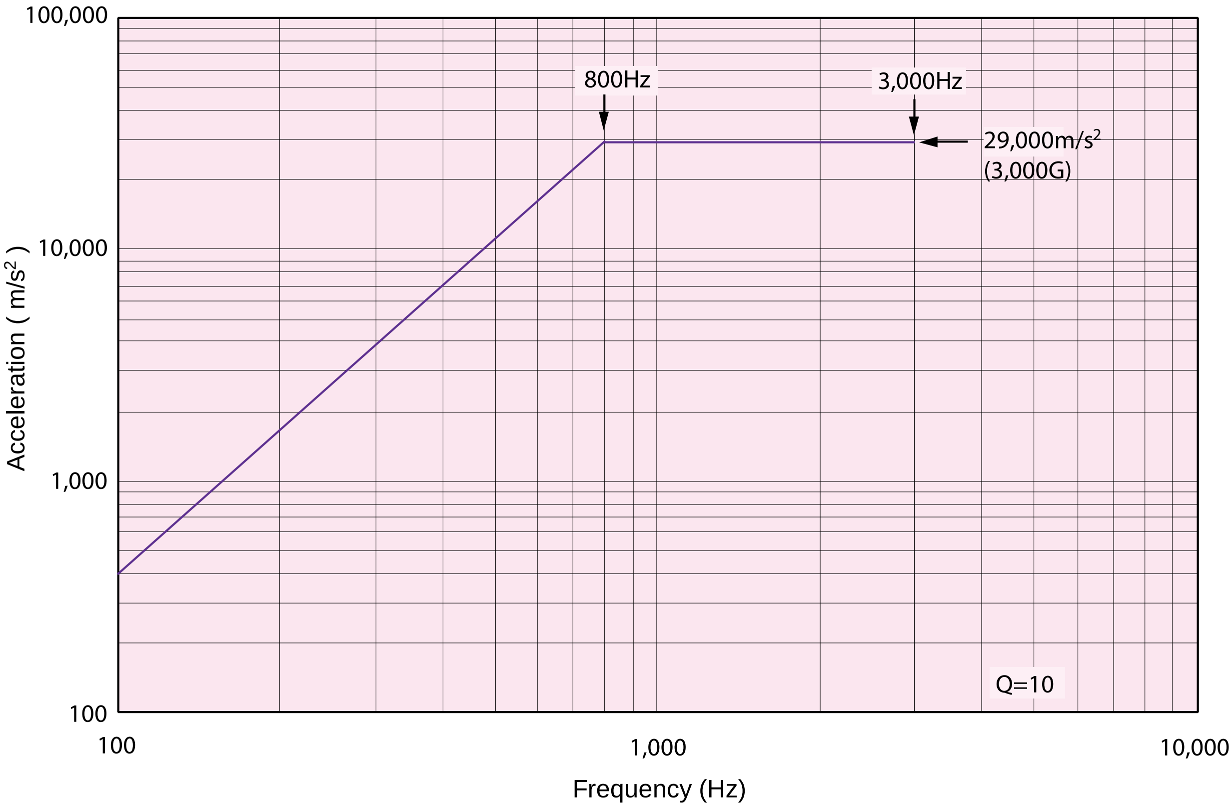
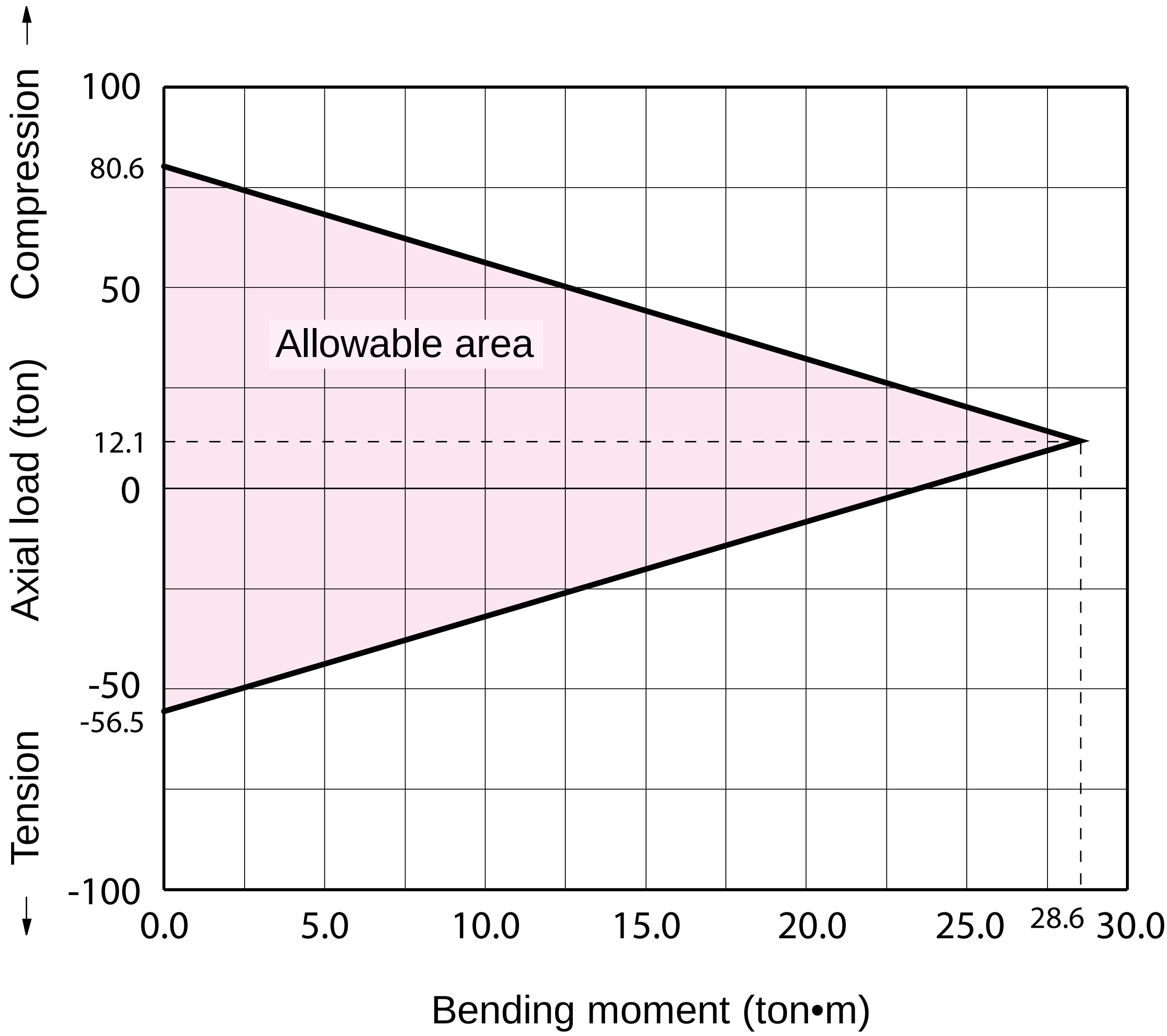


Figure A4.2-10　Spacecraft separation shock spectrum of the 1666MA adapter



This view graph shows non- C.G. offset case in the radial direction.
It is required to be less than 10% variation in the load distribution in the circumferential direction of the satellite.
Detailed data will be shown in TIM.

Figure A4.2-11 Limit load at separation plane of the 1666MA adapter

Appendix 4.3 PAYLOAD FAIRINGS

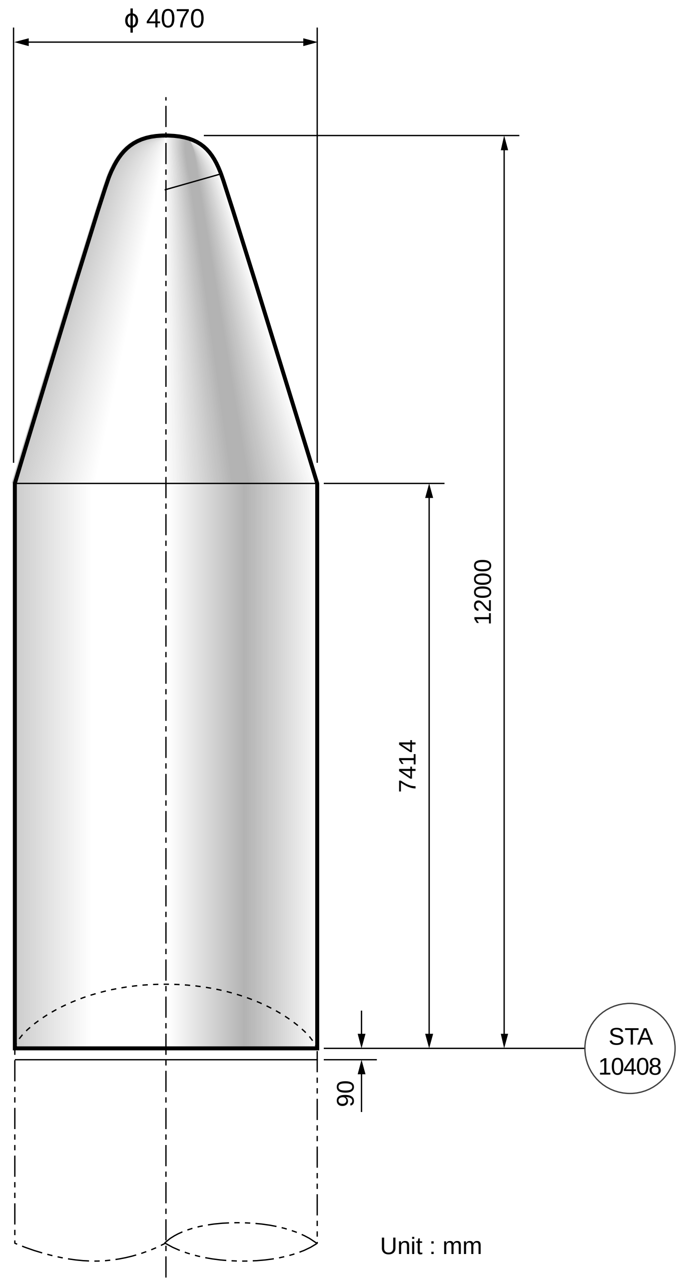
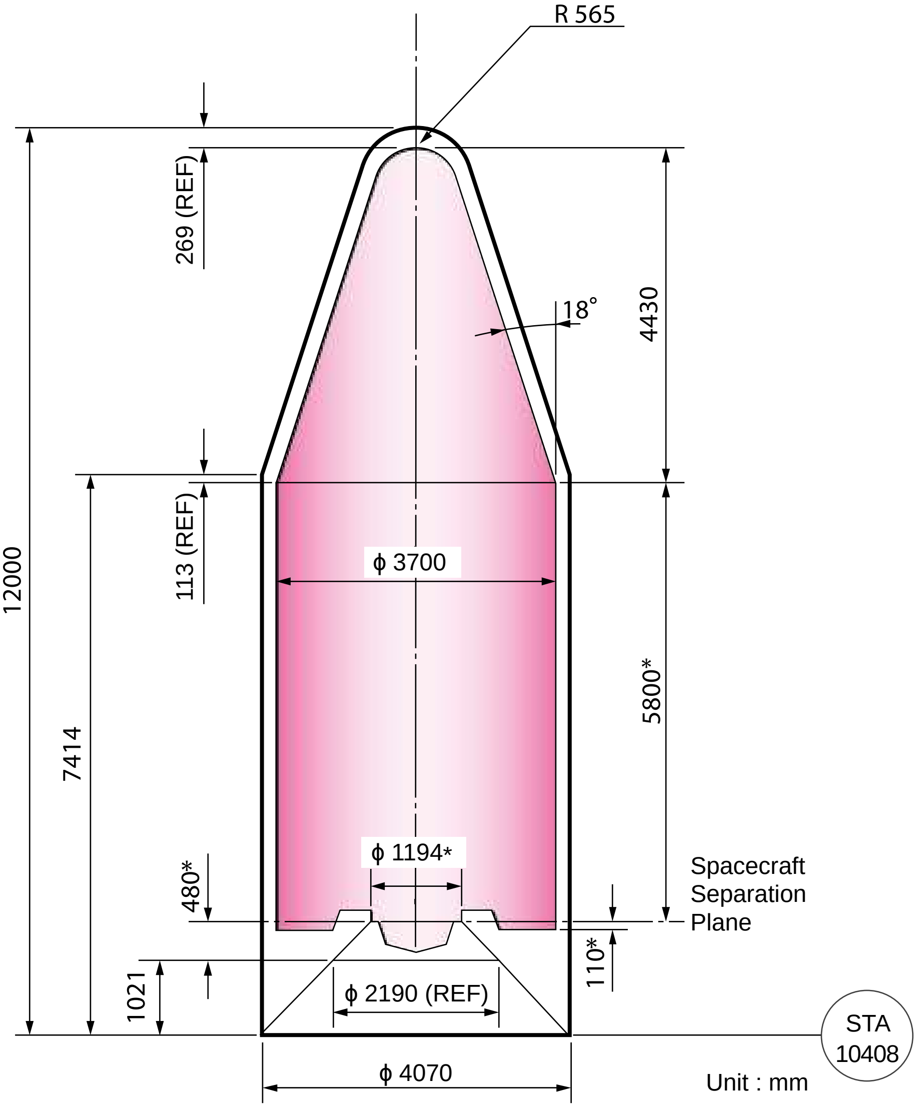


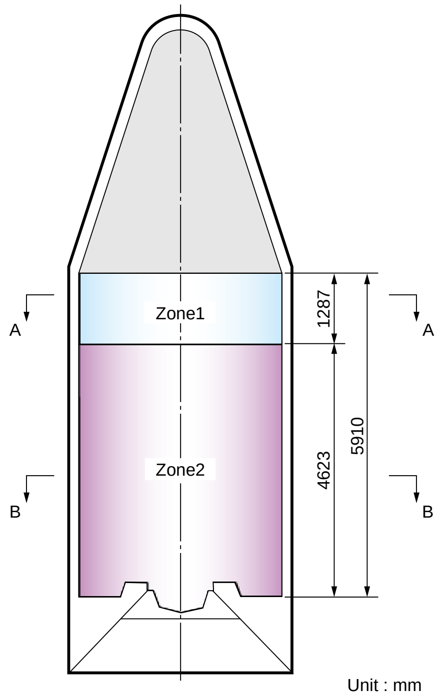
Figure A4.3-1 Model 4S



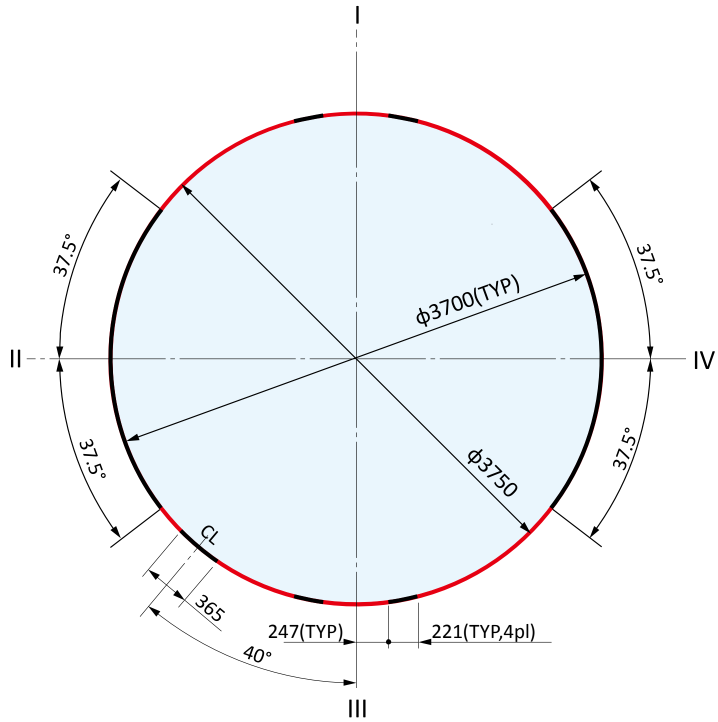
* These values will vary with adapter model.

Note) The radius of usable volume of the 4S fairing cylinder part is ϕ3700mm ~ ϕ3750mm.
Detailed is shown in fig A5.3-3.
If Spacecraft will protrude from the usable volume, please contact LSP for more detailed information.

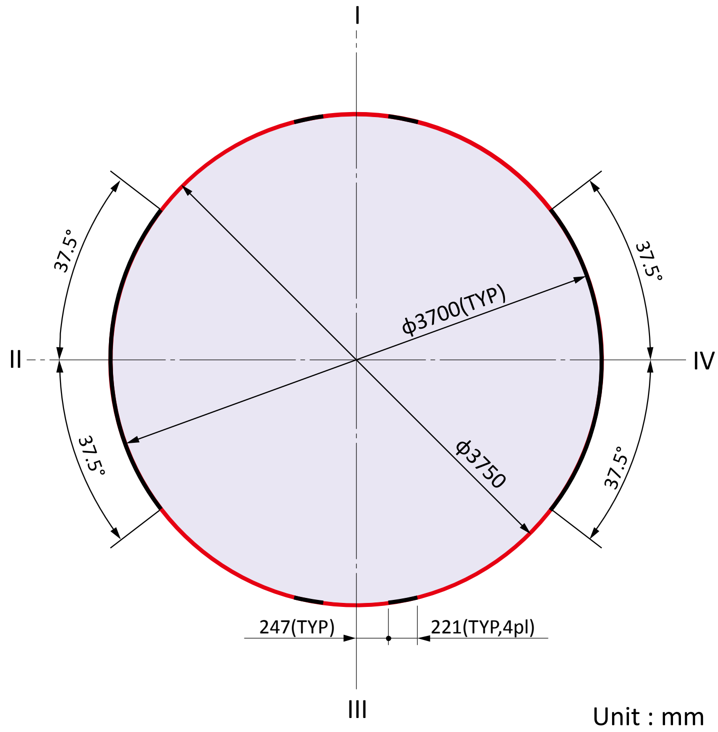
Figure A4.3-2 Usable volume of Model 4S (1/2)



Definition of
4S fairing cylinder part



Usable Volume of Section A-A(Zone1)



Usable Volume of Section B-B(Zone2)

Usable volume diameter = ϕ3700mm : black area,
Usable volume diameter = ϕ3750mm : red area

Figure A4.3-3 Usable volume of Model 4S (2/2)

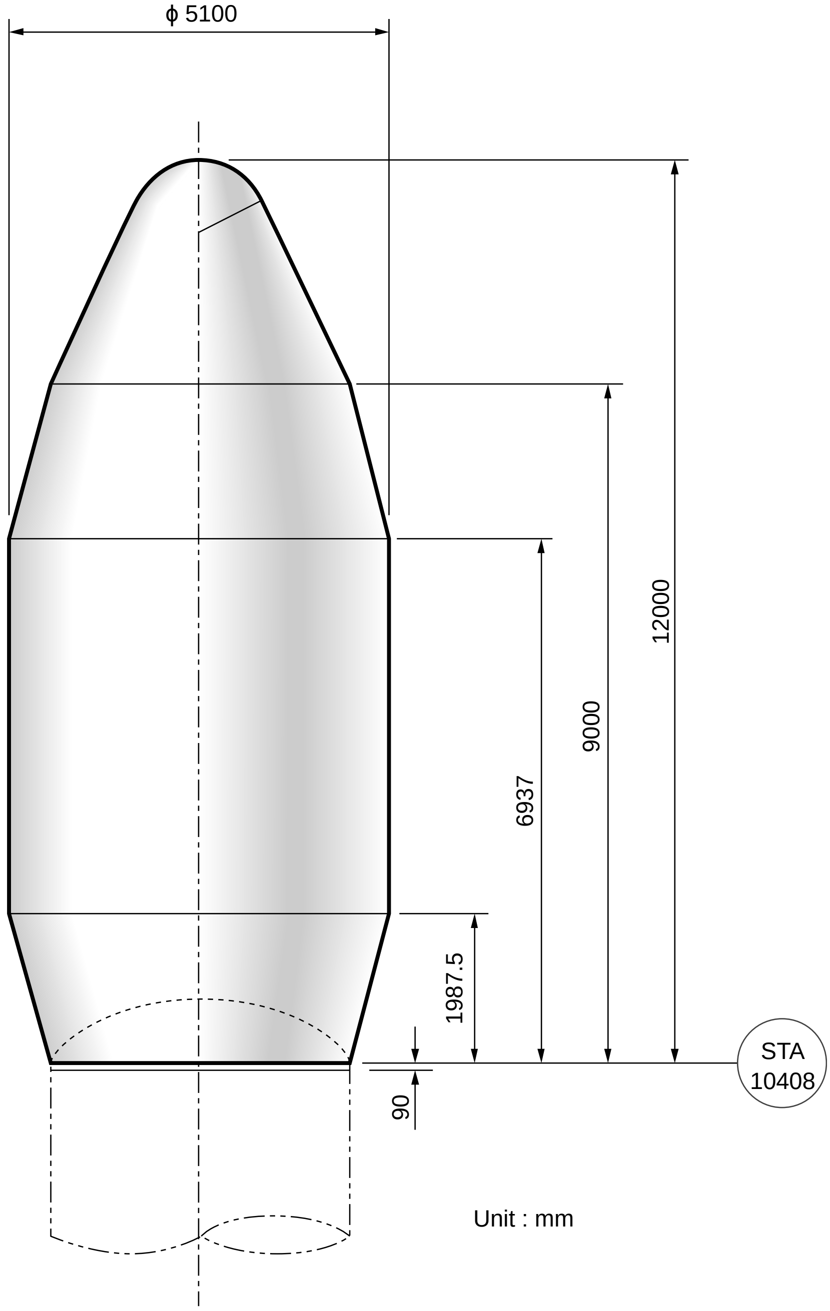
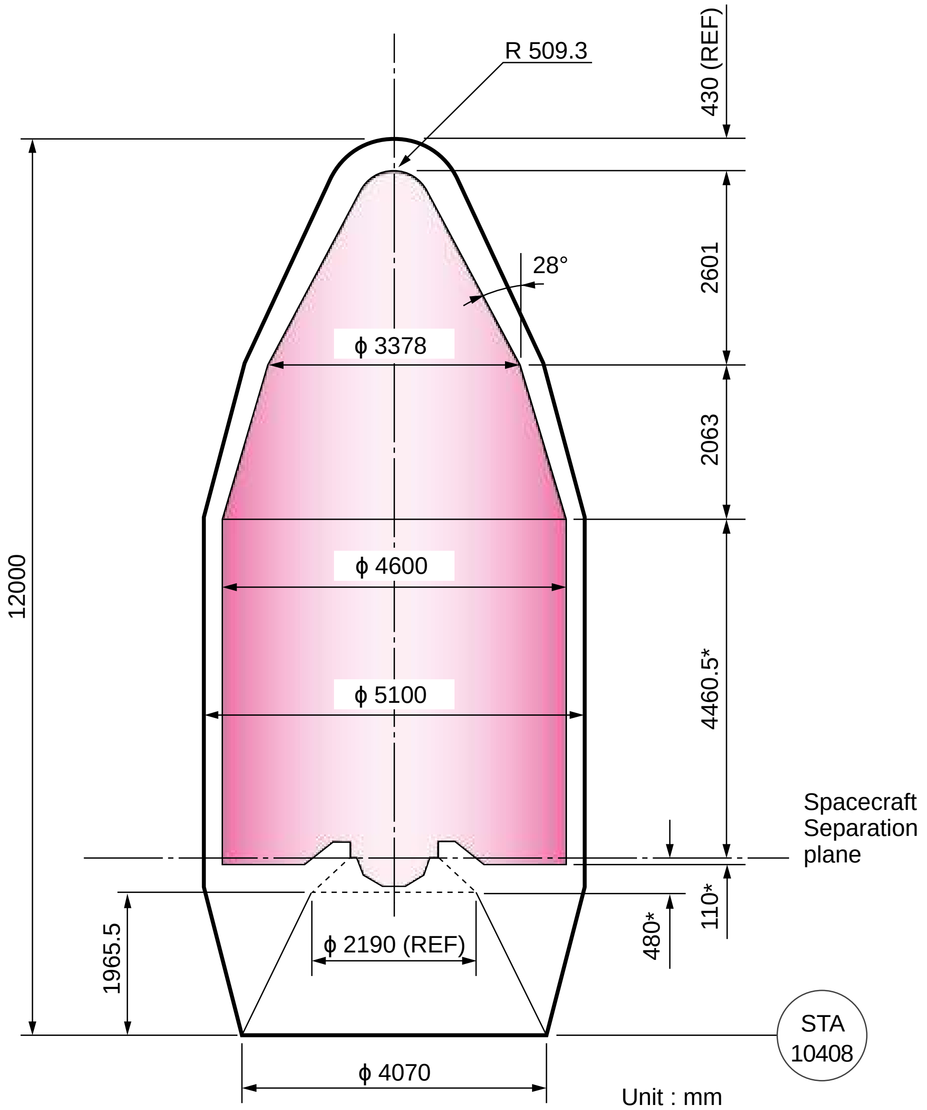


Figure A4.3-4 Model 5S



* These values will vary with adapter model.

Figure A4.3-5 Usable volume of Model 5S

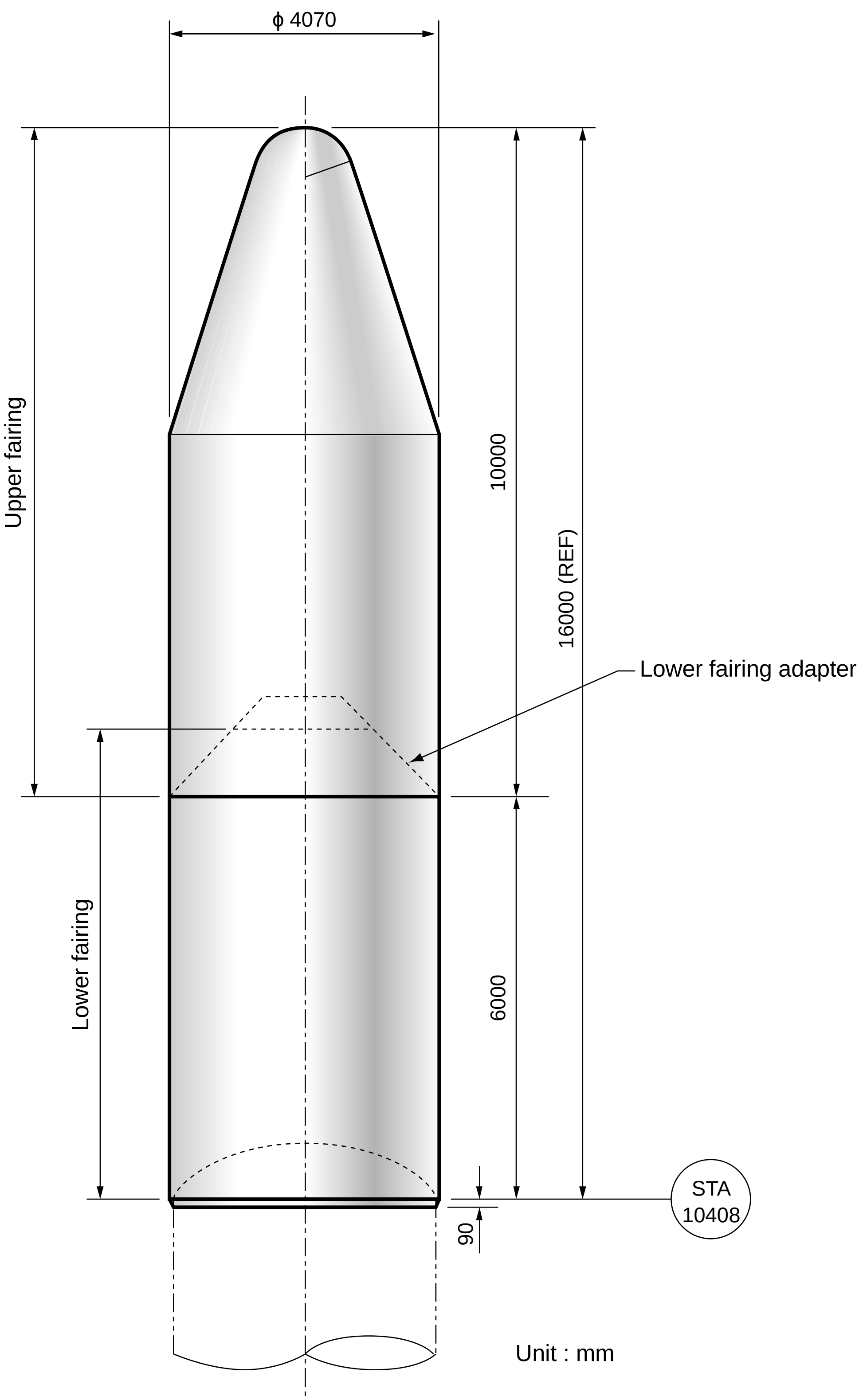
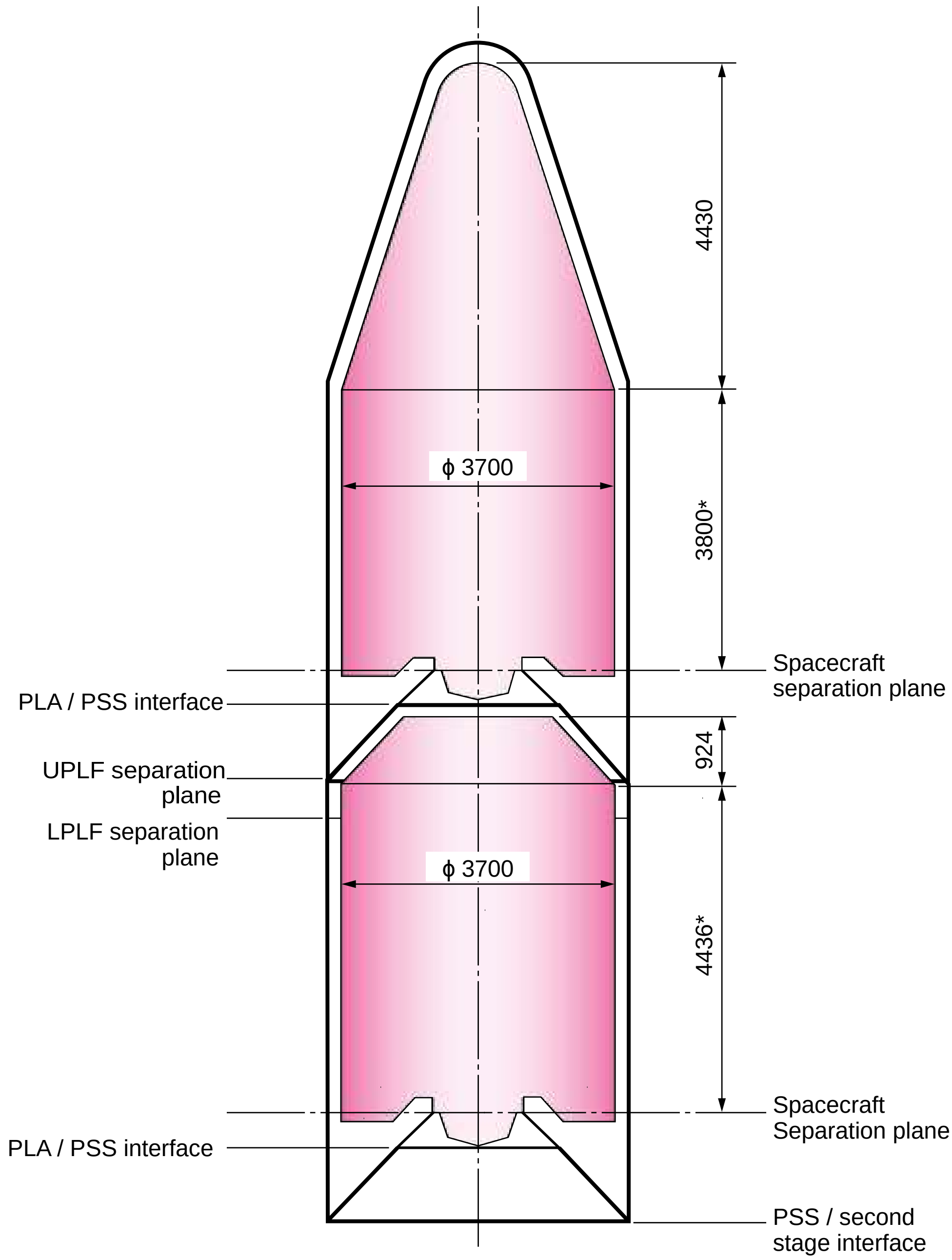


Figure A4.3-6　Model 4/4D-LC (1/2)



* These values will vary with adapter model.

Unit : mm

Figure A4.3-7 Model 4/4D-LC (2/2)

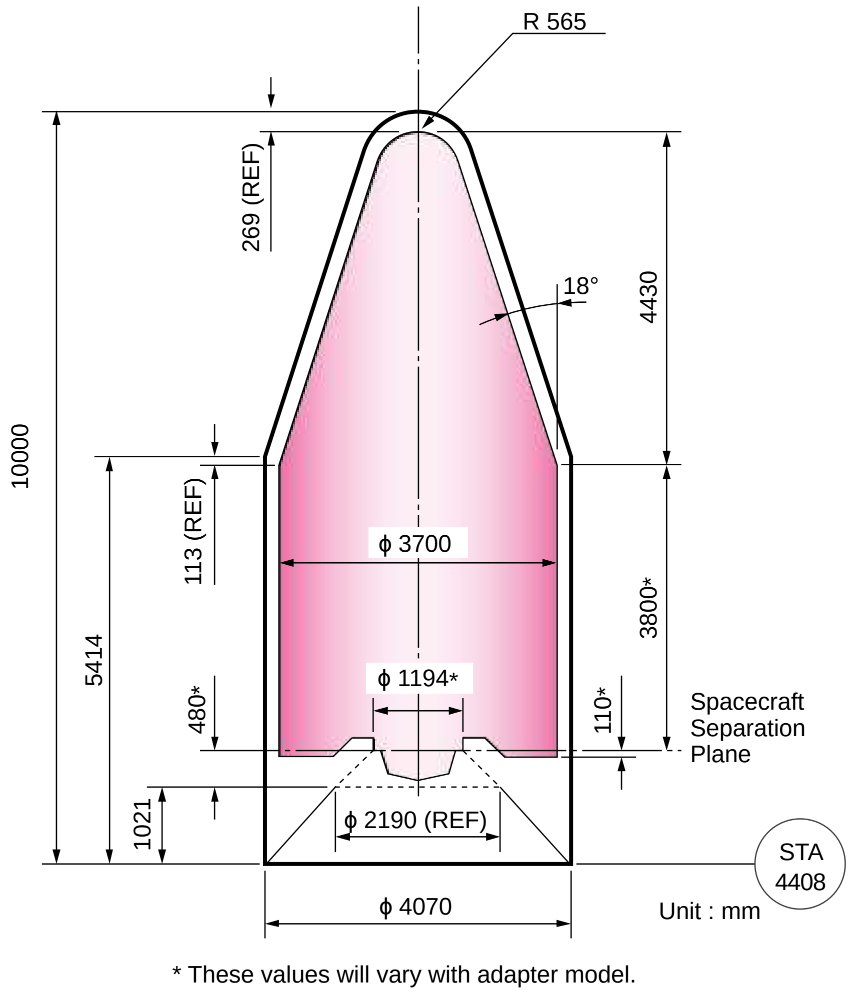
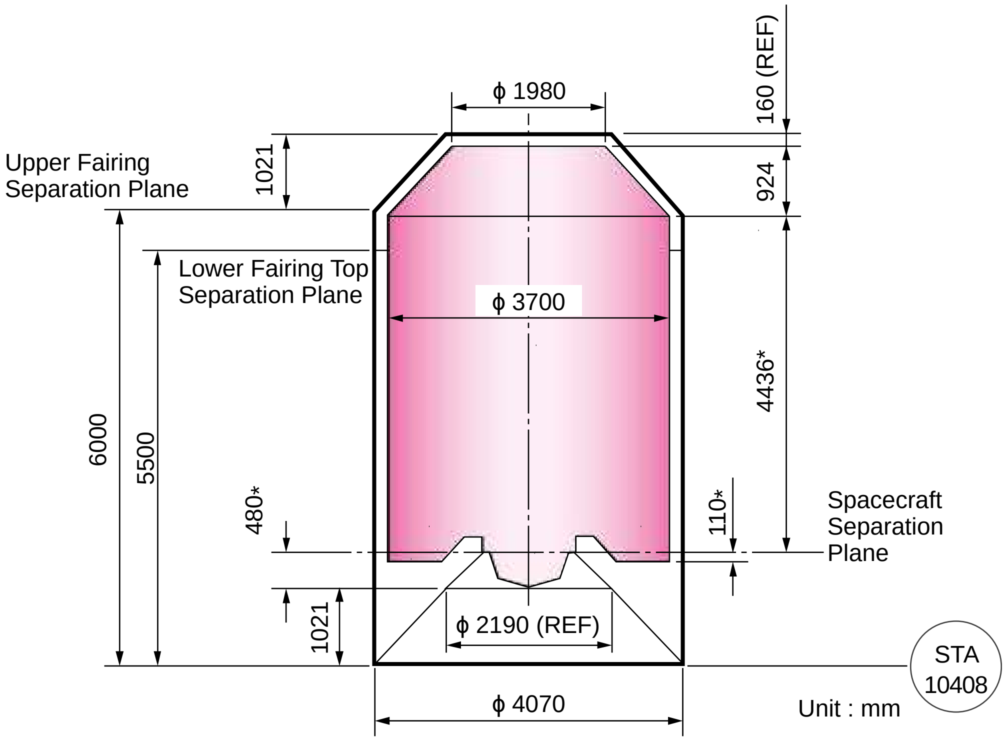
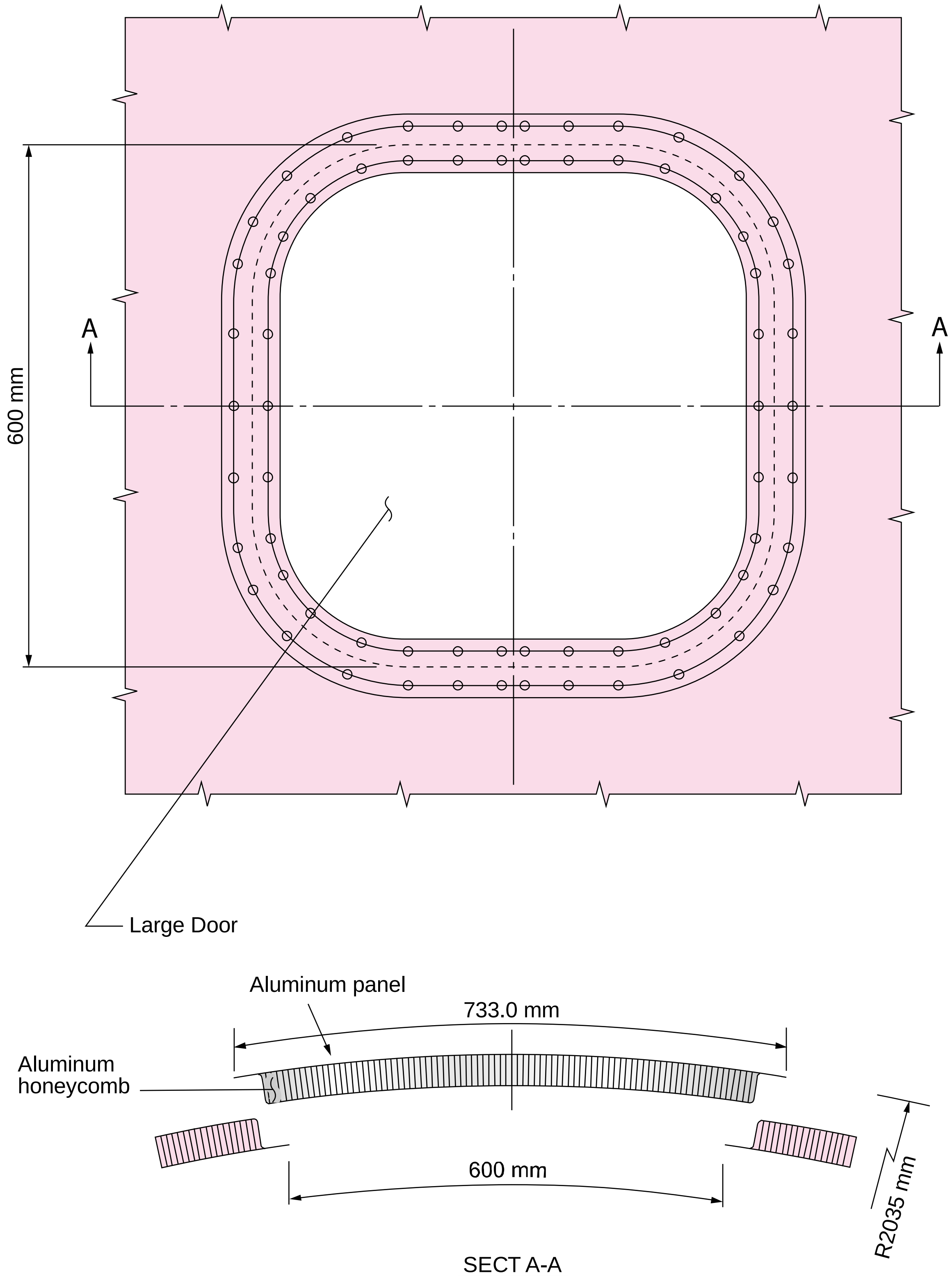


Figure A4.3-8 Usable volume of 4/4D-LC upper fairing



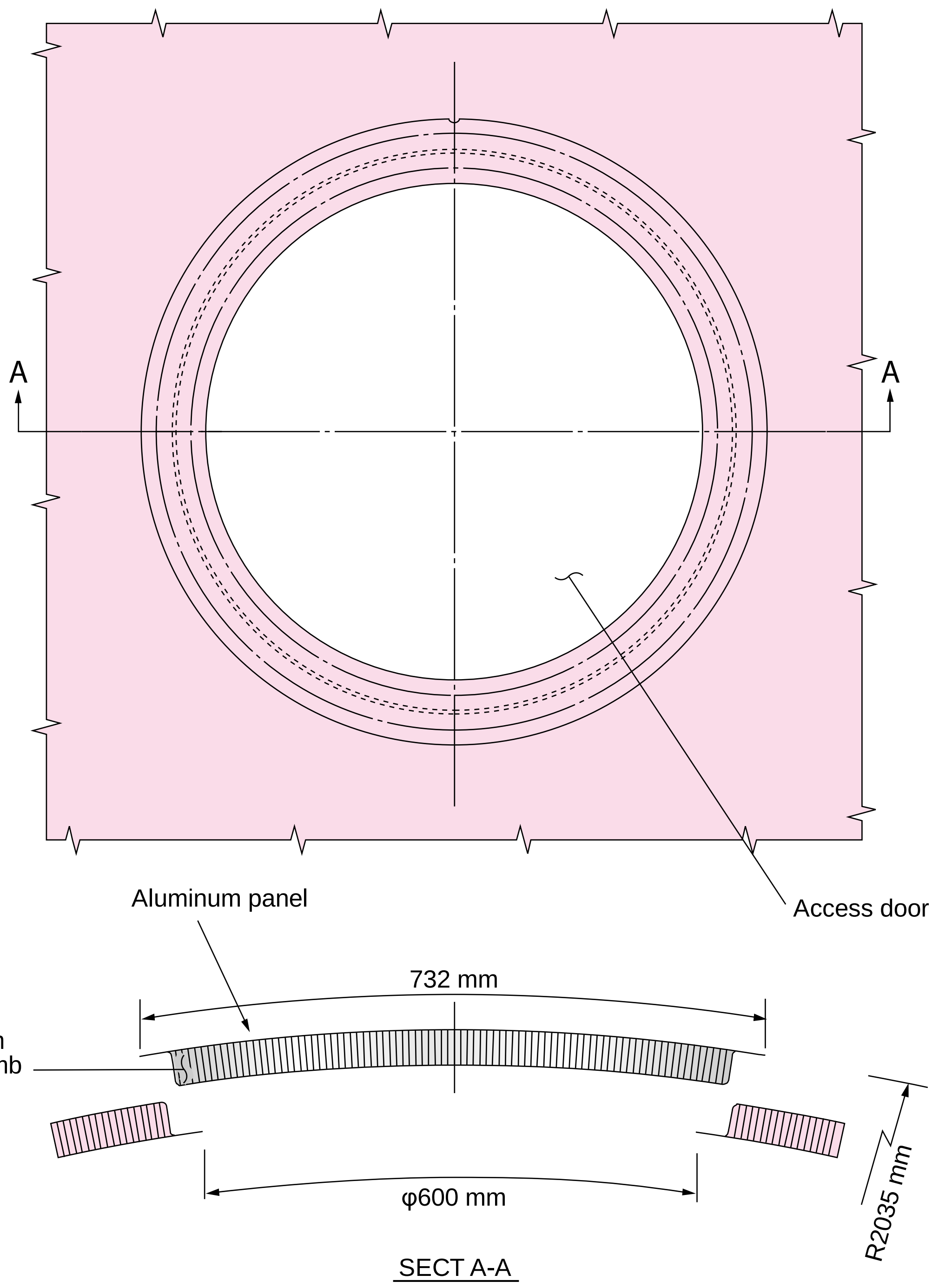
* These values will vary with adapter model.

Figure A4.3-9 Usable volume of 4/4D-LC lower fairing



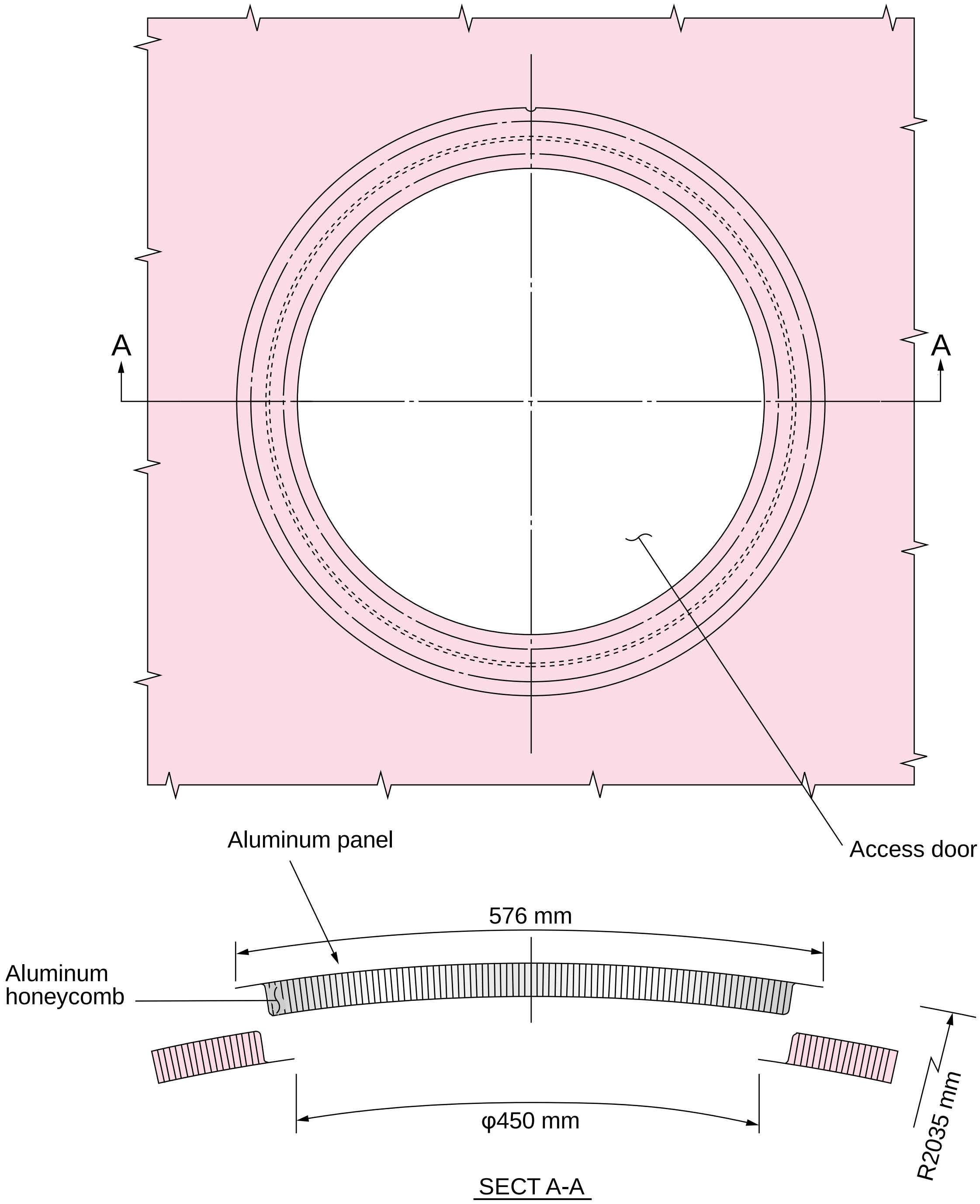
Note) This figure shows the configuration of the large door in case of the Model 4S fairing.

Figure A4.3-10 Large door



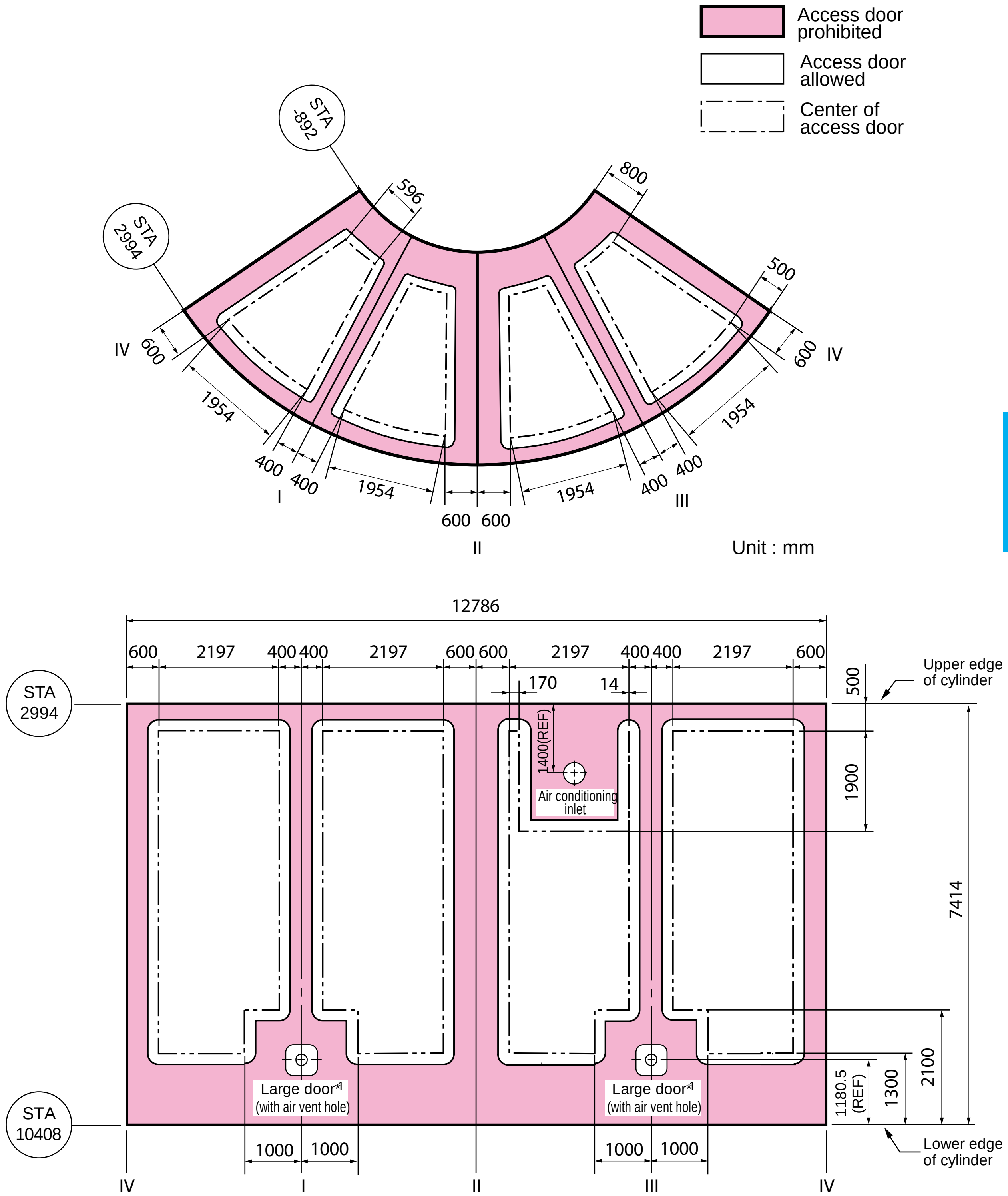
Note) This figure shows the configuration of the access door in case of the Model 4S fairing.

Figure A4.3-11 ø 600 access door



Note) This figure shows the configuration of the access door in case of the Model 4S fairing.

Figure A4.3-12 ø 450 access door (option)



Note)
*1: These doors are pre-fixed and for launch operations.

Figure A4.3-13 Allowable areas of ø 450 radio transparent window on Model 4S fairing

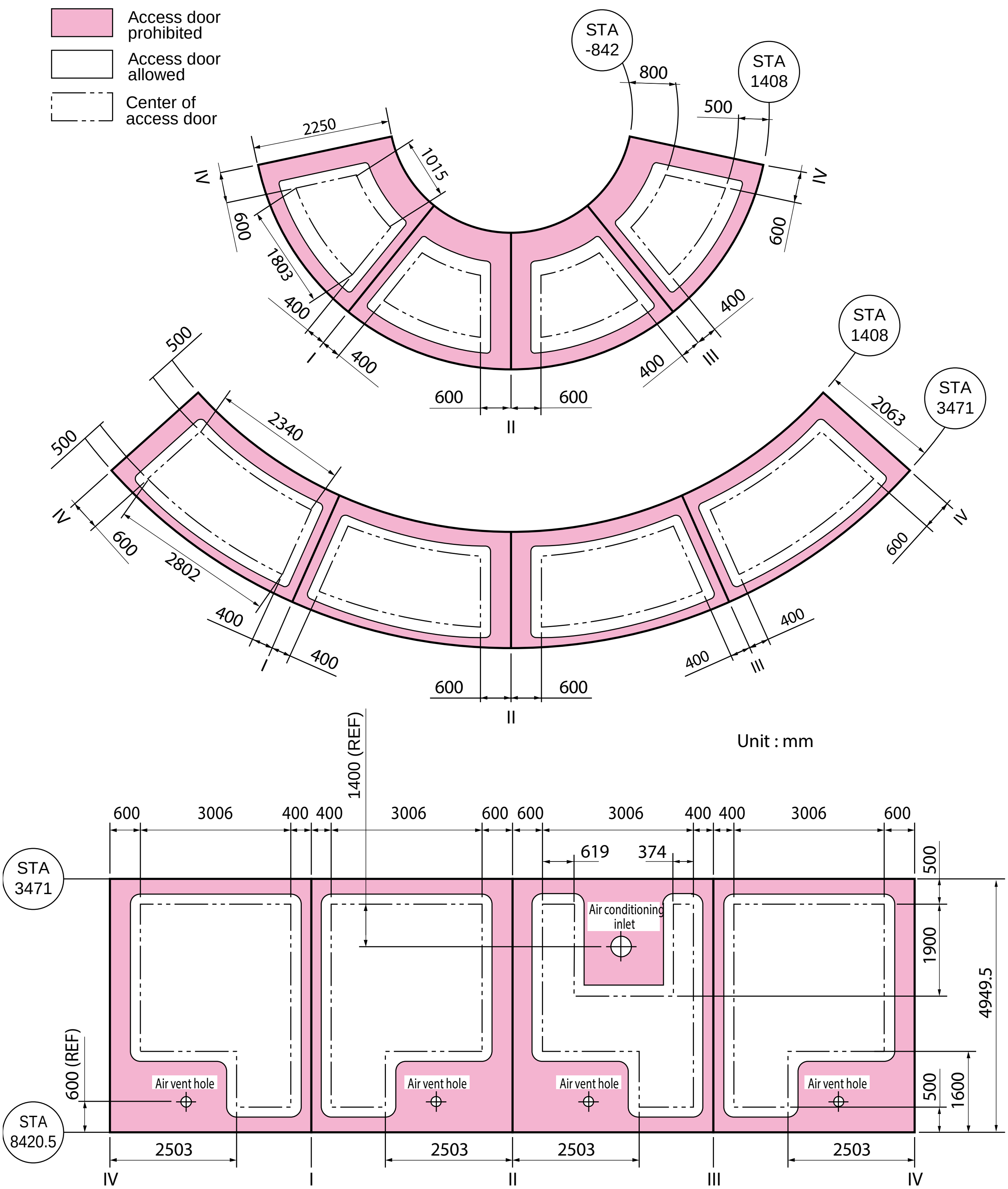
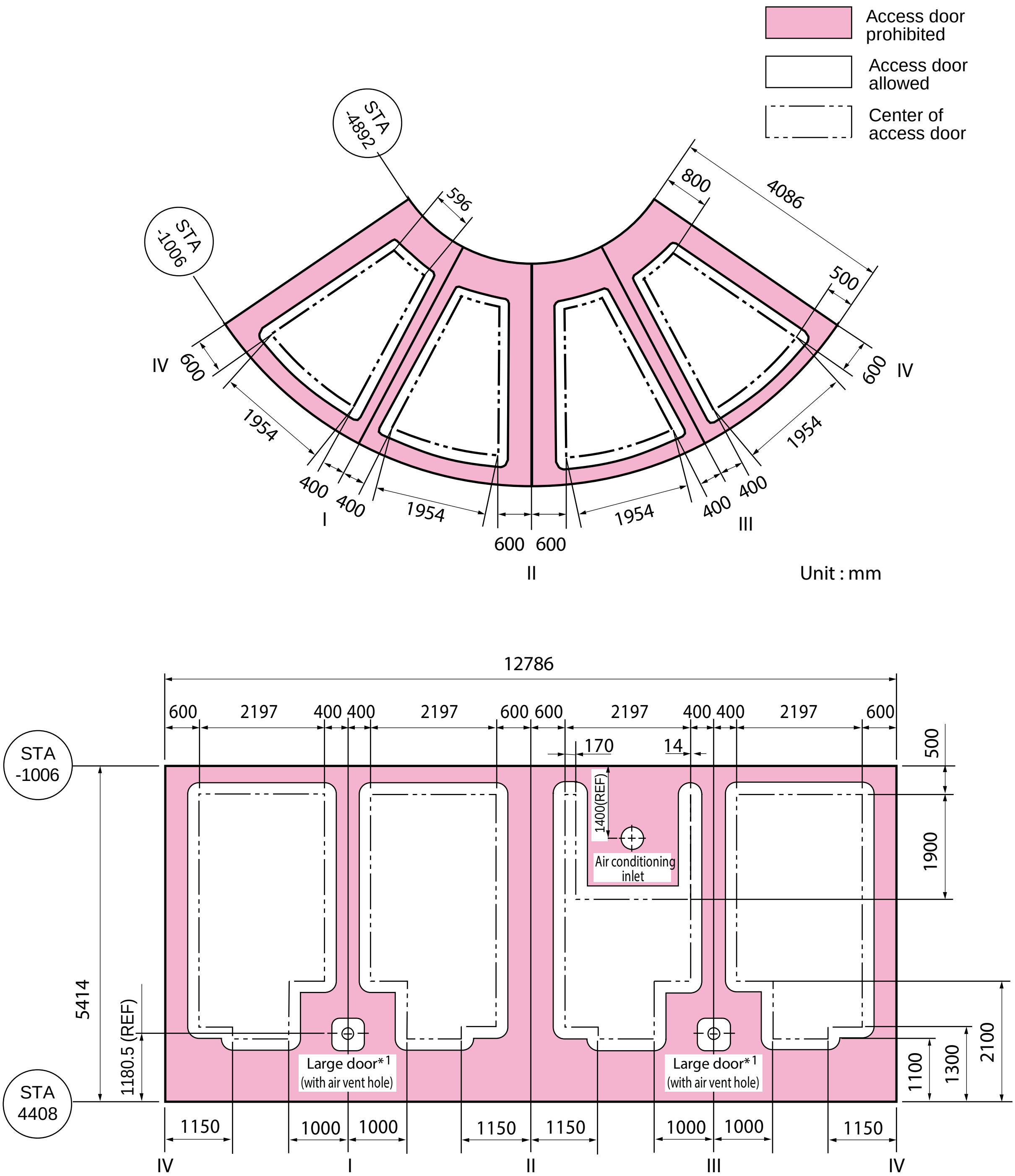
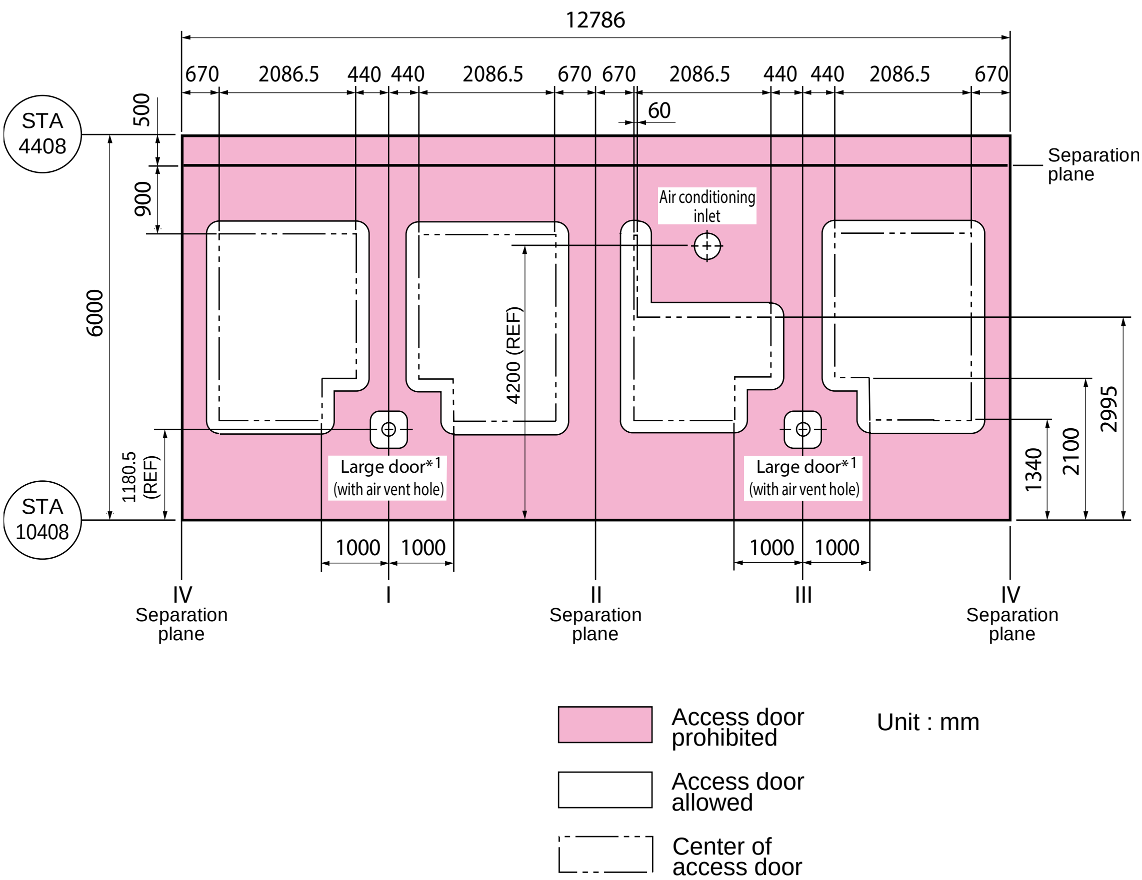


Figure A4.3-14 Allowable areas of ø 450 radio transparent window on Model 5S fairing



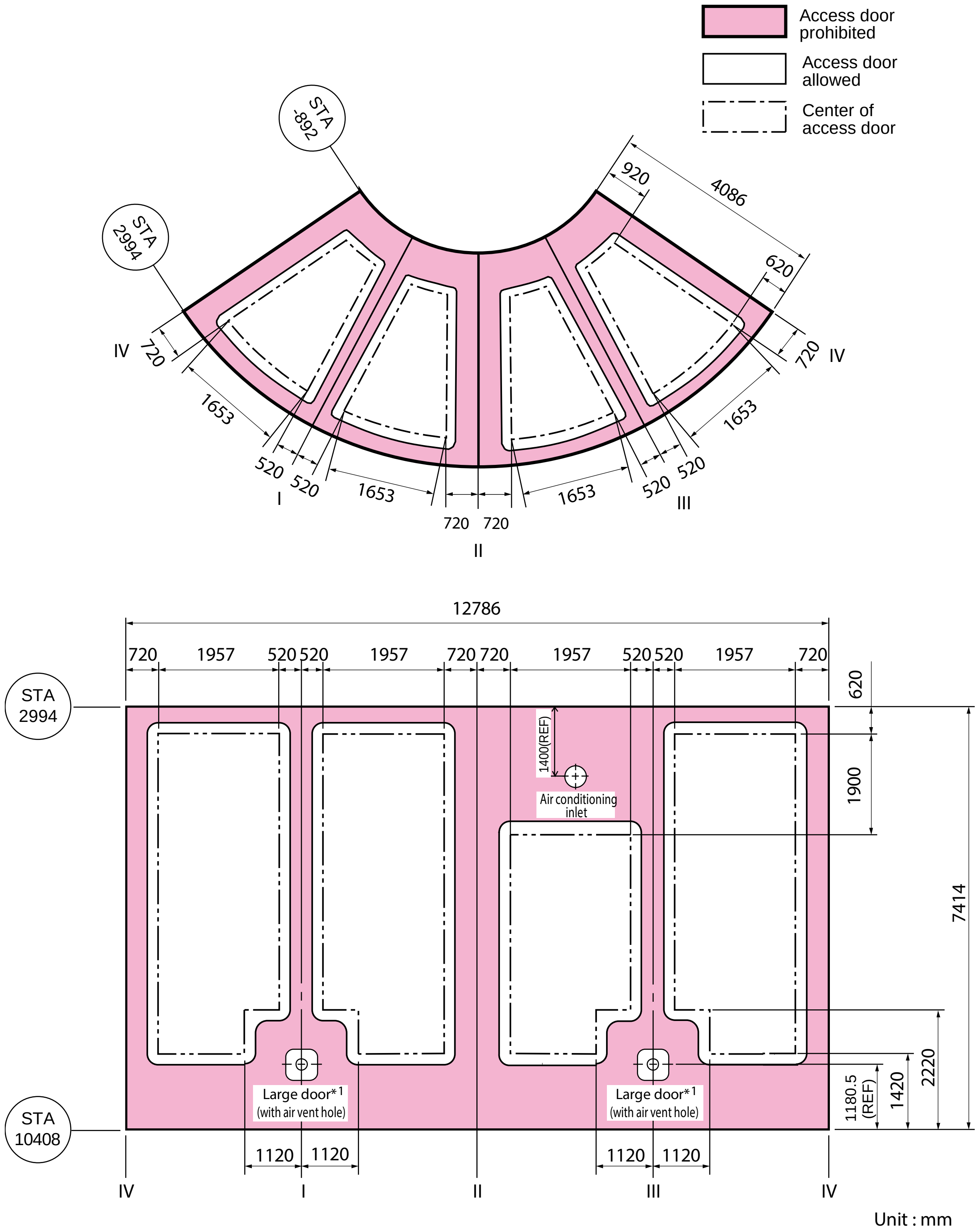
Note)
*1: These doors are pre-fixed and for launch operations.

Figure A4.3-15 Allowable areas of ø 450 radio transparent window on Model 4/4D-LC upper fairing



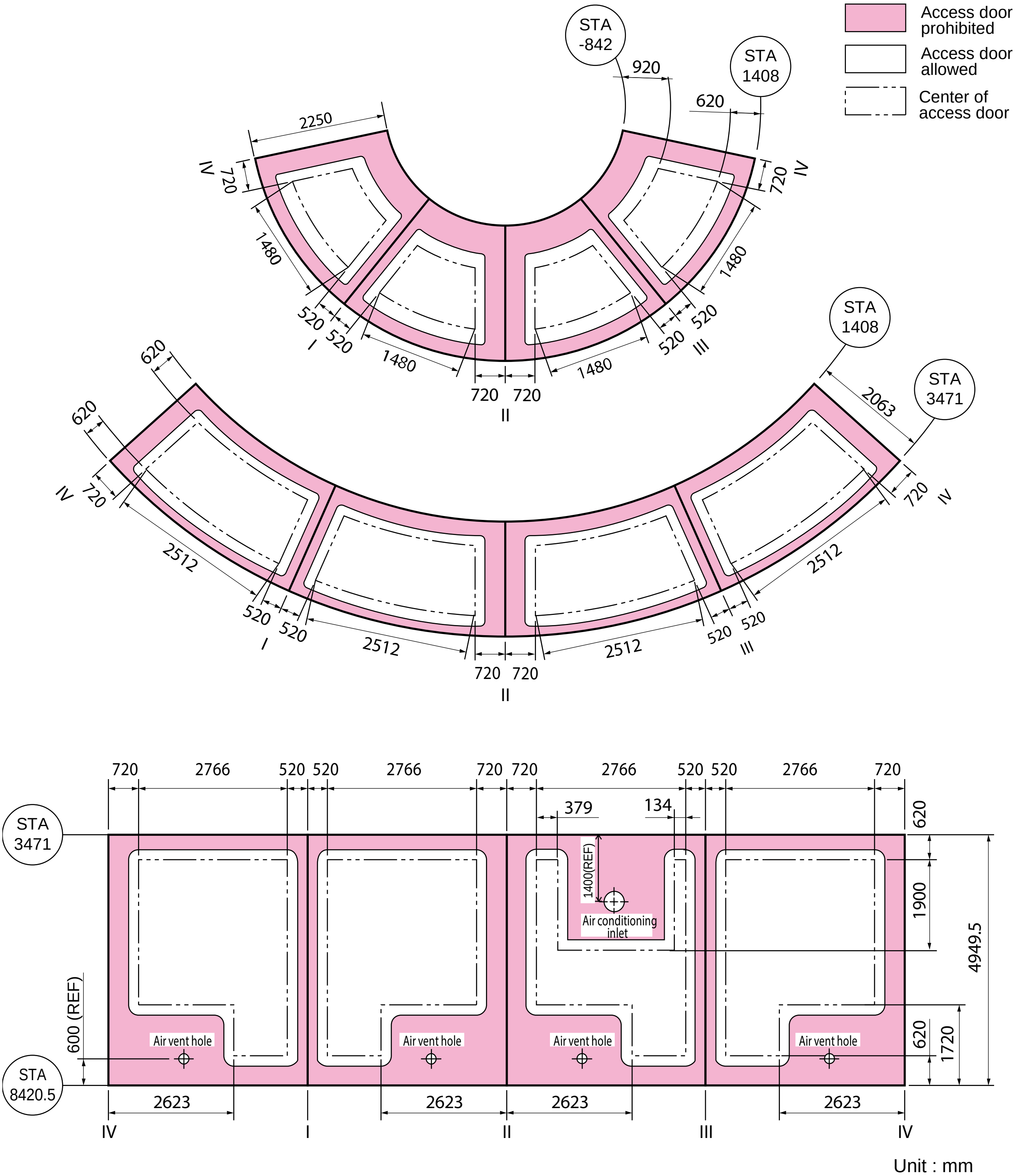
Note)
*1: These doors are pre-fixed and for launch operations.

Figure A4.3-16 Allowable areas of ø 450 radio transparent window on Model 4/4D-LC lower fairing



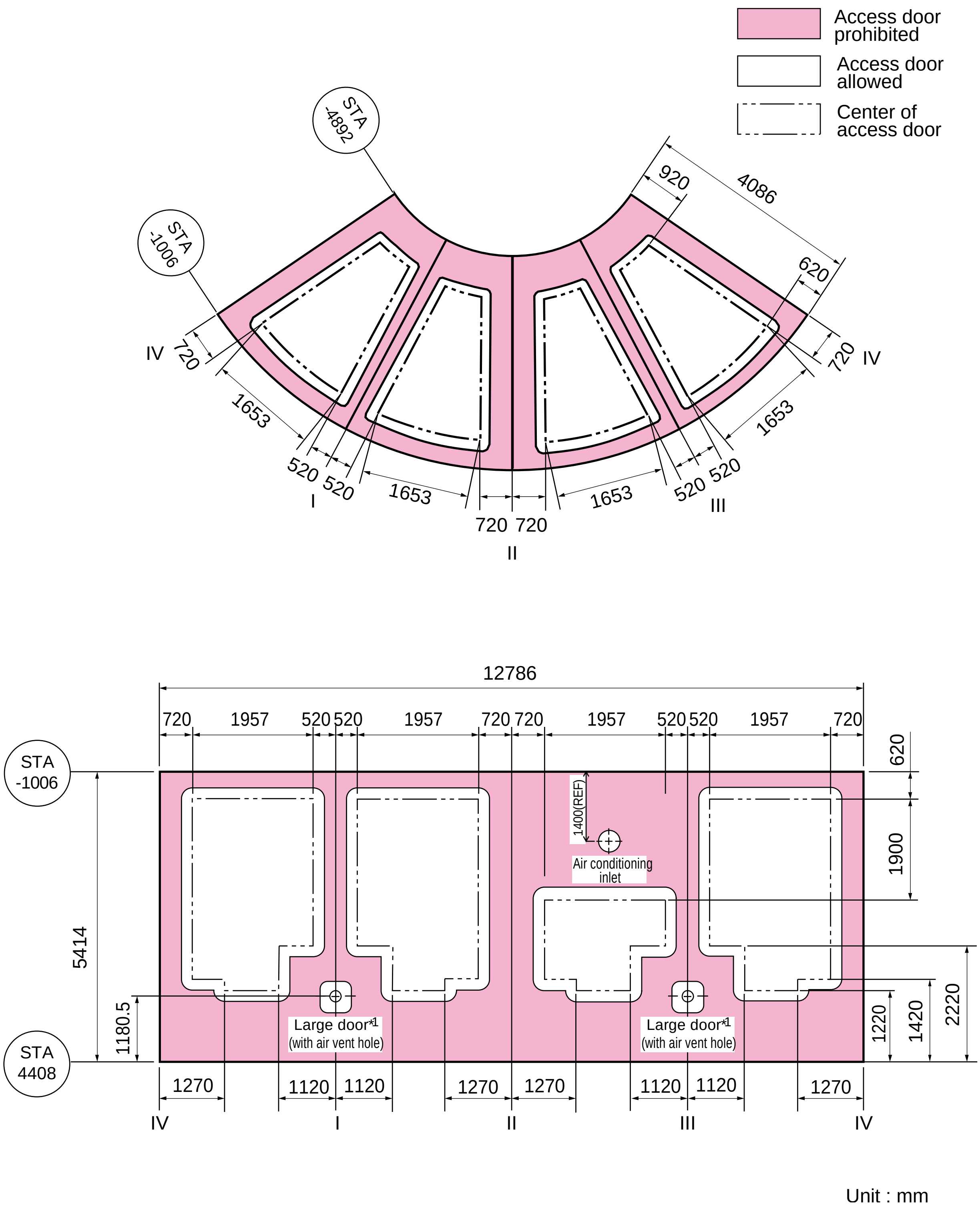
Note)
*1: These doors are pre-fixed and for launch operations.
*2: Access doors should be located in ±120º area from IV axis, when SCO uses diving board to access SC.

Figure A4.3-17 Allowable areas of ø 600 access door on Model 4S fairing



Note)
*1: These doors are pre-fixed and for launch operations.
*2: Access doors should be located in ±120º area from IV axis, when SCO uses diving board to access SC.

Figure A4.3-18 Allowable areas of ø 600 access door on Model 5S fairing

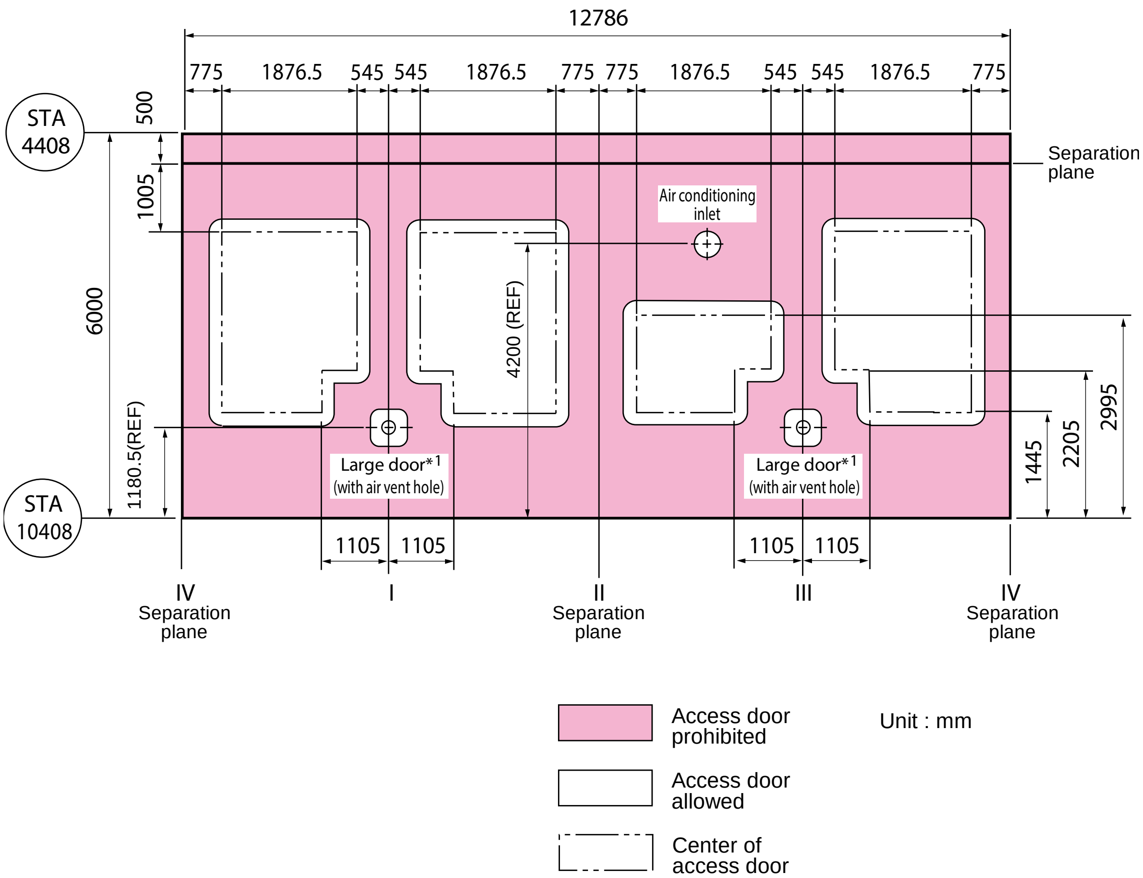


Note)

*1: These doors are pre-fixed and for launch operations.

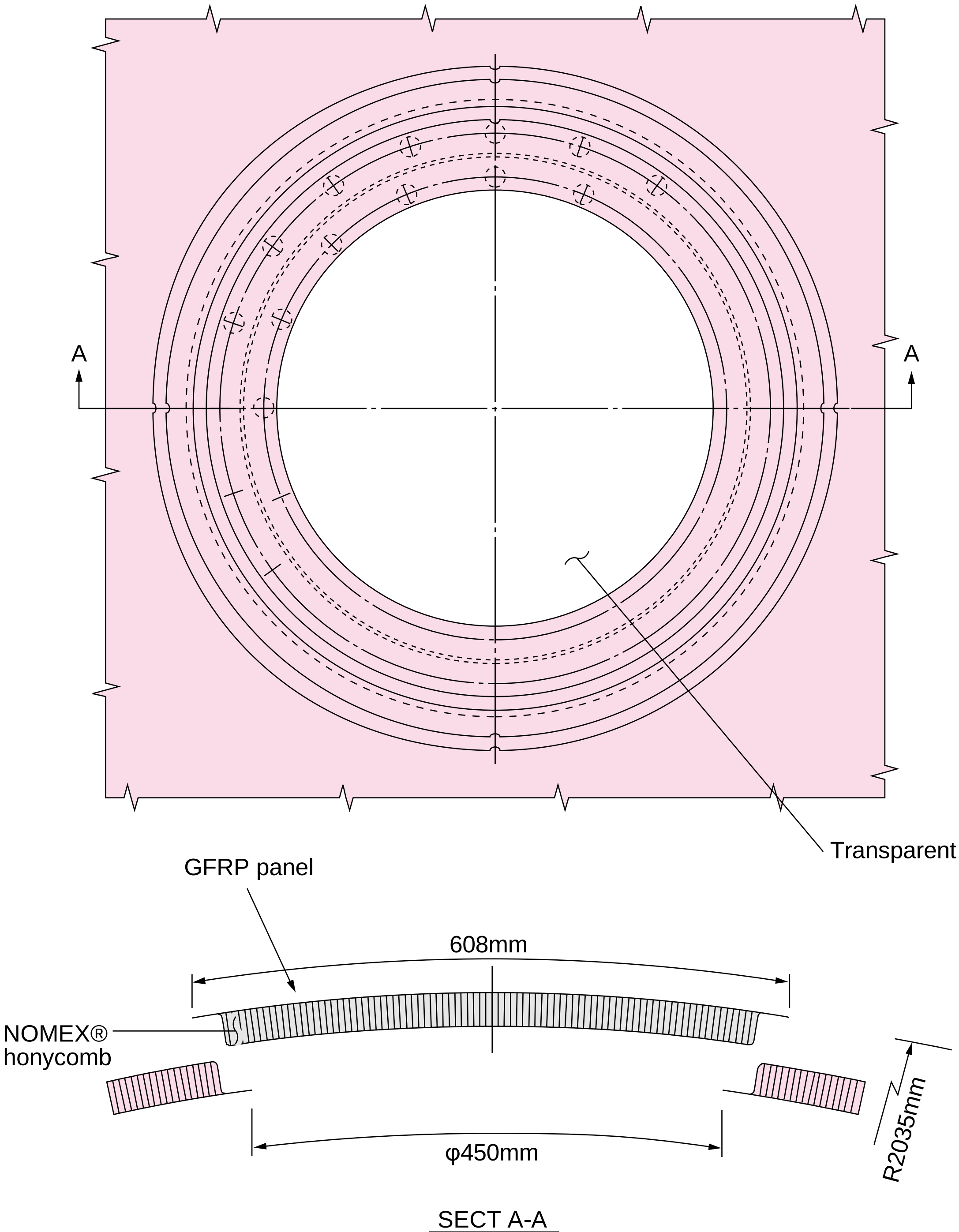
*2: Access doors should be located in ±120º area from IV axis, when SCO uses diving board to access SC.

Figure A4.3-19 Allowable areas of ø 600 access door on Model 4/4D-LC upper fairing



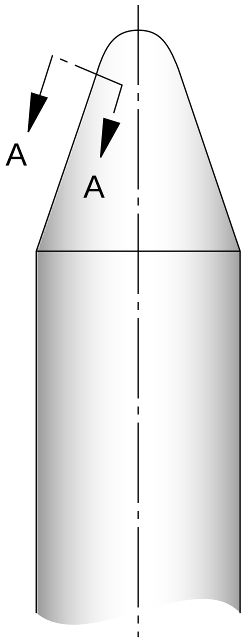
Note)
*1: These doors are pre-fixed and for launch operations.
*2: Access doors should be located in ±120º area from IV axis, when SCO uses diving board to access SC.

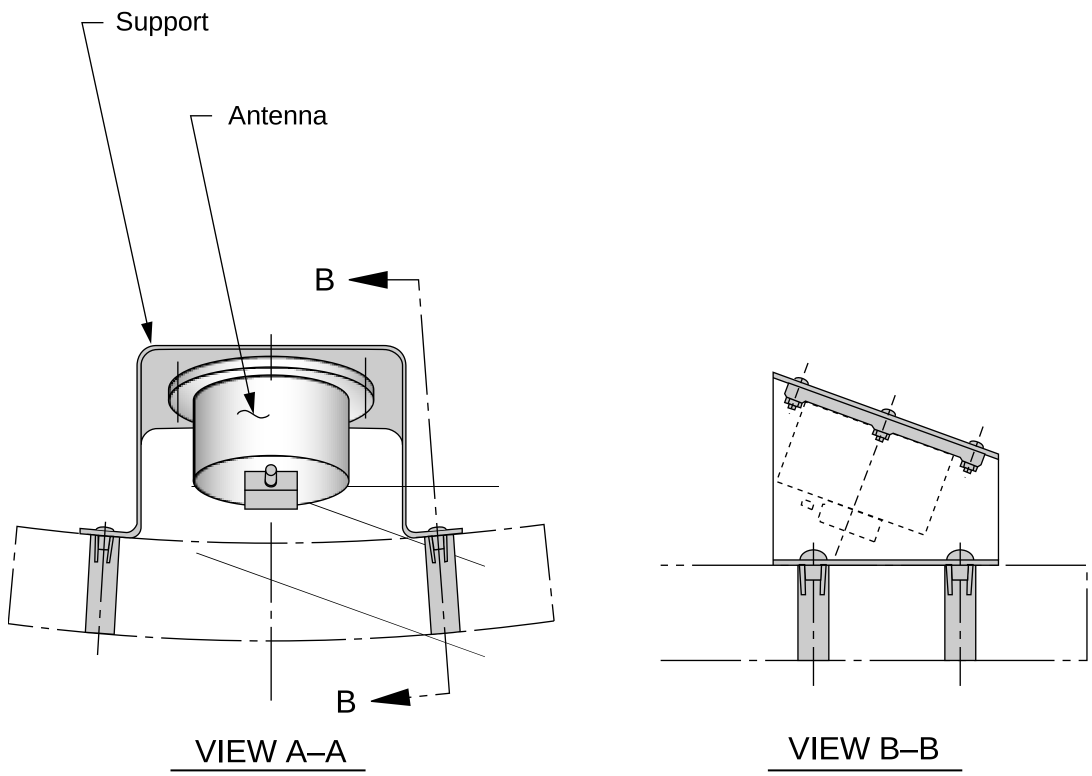
Figure A4.3-20　Allowable areas of ø 600 access door on Model 4/4D-LC lower fairing



Note)
This figure shows the configuration of the radio transparent window installed in the cylinder section of the Model 4S fairing.
The size of the radio transparent window is equivalent when it is installed in the cone section.

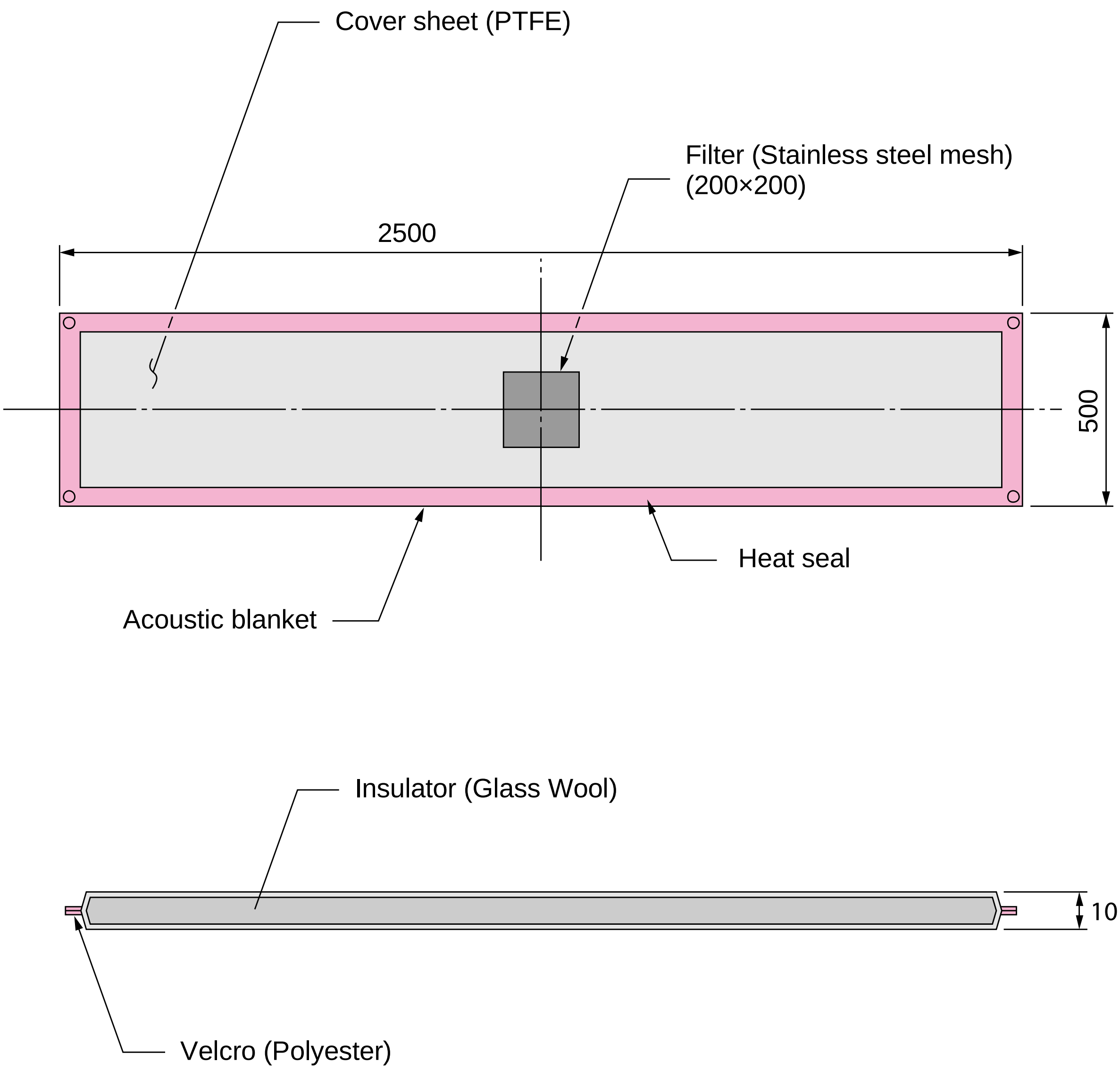
Figure A4.3-21 Radio transparent window





Unit : mm

Figure A4.3-22 Typical Installation of internal antenna



Unit : mm

Note)
*1: Blanket size will vary depending on equipment inside the fairing.

Figure A4.3-23 Typical Configuration of acoustic blanket

Appendix 4.3

Intentionally blank