NASDA-HDBK-1007D

| C

H-IIA User's Manual

Second Edition

December 2001

Published by
NATIONAL SPACE DEVELOPMENT AGENCY OF JAPAN

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:

National Space Development Agency of Japan

Mission Operations Department
Office of Space Transportation Systems

World Trade Center Building 26F
2-4-1 Hamamatsu-cho, Minato-ku Tokyo, 105-8060 Japan
Telephone + 81-3-3438-6465
Fax + 81-3-5402-6527

H-IIA User's Manual revision control sheet

A

<table>
<tr><th>Revision</th><th>Date</th><th colspan="3">Revised page</th><th>Approved</th></tr>
<tr><td>First</td><td>June. 1997</td><td colspan="3">————</td><td>————</td></tr>
<tr><td>Second</td><td>Dec. 1997</td><td colspan="3">All pages are redesigned.
Major revised items :
- 4.6.9 RF Link Interface
- Table A2.5.1</td><td></td></tr>
<tr><td rowspan="8">Second
Revision A</td><td rowspan="8">Dec. 1998</td><td rowspan="2">Contents

etc.</td><td>Technicals</td><td>Others</td><td rowspan="8"></td></tr>
<tr><td></td><td>0-1 ~ 0-15
0-17
0-18
0-19
0-21</td></tr>
<tr><td rowspan="2">Chapter 1</td><td>Technicals</td><td>Others</td></tr>
<tr><td>1-2
1-3
1-4
1-5
1-8
1-11
1-16 ~ 1-20
1-22
1-23</td><td>1-3
1-4
1-7
1-9
1-10
1-13
1-15
1-22</td></tr>
<tr><td rowspan="2">Chapter 2</td><td>Technicals</td><td>Others</td></tr>
<tr><td>2-1 ~ 2-12
2-15
2-17 ~ 2-23</td><td>2-1
2-3
2-4
2-8
2-9
2-13</td></tr>
<tr><td rowspan="2">Chapter 3</td><td>Technicals</td><td>Others</td></tr>
<tr><td>3-3
3-5
3-11
3-12
3-13
3-23
3-24</td><td>3-1
3-2
3-3
3-5 ~ 3-11
3-21
3-24
3-25</td></tr>
</table>

Revision	Date	Revised page			Approved
Second Revision A	Dec. 1998	Chapter 4	Technicals	Others	Y. Shibata
			4-6 ~ 4-18 4-26 ~ 4-33 4-38 ~ 4-43 4-48 4-52 4-56 4-59 4-60 4-61 4-64 ~ 4-68 4-71 4-72 4-73	4-1 4-2 4-41 4-53 4-54 4-55 4-57 4-58 4-63 4-69 4-74	
		Chapter 5	Technicals	Others	
			5-1 5-2 5-4 5-7 5-10 5-12 5-14 5-15 5-20 5-21 5-23 5-25 5-26 5-28 5-29 5-31 5-32	5-1 5-3 5-5 5-6 5-7 5-11 5-18 5-19	

A

<table>
<tr><th>Revision</th><th>Date</th><th colspan="3">Revised page</th><th>Approved</th></tr>
<tr><td rowspan="8">Second
Revision A</td><td rowspan="8">Dec. 1998</td><td rowspan="2">Chapter 6</td><td>Technicals</td><td>Others</td><td rowspan="8">Y. Shibata</td></tr>
<tr><td>6-1
6-2
6-10 ~ 6-14
6-19
6-20</td><td>6-3
6-4
6-5
6-9
6-14
6-19
6-22</td></tr>
<tr><td rowspan="2">Appendix 1</td><td>Technicals</td><td>Others</td></tr>
<tr><td>A1-2</td><td></td></tr>
<tr><td rowspan="2">Appendix 2</td><td>Technicals</td><td>Others</td></tr>
<tr><td>A2-9
A2-10
A2-15
A2-16
A2-17
A2-20 ~ A2-21
A2-24
A2-27</td><td>A2-4
A2-5
A2-6
A2-12
A2-13
A2-17
A2-18
A2-21
A2-22
A2-23
A2-25
A2-26
A2-28 ~ A2-32</td></tr>
<tr><td rowspan="2">Appendix 3</td><td>Technicals</td><td>Others</td></tr>
<tr><td>All
(except the pages mentioned in the right table)</td><td>Appendix 3.3
A3.5-2
A3.5-3
A3.5-6 ~ 9
Appendix 3.8
Appendix 3.9</td></tr>
</table>

A

Revision	Date	Revised page			Approved
Second Revision B	Dec. 1999	Contents etc.	Technicals	Others	Tomio Nakano
				Cover 0-2-3 0-3 0-10 0-11 0-12 0-14 0-16 0-17 0-18	
		Chapter 1	Technicals	Others	
			1-4 1-8 1-19	1-13 1-15	
		Chapter 2	Technicals	Others	
			2-2 2-5 2-7 2-10 2-11 2-12 2-13 2-14 2-15 2-16 2-18 2-19 2-22 2-23	2-8 2-15 2-16 2-24 2-25	
		Chapter 3	Technicals	Others	
			3-5 3-12 3-14	3-1 3-2 3-9 3-10 3-13 3-15	

B

Revision	Date	Revised page			Approved
Second Revision B	Dec. 1999	Chapter 4	Technicals	Others	Tomio Nakano
			4-3 4-24 4-38 4-39 4-40 4-41 4-42 4-53 4-54 4-55 4-57 4-63 4-64 4-65 4-67 4-68 4-69 4-70	4-36 4-56	
		Chapter 5	Technicals	Others	
			5-10 5-20 5-31 5-32	5-18 5-11	
		Chapter 6	Technicals	Others	
				6-9 6-10	
		Appendix 1	Technicals	Others	
				A1-1 A1-2	
		Appendix 2	Technicals	Others	
				A2-12 A2-22 A2-26	

B

Revision	Date	Revised page			Approved
			Technicals	Others	
Second Revision B	Dec. 1999	Appendix 3	A3.1-2 A3.1-7 A3.2-1 A3.2-2 A3.2-3 A3.2-7 A3.4-1 A3.4-2 A3.4-3 A3.4-7 A3.5-10 A3.6-2 A3.6-5 A3.6-6 A3.6-9 A3.7-2 A3.7-5 A3.7-6 A3.7-9 A3.9-1 A3.9-2	A3.1-1 A3.1-6 A3.2-6 A3.3-1 A3.4-6 A3.5-1 A3.5-9 A3.6-1 A3.7-1 A3.8-1 A3.9-1	Tomio Nakano

B

Revision	Date	Revised page			Approved
Second Revision C	Dec. 2001	Contents etc.	Technicals	Others	
				0-3~6 0-10~19	
		Chapter 1	Technicals	Others	
			1-4~7 1-18~19	1-1~3 1-13 1-20~21	
		Chapter 2	Technicals	Others	
			2-1 2-7 2-14~21		
		Chapter 3	Technicals	Others	
			3-12 3-16~18 3-21~22 3-26		
		Chapter 4	Technicals	Others	
			4-4~8 4-21~23 4-29 4-37~45 4-50~51 4-58~59 4-71	4-24~28 4-30~36 4-46~49 4-54~56	
		Chapter 5	Technicals	Others	
				5-1 5-4 5-7 5-17 5-19	
		Chapter 6	Technicals	Others	
			6-10 6-12	6-3 6-9	

C

<table>
<tr><th>Revision</th><th>Date</th><th colspan="3">Revised page</th><th>Approved</th></tr>
<tr><td rowspan="6">Second
Revision C</td><td rowspan="6">Dec. 2001</td><td rowspan="2">Appendix 1</td><td>Technicals</td><td>Others</td><td rowspan="6"></td></tr>
<tr><td>A1-2</td><td>A1-1</td></tr>
<tr><td rowspan="2">Appendix 2</td><td>Technicals</td><td>Others</td></tr>
<tr><td>A2-25</td><td>A2-6
A2-11
A2-20</td></tr>
<tr><td rowspan="2">Appendix 3</td><td>Technicals</td><td>Others</td></tr>
<tr><td>A3.1-1~2
A3.1-6
A3.2-1~3
A3.2-6~7
A3.3-1
A3.3-3
A3.3-5~7
A3.4-1~3
A3.4-6
A3.5-1
A3.5-4
A3.6-1
A3.7-1
A3.8-1
A3.9-1
A3.10-1~10
A3.11-1~4</td><td>A3.1-3~5
A3.2-4~5
A3.4-2
A3.4-4~5
A3.4-7
A3.5-4
A3.6-2
A3.6-5
A3.7-2</td></tr>
</table>

C

CONTENTS

CHAPTER 1. INTRODUCTION

CHAPTER 2. MISSION PERFORMANCE

CHAPTER 3. ENVIRONMENTS

CHAPTER 4. INTERFACE REQUIREMENTS

C

CHAPTER 5. LAUNCH OPERATIONS

A

CHAPTER 6. INTERFACE MANAGEMENT

APPENDIX 1. HISTORY OF NASDA LAUNCH VEHICLES

A

APPENDIX 2. OUTLINE OF TANEGASHIMA ISLAND AND TANEGASHIMA SPACE CENTER

A

APPENDIX 3. PAYLOAD ADAPTER

B

C

FIGURES

C

TABLES

ABBREVIATIONS AND DEFINITIONS

AB	Administration Building	
ADEOS	Advanced Earth Observation Satellite	
AGE	Aerospace Ground Equipment	
AH	Ampere - Hour	
BET	Best Estimate Trajectory	
B / H	Blockhouse	
CAM	Collision Avoidance Maneuver	A
CCW	Counterclockwise	
CDR	Command Destruct Receiver	
CG	Center of Gravity	
CLA	Coupled Loads Analysis	
COMETS	Communications and Broadcasting Engineering Test Satellite	
DLF	Design Load Factor	
DOP	Dilution of Precision	
EMC	Electromagnetic Compatibility	
EMI	Electromagnetic Interference	
ESA	European Space Agency	
ETS	Engineering Test Satellite	
FM	Flight Model	
FRR	Flight Readiness Reviewer	
FSO	Flight Safety Officer	
GCC	Guidance Control Computer	
GEO	Geostationary Earth Orbit	
GFRP	Glass Fiber Reinforced Plastic	
GHe	Gaseous Helium	
GMS	Geostationary Meteorological Satellite	
GN_2	Gaseous Nitrogen	
GOX	Gaseous Oxygen	
GPSR	Global Positioning System Receiver	C
GSE	Ground Support Equipment	
GTO	Geostationary Transfer Orbit	
h	Altitude	
ha	Apogee Altitude	
hp	Perigee Altitude	
HTV	H-II Transfer Vehicle	C

ICS	Interface Control Specification	
i	Inclination	
IDF	Intermediate Distributing Frame	A
in.	inch	
INMARSAT	International Maritime Satellite Organization	
ISAS	Institute of Space and Astronautical Science	
LB	Launch Building	
LCDR	Launch Conductor	
LE	Liquid Rocket Engine	
LEO	Low Earth Orbit	
LH_2	Liquid Hydrogen	
LOX	Liquid Oxygen	
LP	Launch Pad	
LPLF	Lower Payload Fairing	
LRB	Liquid Rocket Booster	
LSOM	Launch Site Operations Manager	
LVRR	Launch Vehicle Readiness Reviewer	
MECO	Main Engine Cutoff	
MECOM	Main Engine Cutoff Command	
ML	Movable Launcher	A
MOD	Mission Operations Department	A
MRR	Mission Readiness Reviewer	
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	
NMD	NASDA Mission Director	
NQA	NASDA Quality Assurance Monitor	
NSAFE	NASDA Pad Safety Officer	
NSO	NASDA Safety Officer	
NTO	Nitrogen Tetra Oxide	
OA	Overall	
OIS	Operational Intercommunication Telephone System	
OLR	Osaki Launch Range	

OSTS	Office of Space Transportation Systems
PC_2	Second Stage Propellant Consumption
PCD	Pitch Circle Diameter
PFM	Protoflight Model
PIF	Poly Iso-cyanurate Form
pl	Place
PLA	Payload Adapter
PLF	Payload Fairing
PSS	Payload Support Structure
PST	Pad Service Tower
QD	Quick Disconnector
R	Radius
RCC	Range Control Center
RCO	Range Control Officer
RCS	Reaction Control (gas jet) System
REF	Reference
RF	Radio Frequency
RL	Rocket Launcher
Q	Dynamic Pressure
SBB	Solid Booster Test Building
SC	Spacecraft Conductor
S / C	Spacecraft
SCC	Satellite Control Center
SCRR	Spacecraft Readiness Reviewer
SDB	Sequence Distribution Box
SECO	Second Engine Cutoff
SECOM	Second Engine Cutoff Command
SECT	Section
SEIG	Second Engine Ignition
SEP	Separation
SFA	Spacecraft and Fairing Assembly Building
SFU	Space Free Flyer Unit
SLB	Supporting Launch Building
Sm^3	Standard Cubic Meter

A

SOB	Strap-on Booster
SPL	Sound Pressure Level
SRB	Solid Rocket Booster
SSB	Solid Strap-on Booster
SSO	Sun-synchronous Orbit
STA	Spacecraft Test and Assembly Building
STA	Station
STM	Structural Test Model
T.B.C.	To Be Confirmed
T.B.D.	To Be Determined
T.B.R.	To Be Revised
TNSC	Tanegashima Space Center
TRMM	Tropical Rainfall Measuring Mission
TSA	Third Stage and Spacecraft Assembly Building
TT/C	Telemetry, Tracking, and Command
TVC	Thrust Vector Control
UHF	Ultra High Frequency
UM	Umbilical Mast
UPLF	Upper Payload Fairing
UPS	Uninterruptible Power System
VAB	Vehicle Assembly Building
VDC	Voltage Direct Current
VHS	Video Home System
VOS	Vehicle On Stand
α	Angle of Attack
ϕ	Diameter
ω	Argument of Perigee
Ω	Ascending Node

| A

| A

CHAPTER 1.

INTRODUCTION

CHAPTER 1. INTRODUCTION

1.1 Purpose of the User's Manual

This manual provides users with information on the H-IIA launch system, spacecraft / launch vehicle interfaces and the related NASDA launch services.

It presents mission performance, launch and flight environments and interface requirements to be considered in the spacecraft design. Launch operations of the spacecraft, interface management and documentation, and outlines of the H-IIA launch vehicle and Tanegashima Space Center (TNSC) are also described.

There are two more documents related to H-IIA launch services on launch facilities and range safety. A brief description of these documents is provided in § 1.3.

1.2 H-IIA Launch System

The H-IIA launch system, which includes the launch vehicle, launch facilities, etc., is developed by NASDA as an upgraded version of the H-II launch system to answer the various mission demands for many types of payloads, such as the spacecraft of 2-ton class to 3-ton class Geostationary Earth Orbit* (GEO) in the beginning of the 21st century.

The H-IIA development policies are shown below:

a) To support various demands for many kinds of payloads
b) To keep the high reliability and heritage using the technology, experience, personnel and facilities demonstrated in the H-II program
c) To improve operability
d) To optimize launch opportunities
e) To reduce launch costs

This section provides a brief description of the H-IIA launch vehicle and launch facilities.

<Note>
* : The H-IIA launch vehicle injects the spacecraft to Geostationary Transfer Orbit (GTO) instead of injecting it to GEO directly. So, 2-ton or 3-ton class GEO means a general conversion capability from GTO to GEO.

1.2.1 H-IIA launch vehicle

(1) General

Figure 1.2.1 shows H-IIA launch vehicle configurations, and Table 1.2.1 shows a summary of H-IIA subsystems and characteristics.

Discrimination names are used to distinguish a H-IIA launch vehicle configuration as shown in the following:

discrimination name format: H2Aabcd, where

a : type of stage (single-stage type = 1, two-stage type = 2) | C b : number of Liquid Rocket Boosters (LRB)

c : number of Solid Rocket Booster-As (SRB-A)

d : number of Solid Strap-on Boosters (SSB) (if SSB is not used, this number is omitted)

The H-IIA launch vehicle is designed with two stages, each powered by engines using liquid hydrogen and liquid oxygen propellant.

There are two models, H2A202 Series and H2A212 (or whether LRB is applied or not). Furthermore, in the H2A202 Series, there are three variations with different numbers of SSB.

The H2A202 Series H-IIA launch vehicle, which is named H2A202d (d is the number of SSB), consists of the first stage, two SRB-As, the second stage and the payload fairing as shown in Figure 1.2.2. In addition, two or four SSBs are strapped to the side of the first stage according to a user's requirement.

The H2A212 consists of the first stage, two SRB-As, one LRB, the second stage and the payload fairing as shown in Figure 1.2.3. The H2A212 is not equipped with SSBs.

The first stage, the SRB-A, the second stage and the payload fairing are commonly used for H2A202 Series and H2A212 class configurations.

(2) First stage

The first stage is composed of LOX / LH_2 propellant tanks, the engine section, the LE-7A engine, the center body section, the interstage section and the avionics system. In addition, two or four SSBs are attached according to mission requirements.

Both propellant tanks are enlarged to increase capacity, using the same aluminum alloy isogrid structures, insulation (PIF) and cryogenic technology as those of the H-II first stage, although spun-formed tank domes are used for both tanks (instead of orange-peel domes) and an attachment ring is added to the LH_2 tank cylinder to attach SRB-As and SSBs.

The LE-7A engine is a modified model of the LE-7 engine, with improvement in reliability and operability while retaining most of the characteristics and performance (excepting ISP) of the LE-7 engine.

The engine section structure, which is an aluminum alloy monocoque, is modified based on the structure of H-II vehicle so that the first stage can stand directly on the launch pad and support two SRB-As, LRBs, and also SSBs.

The center body section is an aluminum alloy semi-monocoque structure, and connects the LOX tank and the LH_2 tank as in the structure of the H-II vehicle; the structure which straps the LRB can be added to this section. A

The interstage section is newly made of CFRP. The aft end of this section is connected to the front end of the first stage LOX tank. On the other hand, its front end is bolted to the aft end of the second stage LH_2 tank, where the first and second stage separation mechanism which uses a linear shaped charge is installed.

The avionics system whose main components are installed in the center body section is newly developed. This system uses the serial data bus system (MIL-STD-1553B) and is connected electrically to each avionics system of the second stage and the LRB by this bus system. The command signals from the guidance control computer of the second stage (GCC2) are transmitted to the guidance control computer of the first stage (GCC1). The GCC1 sends command signals to each component and controls each nozzle angle of the LE-7A engine and the SRB-A.

Two or four SSBs are attached to the engine section and the attachment ring of the first stage LH_2 tank. In the case of the H2A2022, two SSBs are ignited at about 50 seconds after lift-off. In the case of the H2A2024, four SSBs are ignited separately in pairs after lift-off. One pair is ignited at about 10 seconds after lift-off and another pair is ignited at about 76 seconds.

(3) LRB (Liquid Rocket Booster)

The LRB has design features similar to those of the first stage except the number of LE-7A engines and application of a nose cone instead of the interstage section. The LRB has two LE-7A engines which are clustered and ignited prior to lift-off. Engine nozzles are controlled by the guidance control computer of the LRB (GCCL). Each engine is cut off separately with time lag to minimize a bias angle of the each engine nozzle.

(4) SRB-A (Solid Rocket Booster)

The SRB for the H-IIA, named SRB-A, is redesigned based on the solid rocket technology of the H-II SRB to enhance its reliability and operability. Two SRB-As are strapped on the core stage and augment LE-7A engine thrust from lift-off for about 100 seconds.

The motor case of the SRB-A is monolithic and made of filament winding composite material (CFRP) using Thiokol Castor 120 technology. To enhance the launch capability, the propellant capacity is increased by about 6 tons optimizing the thrust pattern. Further, to enhance reliability, electromechanical thrust vector control (TVC) system for gimbaling the nozzle is applied.

(5) Second stage

The second stage is composed of LOX / LH_2 propellant tanks, the LE-5B engine, the reaction control (gas jet) system (RCS), and the avionics system.

In the H-II, it has the integrated tank with common bulkhead between the LOX and LH_2 tanks. In the H-IIA vehicle, separated LOX and LH_2 tanks are adopted to improve operability.

The LH_2 propellant tank uses the same aluminum alloy isogrid structure, insulation (PIF) and cryogenic technology as the H-II, although spun-formed tank domes are used instead of orange peel domes. The LOX propellant tank is made by welding two elliptical spun-formed tank domes of the same aluminum alloy as the LOX tank of the H-II vehicle.

The LE-5B engine is a modified model of the LE-5A engine with improved reliability and operability, enhancing the thrust from 12.4 tonf to 14 tonf. This engine is able to operate at two thrust levels of 100 % and 60 % for the rated thrust, and in the idle mode where the thrust level is about 3 % for the rated thrust. Further, this engine has a multiple restart capability and a long duration coast capability.

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 component equipment panel of the guidance section and uses hydrazine as its propellant.

Most components of the avionics system are mounted on the component equipment panel. Most of data processing for the equipment mounted on the second stage and output of the control signal to the equipment are performed by the GCC2. The programs mounted on the GCC1 and GCCL are controlled by the program loaded in the GCC2.

(6) Payload fairings

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

Users are able to choose from 6 types described in § 1.2.3 according to user's requirements. Each type is compatible with every H-IIA launch vehicle.

Description on payload fairings is provided further in Chap. 4.

(7) Payload adapters / separation system

The spacecraft is mounted on the launch vehicle using a payload adapter described in § 1.2.3. Detailed information on the adapters and their separation systems is presented in Chap. 4.

Each H-IIA launch vehicle is compatible with each type of payload adapters (including separation systems).

1.2.2 Launch facilities

Launch operations are carried out in Osaki Launch Range (OLR) of TNSC which is located at the southeastern end of Tanegashima island. TNSC is the largest rocket range in Japan with the total site area of 8.6 million square meters and accommodates various necessary facilities to launch spacecraft.

Necessary facilities for spacecraft launch operations are mostly located in

OLR. Figure 1.2.4 shows locations of major facilities in OLR. Major facilities related to spacecraft operations are shown in Figures 1.2.5 to 1.2.8. Launch processing in these facilities is normally performed as shown in Figure 1.2.9. If a user requests different processing, the user should coordinate with NASDA.

1.2.3 Payload accommodations

NASDA offers 6 types of payload fairings and several types of payload adapters. Figure 1.2.10 shows payload usable envelops for the model 4S and 5S payload fairings used for a single launch. Figure 1.2.11 shows payload usable envelops for the model 4/4D-LS, 4/4D-LC and 5/4D payload fairing used for dual launch. Figure 1.2.12 shows payload usable envelop for the model 5S-H payload fairing used for a single launch. Each fairing is compatible with each of the H-IIA launch vehicle configurations. More information of the payload fairing is presented in § 4.4. | C

Figure 1.2.13 shows example of payload adapters. Detailed information related to interface with the spacecraft is presented in § 4.5 and APPENDIX 3. | C

1.2.4 Users / NASDA relationship

NASDA offers a full launch service from interface coordination at the beginning of mission planning through a separation of the spacecraft on the required orbit. For executing this service, NASDA provides a single point of contact, the program manager, for each spacecraft. The program manager shall be responsible for contracts and coordination related to launch. Figure 1.2.14 shows a concept of users / NASDA relationship and major responsibilities of the program manager. After establishment of the NASDA launch operations team, although the mission director is designated and responsible only for the technical coordination with the user, the program manager commands this coordination. Figure 1.2.15 shows this relationship. | C

1.2.5 Advantages of H-IIA

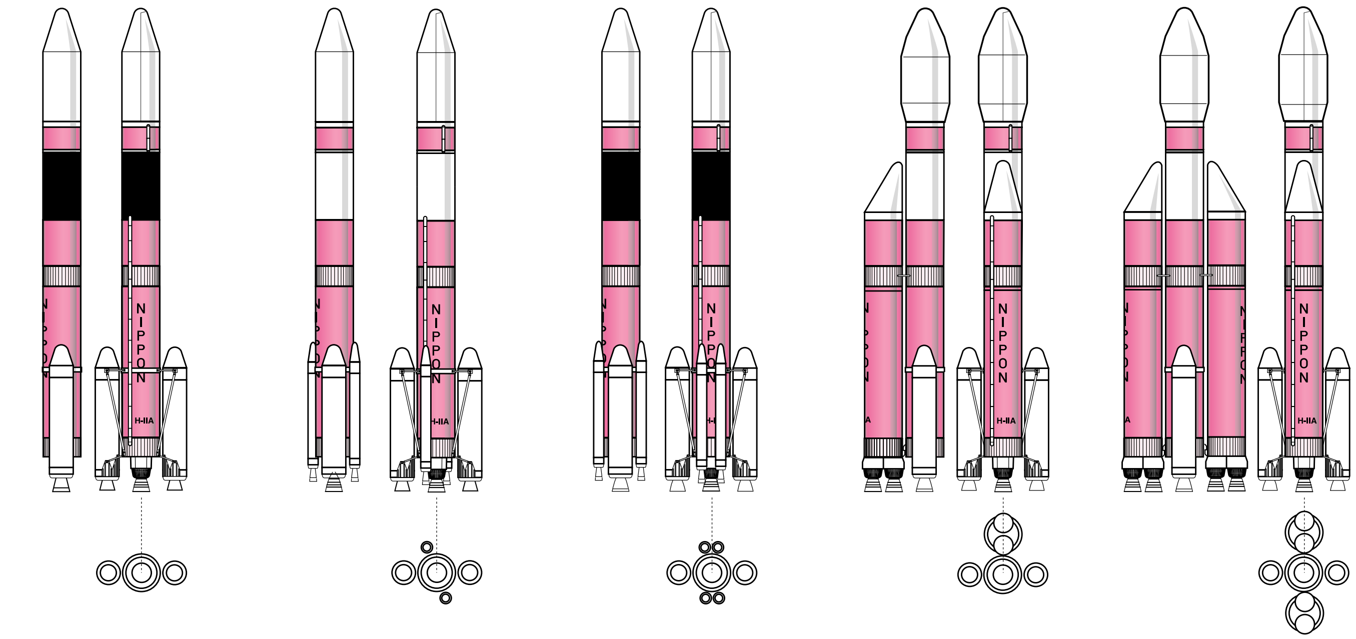
The H-IIA launch system and launch services offer the following advantages.

a) The H-IIA launch vehicle family provides flexible and various launch capabilities for payloads.

b) The H-IIA launch vehicle and launch operations are very simple and reliable.

c) Environmental conditions to which the spacecraft is exposed are comparable with or better than those of other launch vehicles.

d) The H-IIA launch system is capable of dual launch if a user requires.



H2A202	H2A2022	H2A2024	H2A212	H2A222
(GTO 4.1/3.7 ton)	(GTO 4.5/4.1 ton)	(GTO 5.0/4.6 ton)	(GTO 7.5 ton)	(GTO T.B.D. ton)
(*1) (*2)	(*1) (*2)	(*1) (*2)	(*1)	

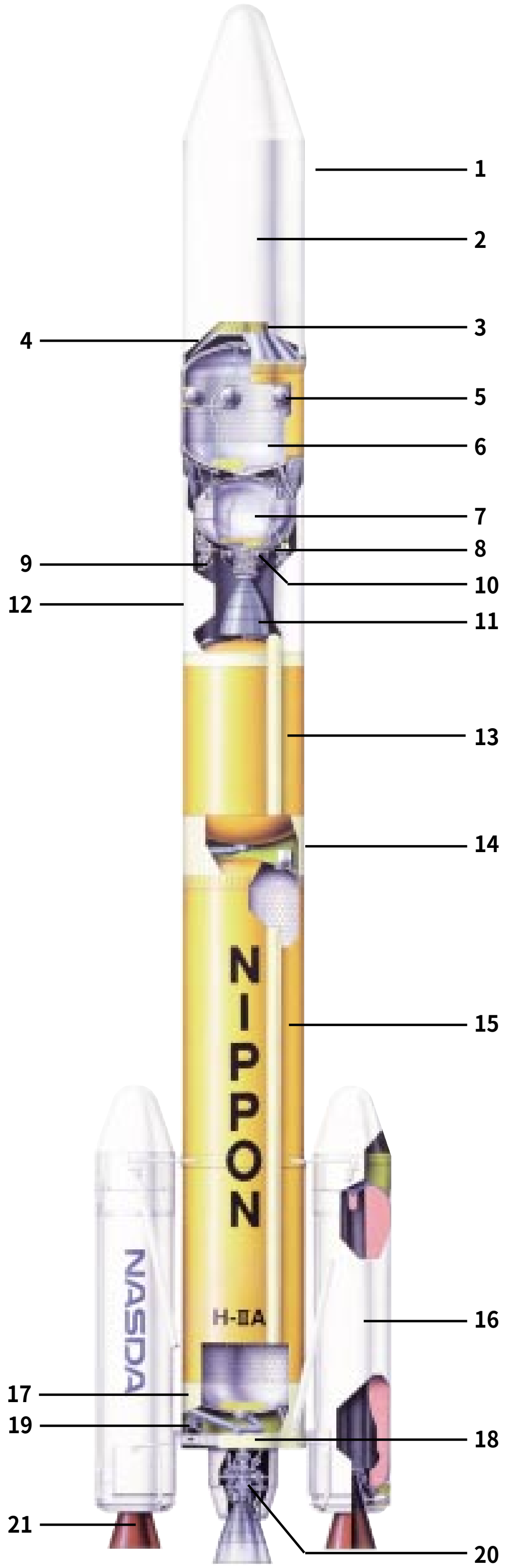
Current Program   Future Program

Figure 1.2.1 H-IIA Launch Vehicle Family

(*1):With the LE-7A lower nozzle skirt
(*2):Without the LE-7A lowor nozzle skirt

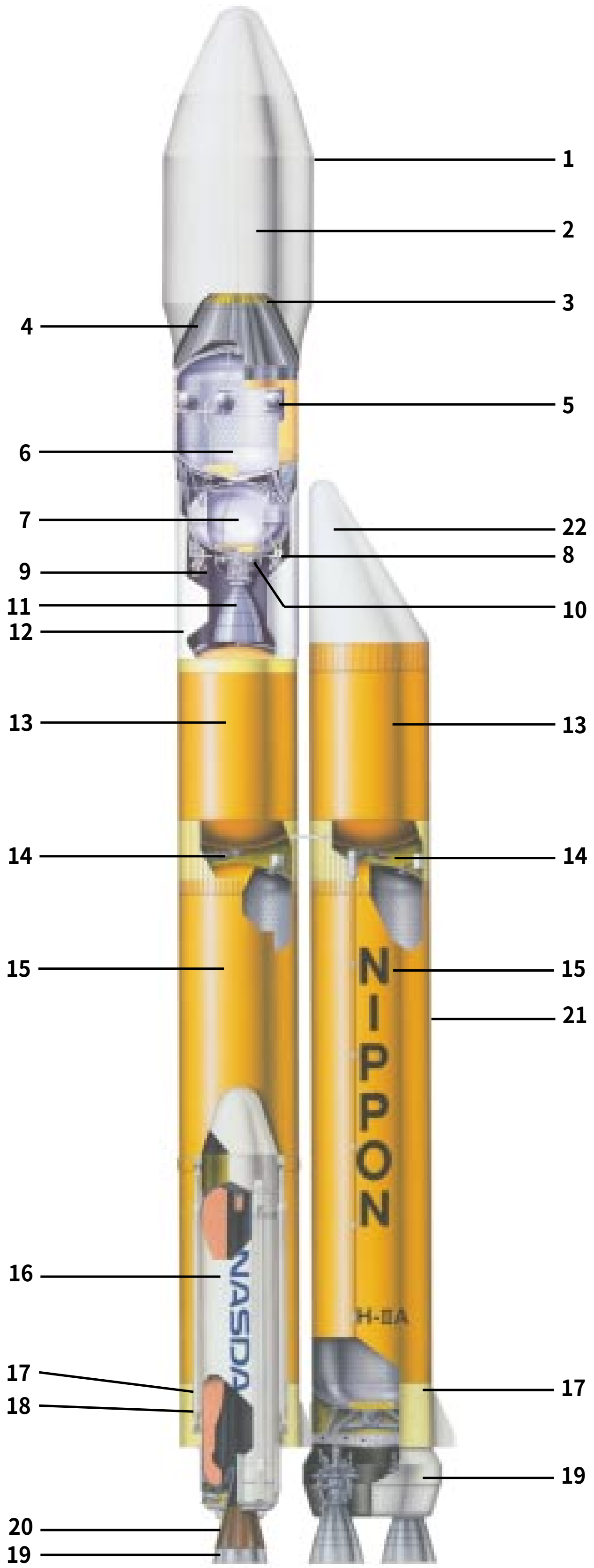
Table 1.2.1 Summary of H-IIA subsystems and characteristics

Item		H2A202	H2A2022	H2A2024	H2A212	H2A222	Note
Overall length (m)		53	53	53	53	53	
Diameter (m)	: Core	4.0	4.0	4.0	4.0	4.0	
	: Fairing	4.07 / 5.1	4.07 / 5.1	4.07 / 5.1	4.07 / 5.1	4.07 / 5.1	4S, 4/4D type / 5S, 5/4D type
Total weight (ton)	: Without Payload	285	316	347	403	520	include payload adapter
Payload weight (ton)	: With the LE-7A lower nozzle skirt	4.1	4.5	5.0	7.5	T.B.D.	(100 kg) GTO for 4S fairing
	: Without the LE-7A lower nozzle skirt	3.7	4.1	4.6			(5S fairing for H2A212)
Payload fairing	: Honeycomb sandwich						
4S,5S type	: Clamshell	✓	✓	✓	✓	✓	
4/4D-LS, 5/4D type	: Upper Clamshell, Lower Tandem	✓	✓	✓	✓	✓	
4/4D-LC type	: Upper Clamshell, Lower Clamshell	✓	✓	✓	✓	✓	
5S-H type	: Clamshell				✓	✓	
Payload adapter							
937M-Spin, 937M-Spin-A	: Aluminum, V band, Spin table	✓	✓	✓	✓	✓	
937M, 937MH	: CFRP, V band	✓	✓	✓	✓	✓	
1666M, 1194M, 1666MA	: Aluminum, V band	✓	✓	✓	✓	✓	
2360S, 3470S, 1666S	: Separation Nuts	✓	✓	✓	✓	✓	
First stage		1	1	1	1	1	
Propellant	: LH_2 / LOX	100 ton	100 ton	100 ton	100 ton	100 ton	usable weight
Tank	: LH_2 aluminum isogrid						
	: LOX aluminum isogrid						
Propulsion	: LE-7A engine x 1						
	Thrust 1100 kN						in vacuum
	: With the LE-7A lower nozzle skirt						
	1073 kN						in vacuum
	: Without the LE-7A lower nozzle skirt						
	Isp 440 sec						in vacuum
	: With the LE-7A lower nozzle skirt						
	429 sec						in vacuum
	: Without the LE-7A lower nozzle skirt						
	MR 5.9						
	Burning time 390 sec						
	: Auxiliary engine time 390 sec						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)
LRB		N / A	N / A	N / A	1	2	
Propellant	: LH_2 / LOX				99.2 ton	99.2 ton	usable weight
Tank	: Same as first stage						
Propulsion	: LE-7A engine x 2						cluster
	Burning time 195 sec (no throttling)						
Avionics	: Guidance Control Computer, Flight Termination, Electrical Power						use data bus system (MIL-STD-1553B)
SRB-A		2	2	2	2	2	
Propellant (per each)	: HTPB composite						
	Propellant weight 65.04 ton						
	Thrust (max) 2260 kN						in vacuum
	Isp 280 sec						in vacuum
	Burning time 100 sec						
Motor case	: Monolithic CFRP						
	Diameter 2.5 m						
SSB		0	2	4	N / A	N / A	two SSB are one pair
Propellant (per each)	: HTPB composite						
	Propellant weight 13.1 ton						
	Thrust (Max) 710 kN						in vacuum
	Isp (mean) 283 sec						in vacuum
	Burning time 60 sec						
Motor case	: Steel						
	Diameter 1.02 m						
Second stage		1	1	1	1	1	
Propellant	: LH_2 / LOX	16.6 ton	16.6 ton	16.6 ton	16.6 ton	16.6 ton	usable weight
Tank	: LH_2 aluminum isogrid						
	: LOX aluminum						
Propulsion	: LE-5B engine x 1						
	Thrust 137 kN						
	Isp 447 sec						
	MR 5.0						for rated thrust
	Throttling 60 %						about 3 % for rated thrust
	Idle mode function						
	Multi-restart function						
	: Reaction control system (hydrazine)						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)



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 A
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 A

Figure 1.2.2 H-IIA (H2A202) launch vehicle configuration A



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 | A
9. Reaction control system
10. Ambient He bottles
11. Second stage engine (LE-5B engine) | A
12. Interstage section
13. First stage / LRB LOX tank
14. First stage / LRB center body section
15. First stage / LRB LH_2 tank
16. Solid rocket booster (SRB-A)
17. First stage / LRB engine section
18. Auxiliary engine
19. First stage / LRB engine (LE-7A engine)
20. SRB-A movable nozzle
21. Liquid rocket booster (LRB)
22. LRB nose cone

| A

Figure 1.2.3 H-IIA (H2A212) launch vehicle configuration

A

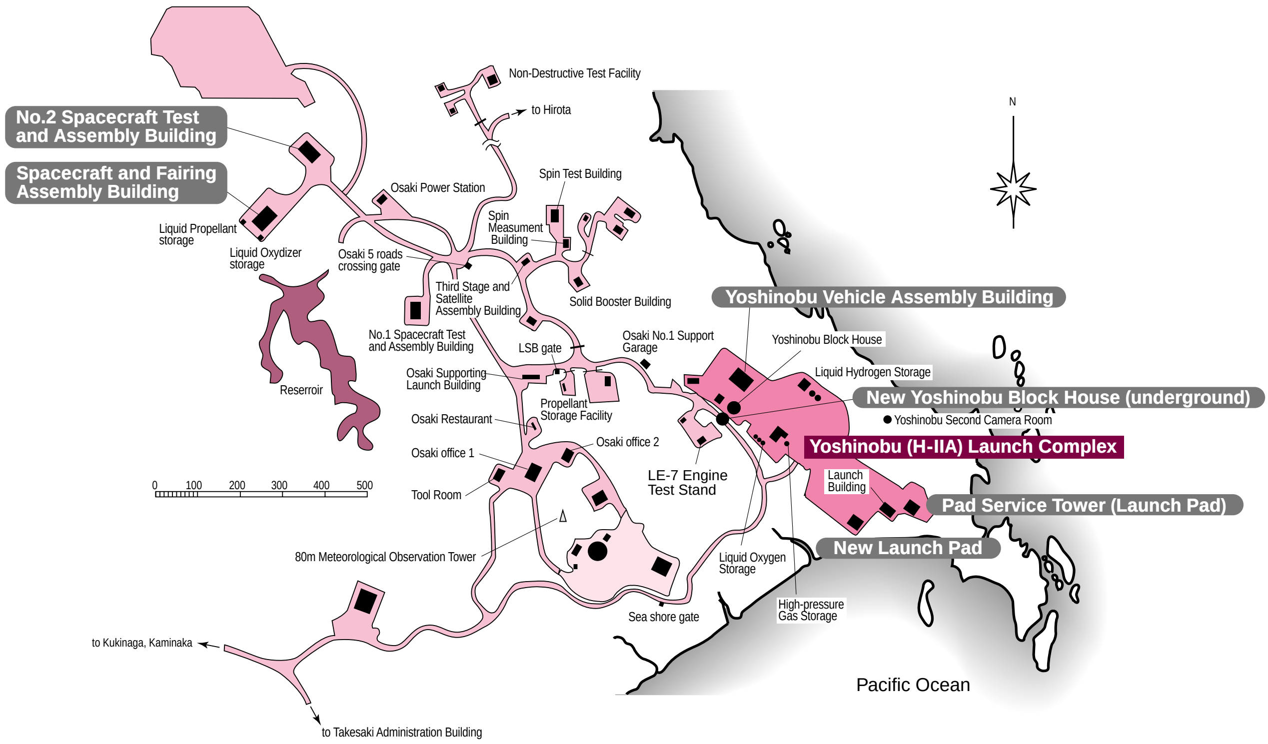


Figure 1.2.4 Location of major facilities in Osaki Launch Range

Figure 1.2.5 No. 2 Spacecraft test and assembly building (STA2)

Figure 1.2.6 Spacecraft and fairing assembly building (SFA)

B

Figure 1.2.7 Vehicle assembly building (VAB) A

Figure 1.2.8 Overview of New Yoshinobu Launch Complex | C

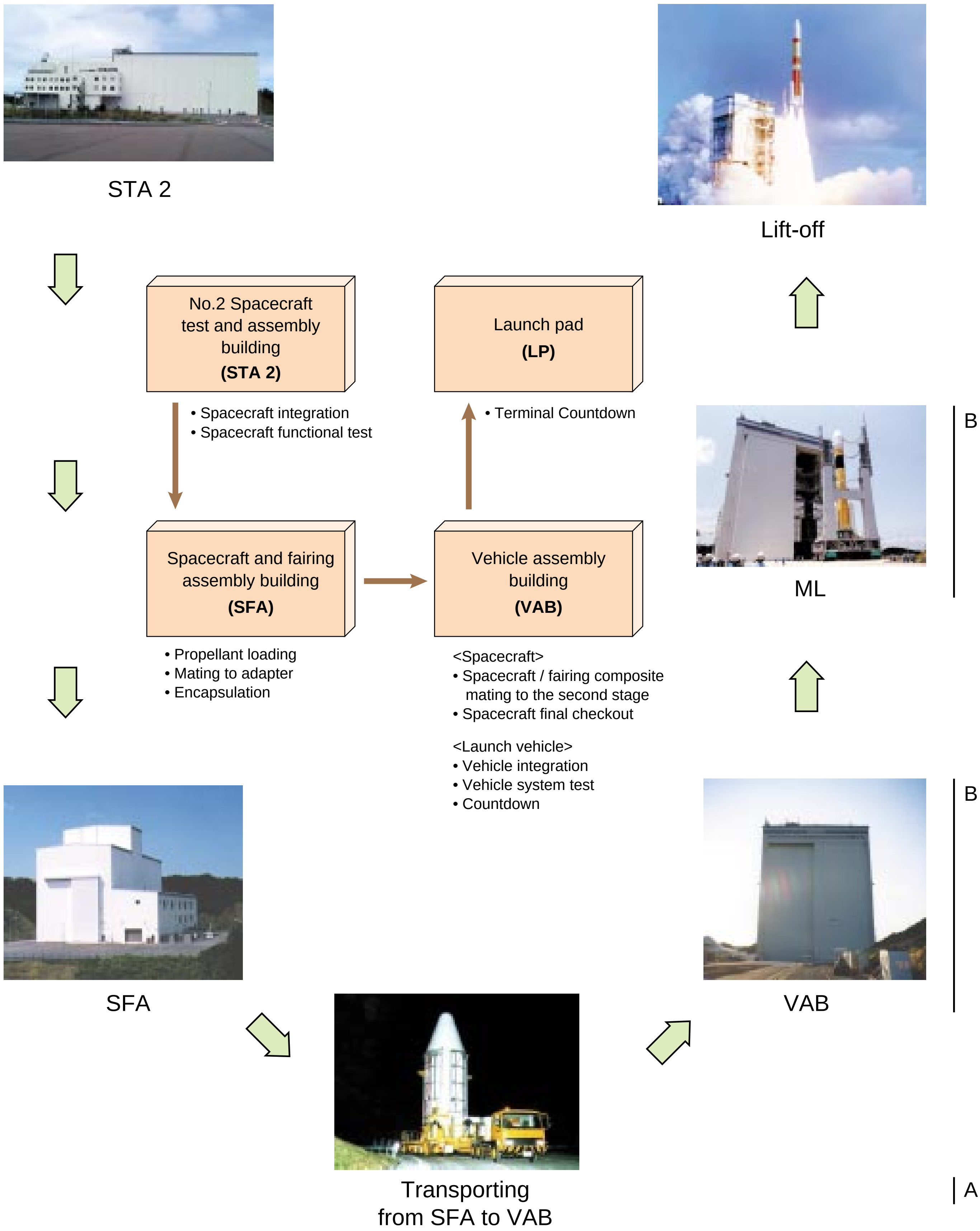
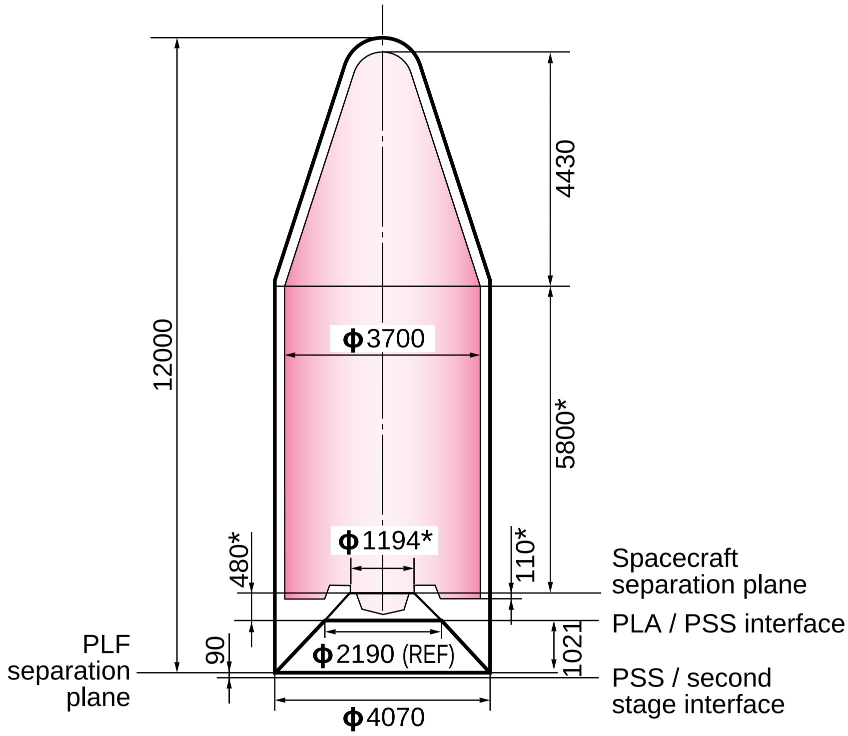
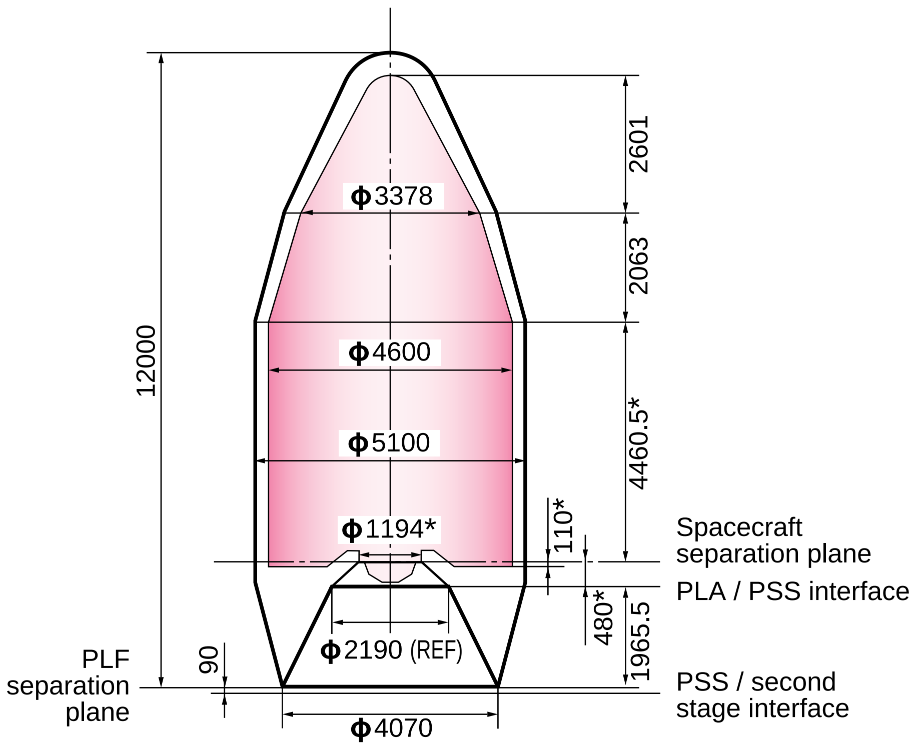


Figure 1.2.9 H-IIA launch operations process



Model 4S



Model 5S

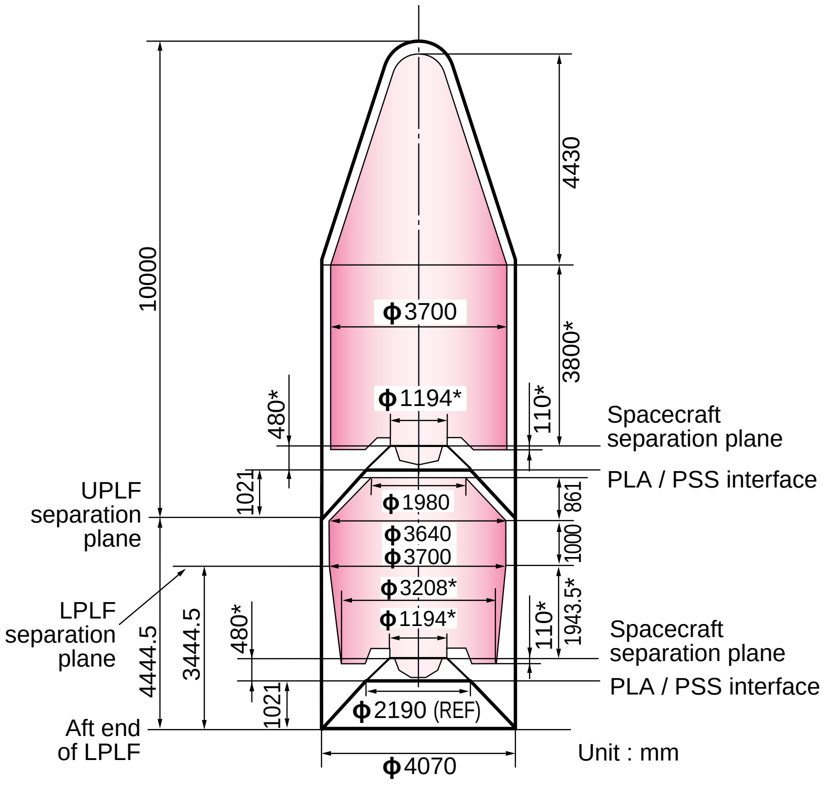
<Note>
PLF : Payload fairing
PLA : Payload adapter
PSS : Payload support structure
* : These values will vary with adapter model

Figure 1.2.10 Payload fairings for single launch with 1194M adapter | A

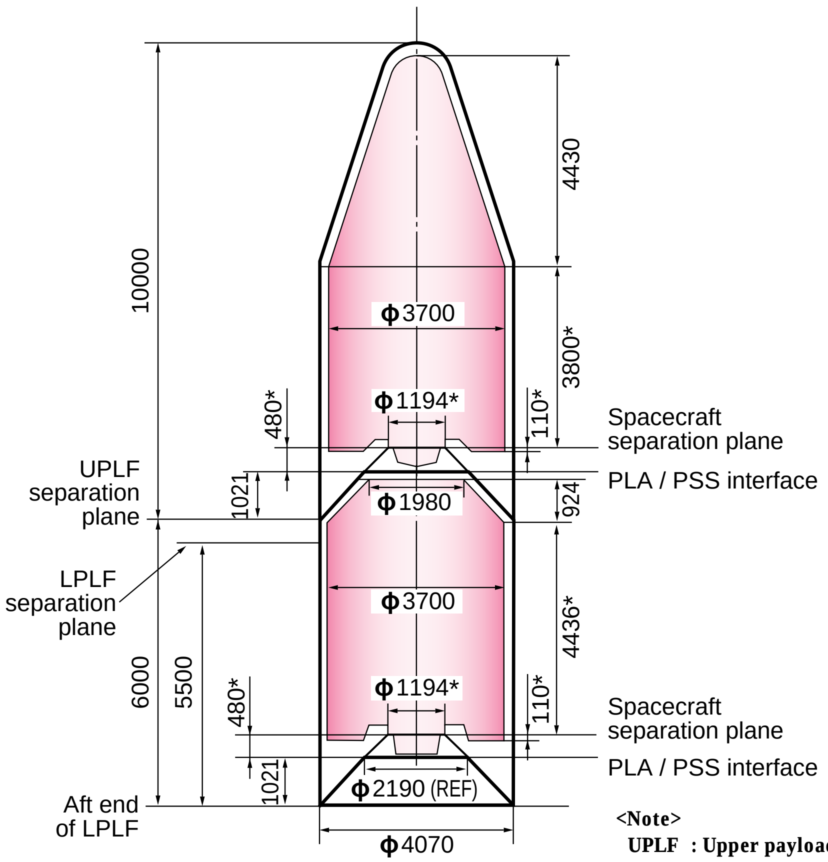
| A

| A

| A



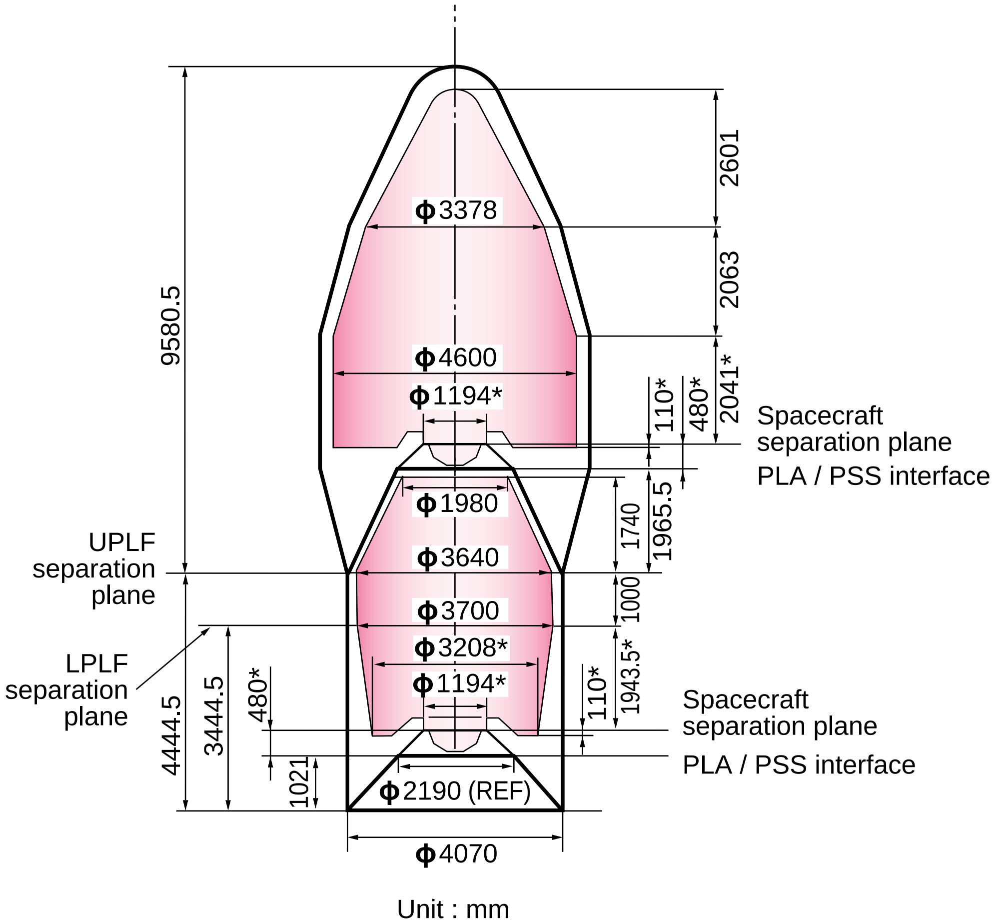
Model 4/4D-LS



Model 4/4D-LC

<Note>
UPLF : Upper payload fairing
LPLF : Lower payload fairing
PLA : Payload adapter
PSS : Payload support structure
* : These values will vary with adapter model

Figure 1.2.11(1/2) Payload fairings for dual launch with 1194M adapter

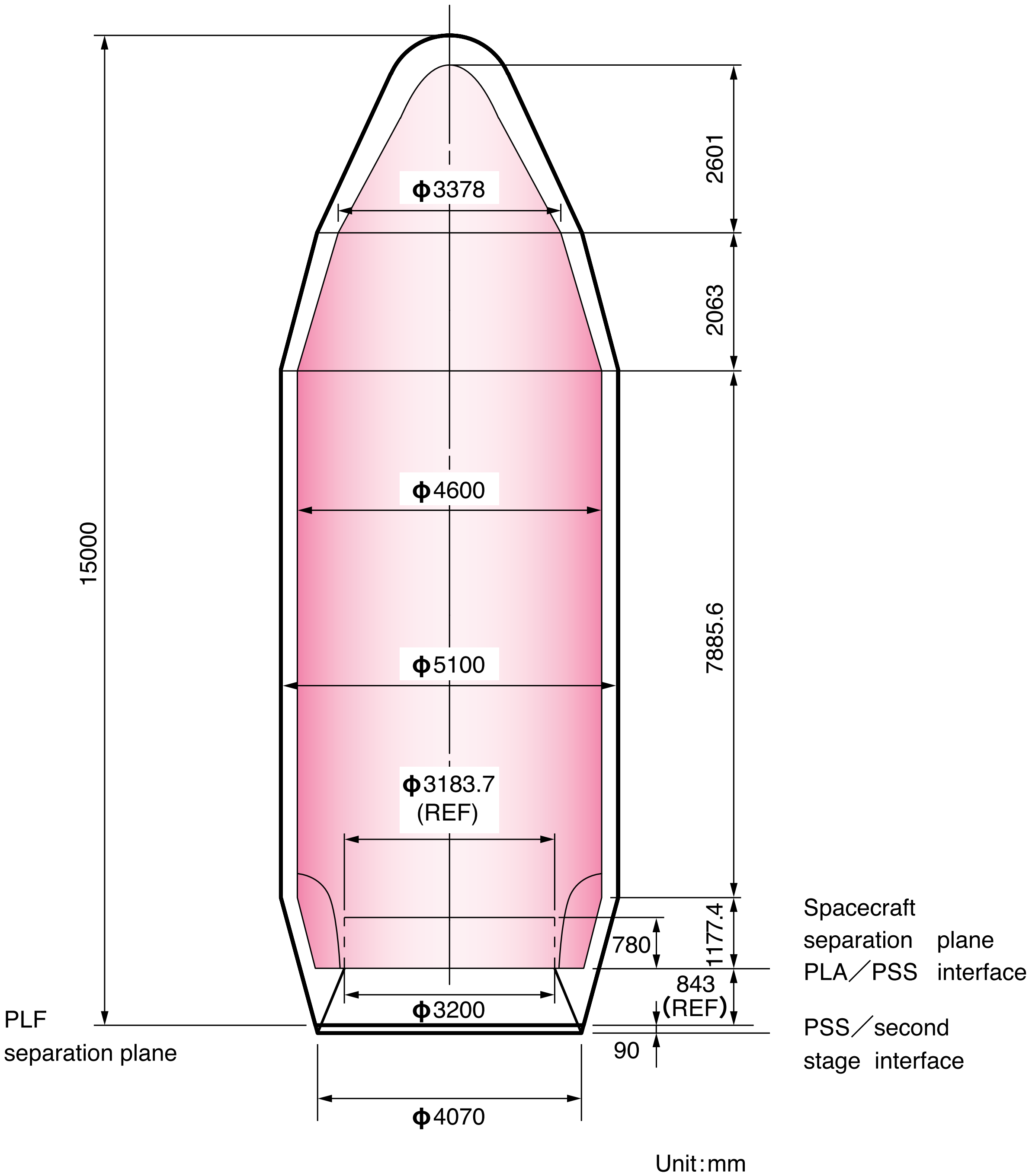


Model 5/4D

<Note>

UPLF : Upper payload fairing
LPLF : Lower payload fairing
PLA : Payload adapter
PSS : Payload support structure
* : These values will vary with adapter model

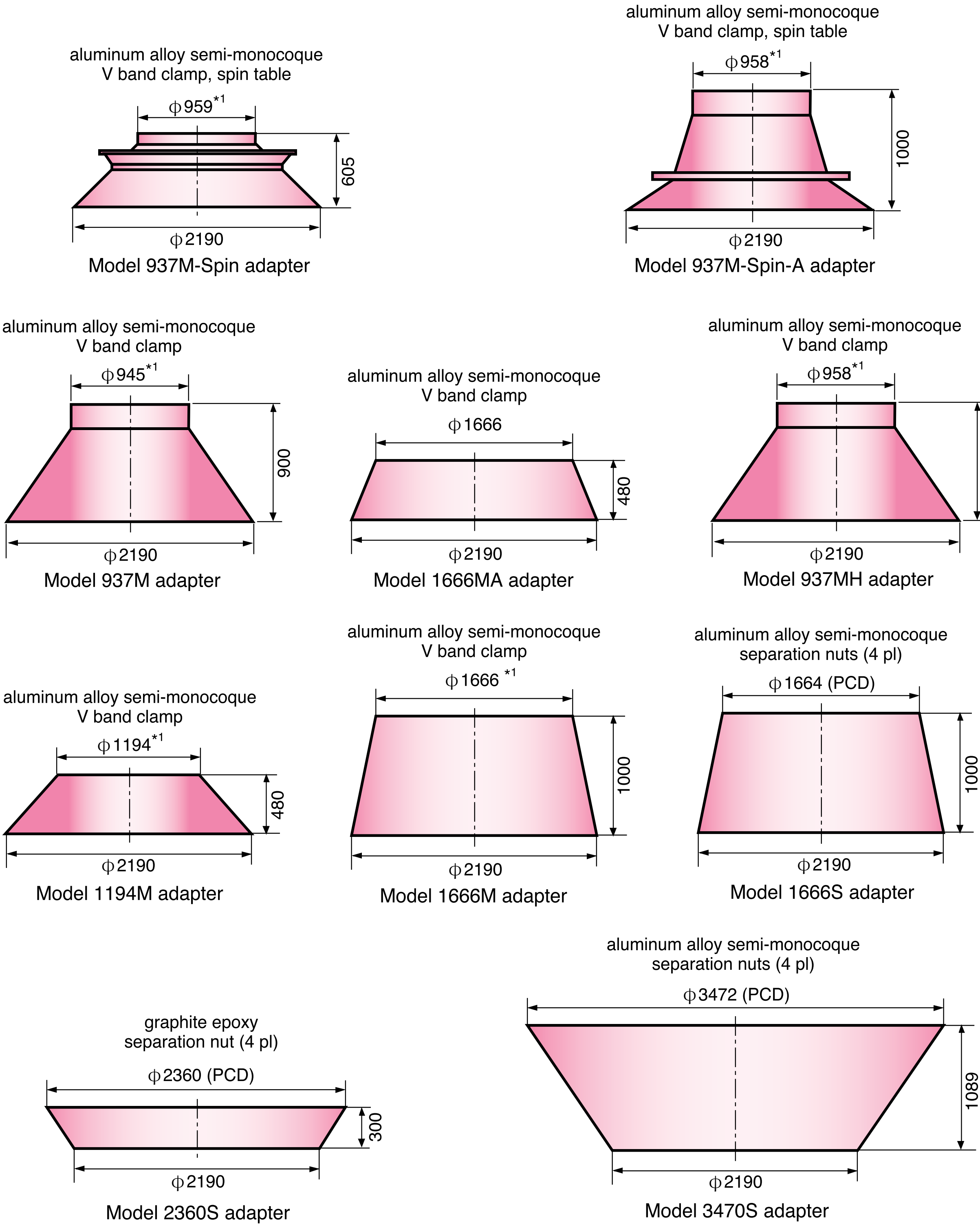
Figure 1.2.11(2/2) Payload fairings for dual launch with 1194M adapter | A



Model 5S-H

<Note>
PLF : Payload fairing
PLA : Payload adapter
PSS : Payload support structure

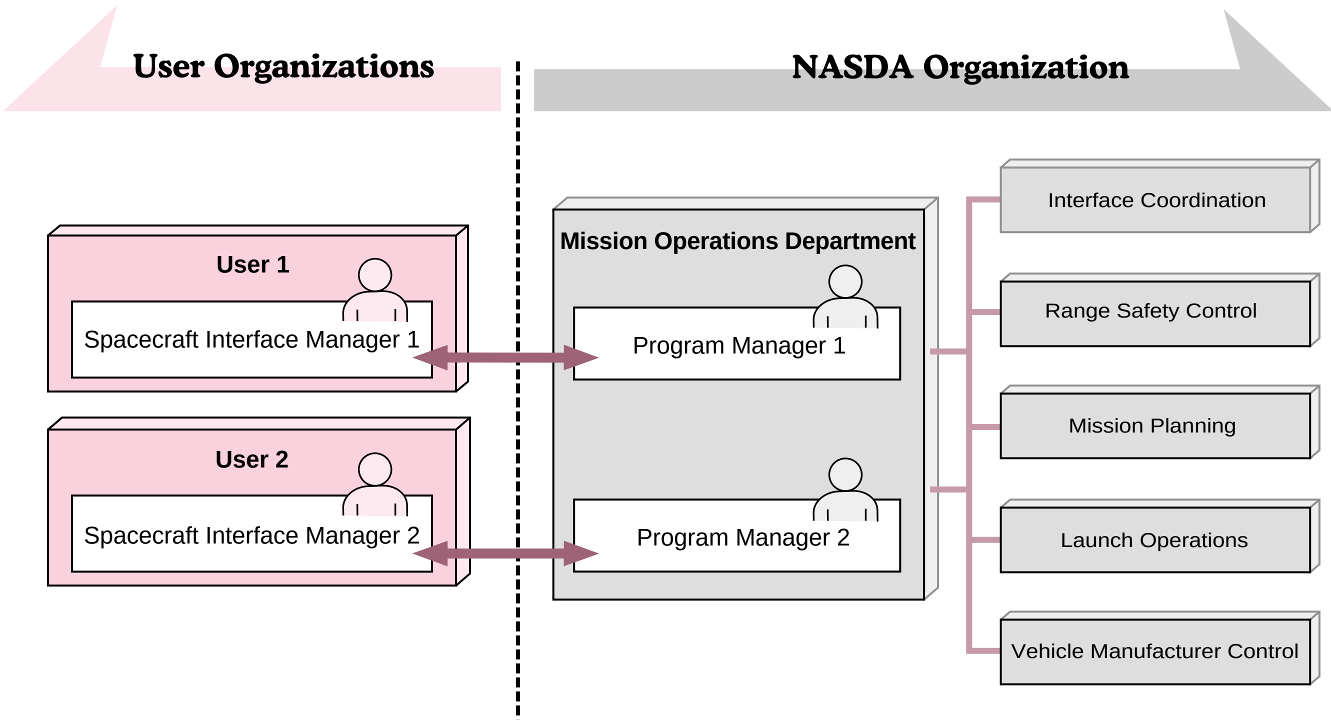
Figure 1.2.12 Payload fairing for single launch



<Note>
*1 : This value shows a nominal interface diameter. (See Appendix 3)

Unit : mm

Figure 1.2.13 Payload adapters (example)



| A

Figure 1.2.14 Concept of users / NASDA relationship | C

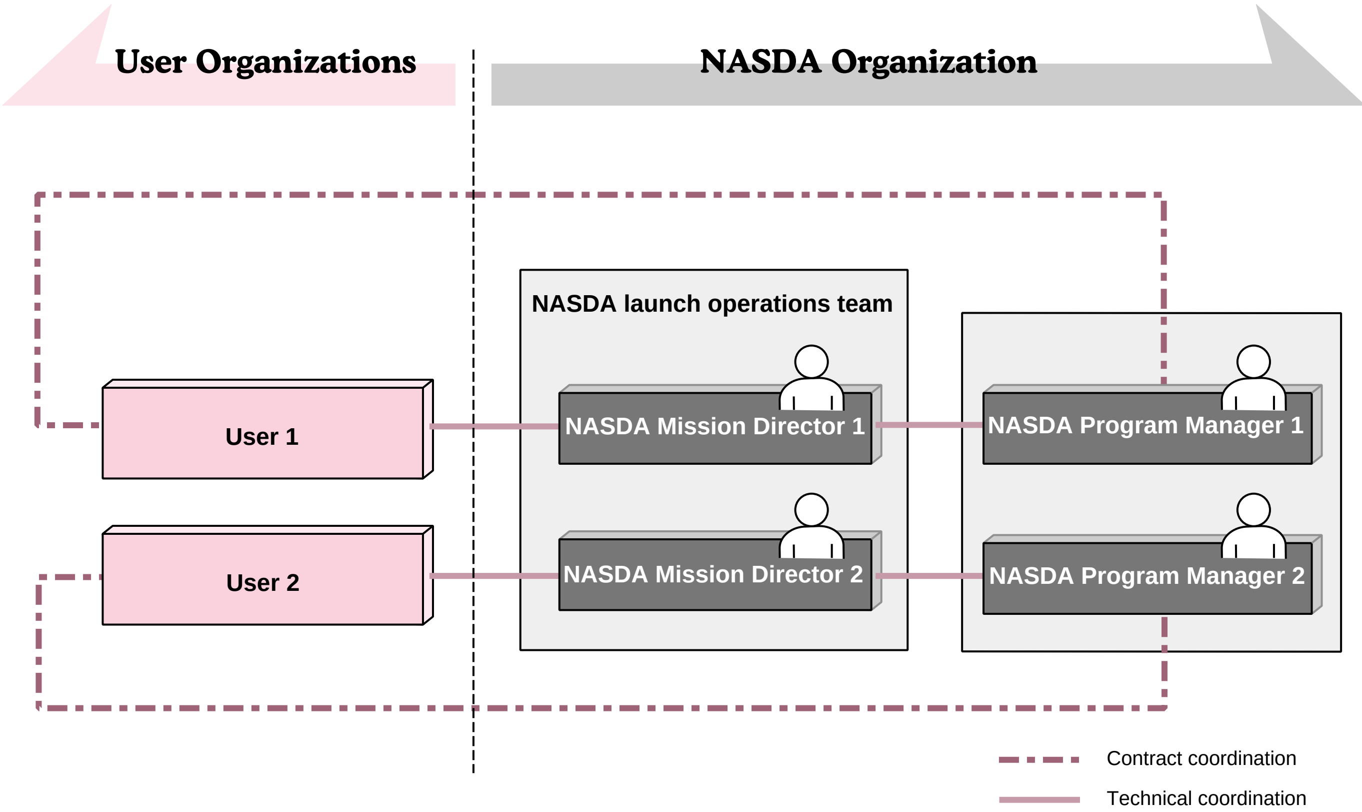


Figure 1.2.15 Concept of users / NASDA relationship after establishment of NASDA launch operations team

| C

1.3 H-IIA Launch System Related Documents

NASDA provides two more documents related to the H-IIA launch system besides this user's manual. These documents provide users with detailed information on launch facilities and range safety requirements as a reference for the preliminary planning phase. They are:

a) H-IIA Payload-Related Facilities and Ground Support Equipment (GSE) Manual
b) Launch Vehicle Payload Safety Requirements (NASDA-STD-14B) | A

1.3.1 H-IIA Payload-Related Facilities and GSE Manual (in preparation)

The H-IIA Payload-Related Facilities and GSE Manual provides information about TNSC and launch preparation operations of spacecraft.

Appendix 2 of this user's manual describes launch facilities and related launch preparation operations briefly. Detailed information is provided in the H-IIA Payload-Related Facilities and GSE Manual which includes a number of pictures and drawings of buildings and equipments.

1.3.2 Launch Vehicle Payload Safety Requirements

This document provides the requirements for safety control, spacecraft safety design, and launch site operations at TNSC. A description related to safety reviews is provided in § 6.5.

1.4 Definition of terms

The terms used in this manual are generally defined in the body of the text. The following terms are defined below: | A

(1) User

An organization or an individual who entrusts or wishes to entrust a launch of the spacecraft to NASDA.

(2) Spacecraft organization

The user or the spacecraft builder to whom the user has entrusted the fabrication, testing and spacecraft launch operations.

In case of a NASDA spacecraft, the spacecraft organization means the NASDA spacecraft organization and the term of NASDA corresponds to the launch vehicle organization.

(3) NASDA launch operations team

A provisional team organized by NASDA to launch the launch vehicle.

(4) Launch vehicle organization

A group within the NASDA launch operations team in charge of launch operations for the launch vehicle.

(5) Launch operations

This is a generic term referring to the assembly, preparation, testing and joint operations of the spacecraft and launch vehicle organizations which are to be implemented at TNSC.

(6) Program manager

A NASDA person responsible for contracts and interface coordination of the launch; a member of the "Mission Operations Department (MOD)." A

(7) Spacecraft interface manager

A user person responsible for the launch activity coordination.

In case of a NASDA spacecraft launch, the term refers to the manager of the spacecraft organization or his representative.

(8) NASDA mission director

A NASDA staff member responsible for the interface and coordination of technical matters related to launch operations (from spacecraft arrival at TNSC through completion of the post lift-off operation) after establishment of the NASDA launch operations team.

(9) Launch vehicle system

A system including the launch vehicle, related facilities, and Aerospace Ground Equipment (AGE).

(10) Spacecraft system

A system including the spacecraft, its related facilities, and GSE.

(11) Y ± and X ±

"Y - (numerical figure)" represents the number of days before the launch date in terms of actual working days.

"Y + (numerical figure)" represents the number of days after the launch date in terms of actual working days.

"Y - 0" represents the lift-off date.

"X - (numerical figure)" represents the time before lift-off (holding (or margin) time not included).

CHAPTER 2.

MISSION PERFORMANCE

CHAPTER 2. MISSION PERFORMANCE

2.1 General

This chapter describes the mission performance of the H-IIA launch vehicle.

The H-IIA launch vehicle provides a wide variety of mission performances. However, this manual describes only the following missions.

a) Geostationary transfer orbit (GTO) mission
b) Sun-synchronous orbit (SSO) mission
c) Low Earth orbit (LEO) mission
d) Earth escape mission

Table 2.1.1 illustrates representative performance capabilities of the H-IIA family. The performance data are based on standard mission modifications of payload fairing (refer to § 4.4.3). Actual launch capability will vary with final configurations. A

2.1.1 Mission profile

To provide users with H-IIA mission sequences, typical mission profile and sequences are briefly explained using a GTO mission for example in the following paragraphs. Figures 2.1.1 and 2.1.2 show flight profiles for typical GTO missions in case of H2A2024 and H2A212 respectively. Table 2.1.2 shows typical GTO mission sequence of events for each H-IIA vehicle. In case of other missions such as SSO, LEO, etc., sequence time of events and flight trajectories are different from a GTO mission, but most of these sequences are held similarly to a GTO mission. These data are representative and actual sequence and profile will be prepared to meet spacecraft requirements. As shown in Table 2.1.2, although sequence of events for each vehicle configuration are the same, the sequence time is different respectively. So, in the following paragraphs, sequence time for H2A2024 and H2A212 is used representatively according to Figures 2.1.1 and 2.1.2.

2.1.1.1 Booster and first stage phase

The main engine LE-7A is ignited at about 4.7 seconds before lift-off (X-0). In case of H2A212, two more LE-7A engines of the LRB are ignited at the same time.

After detecting the rise of the combustion pressure, two SRB-As are ignited (at about 0.5 second before X-0), subsequently the H-IIA vehicle rises away from the ML and the guidance control program on the guidance control computer of the second stage (GCC2) senses the lift-off (that is X-0). About 100 seconds after lift-off (hereafter called X+100), two SRB-As burn out and are separated from the core stage. A

In case of H2A2024, the first pair of SSBs is ignited at about X+10, burns out at about X+70 and is separated from the core stage at about X+111. On the other hand, the second pair of SSBs is ignited at about X+76, burns out at about X+136 and is also separated from the core stage at about X+142. A

In case of H2A212, the LRB will experience a propellant depletion at about X+197 and its main engines are cut off at intervals of a several seconds. And then the LRB is separated from the core stage. The payload fairing (PLF) is jettisoned after a free molecular heat flux becomes less than 1135 W/m^2, resulting the jettison timing of the PLF being different in each mission. At about X+389, the main engine shutdown command (MECOM) is sent from the guidance control computer of the first stage (GCC1). After the main engine has tailed off, the first and second stage separation is executed.

The attitude control is conducted by the engine gimbaling of the core stage, the SRB-As and the LRB, and also auxiliary thrusters. The gimbaling of the core main 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. The LRB engines gimbaling participates in the pitch / yaw control during all phase of the LRB, but in roll control, these engines contribute only after the SRB-A tail off. Auxiliary thrusters roll the H-IIA vehicle throughout the booster and first stage phase.

2.1.1.2 Second stage phase

After the first and the second stage separation, the LE-5B is ignited by the first ignition command from the GCC2 (SEIG1). The LE-5B engine burns about 300 (or 220) seconds. As soon as the second stage (including the payload) is injected to the parking orbit, the engine is cut off by engine cutoff command from the GCC2 (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 and so on. In this phase, the attitude control of the pitch/yaw/roll is performed only by the RCS.

About 710 (or 630) seconds later, the LE-5B is restarted (SEIG2) and when the second stage reaches the planned transfer orbit, the engine cutoff command (SECOM2) is sent from the GCC2. The second burning duration is approximately 210 (or 280) seconds in a normal GTO mission.

After spacecraft / second stage separation, using separation springs or GHe retro motor the collision avoidance maneuver (CAM, that is the collision and contamination avoidance maneuver) is conducted using the RCS and the residual GH_2 in the LH_2 tank.

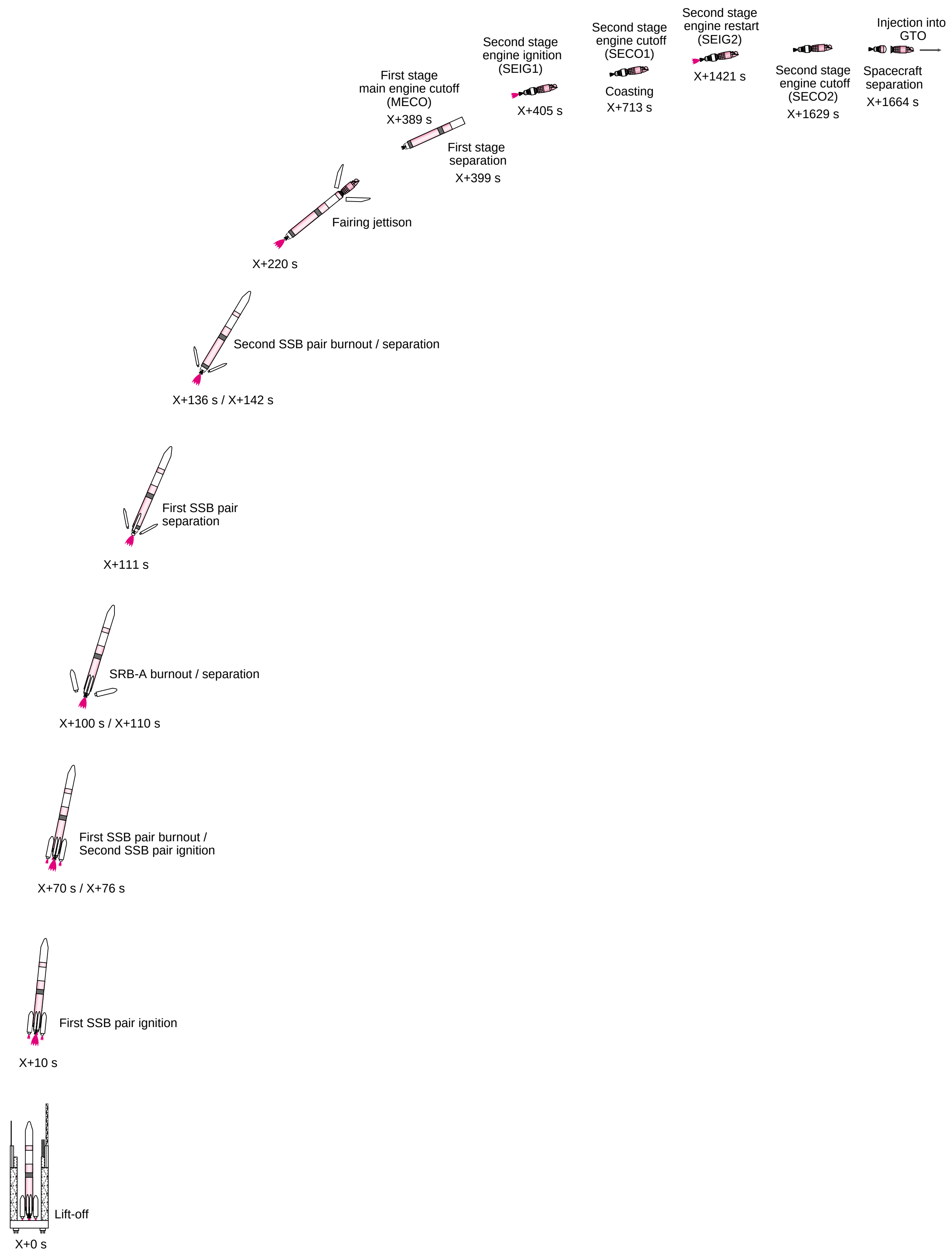


Figure 2.1.1 Typical GTO mission profile for H2A2024

| A

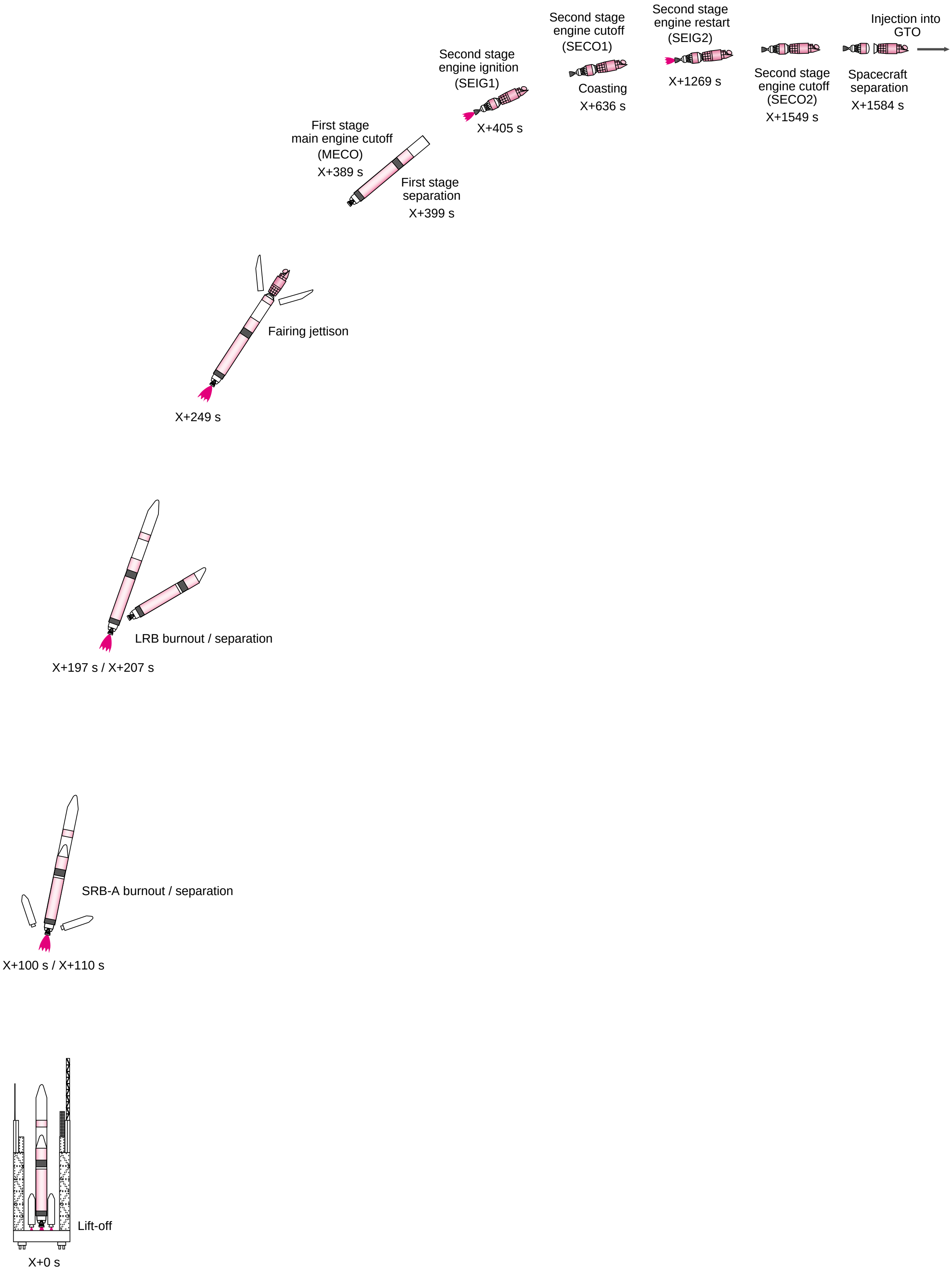


Figure 2.1.2 Typical GTO mission profile for H2A212 | A

Table 2.1.1 Summary of H-IIA Payload capability

Mission (Orbit)	Payload capability ; kg (lb) <With the LE-7A lower nozzle skirt>				Note
	H2A202	H2A2022	H2A2024	H2A212	
GTO ha = 36,226 km (*1) hp = 250 km i = 28.5 deg ω = 179.0 deg	4,100 (9,039) (4S fairing)	4,500 (9,921) (4S fairing)	5,000 (11,023) (4S fairing)	7,500 (16,535) (5S fairing)	Osculating orbit at spacecraft separation *1 : Including kepler bias (190 km) and apogee bias (250 km)
SSO h = 800 km i = 98.6 deg	3,600 (in Summer) (7,937) 4,400 (except Summer) (9,700) (5S fairing)	—	—	—	

Mission (Orbit)	Payload capability ; kg (lb) <Without the LE-7A lower nozzle skirt>				Note
	H2A202	H2A2022	H2A2024	H2A212	
GTO ha = 36,226 km (*1) hp = 250 km i = 28.5 deg ω = 179.0 deg	3,700 (8,157) (4S fairing)	4,100 (9,039) (4S fairing)	46,00 (10,141) (4S fairing)	—	Osculating orbit at spacecraft separation *1 : Including kepler bias (190 km) and apogee bias (250 km)
SSO h = 800 km i = 98.6 deg	3,500 (in Summer) (7,716) 4,200 (except Summer) (9,259) (5S fairing)	—	—	—	

<Abbreviation>
- GTO : geostationary transfer orbit
- SSO : sun-synchronous orbit
- LEO : low Earth 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 6378.142 km.
The mass of the payload adapter is assumed 100 kg including the separation system.

Development of configuration "with the LE-7A lower nozzle skirt" is to be completed in 2005 or later.

Table 2.1.2 Typical sequence of events of H-IIA vehicle family for GTO mission | A

Events (seconds)	H2A202 *1	H2A2022 *1	H2A2024 *1	H2A212 *2	Remarks
Guidance flight mode on	-13.0	-13.0	-13.0	-13.0	
LE-7A (pair) ignition	-6.0	-6.0	-6.0	-6.0	
SRB-A (pair) ignition	-0.6	-0.6	-0.6	-0.6	
Liftoff	0.0	0.0	0.0	0.0	
First SSB pair ignition	N / A	50.0	10.0	N / A	
SRB-A burn out	100.0	100.0	100.0	100.0	
First SSB pair burn out	N / A	109.8	69.8	N / A	
SRB-A separation	110.0	110.0	110.0	110.0	
Second SSB pair ignition	N / A	N / A	76.0	N / A	
First SSB pair separation	N / A	121.0	111.0	N / A	
Second SSB pair burn out	N / A	N / A	135.8	N / A	
Second SSB pair separation	N / A	N / A	142.0	N / A	
LRB main engine cutoff (LMECO)	N / A	N / A	N / A	197.1	
LRB separation	N / A	N / A	N / A	207.1	
Fairing jettison	263.8	275.0	220.2	249.0	
Main engine cutoff (MECOM)	389.3	389.3	389.3	389.3	
First / Second stage separation	399.3	399.3	399.3	399.3	
Second stage ignition 1 (SEIG1)	405.3	405.3	405.3	405.3	
Second stage cutoff 1 (SECO1)	732.0	722.1	713.3	636.0	
Second stage ignition 2 (SEIG2)	1466.2	1445.1	1420.8	1268.8	
Second stage cutoff 2 (SECO2)	1651.7	1639.0	1628.6	1549.1	
Spacecraft separation	1686.7	1674.0	1663.6	1584.1	

A

A

A

<Note>
***1 : with 4S fairing**
***2 : with 5S fairing**

2.2 Performance Ground Rules

H-IIA performance ground rules for various missions are described in this section.

2.2.1 Payload mass definition

A

Performance capabilities referred to throughout this document are presented in terms of payload mass. Payload mass is defined as follows:

A

Payload mass is the total mass of the spacecraft injected to the target orbit, excluding the payload adapter (PLA) whose mass is assumed 100 kg including the separation system. This PLA is a standard type for the H-IIA vehicle and named 1194M. If a different type of the PLA is used or other hardware is required, mass difference between 1194M PLA and other PLA or hardware should be considered in performance capabilities. Information concerning the PLA mass appears in the § 4.5.

A

2.2.2 Launch vehicle configurations

Typical H-IIA performance presented in this document is based on using the 4S fairing unless noted otherwise. If user requires greater volume than the 4S fairing for a spacecraft or dual launch, the H-IIA can also offer other types of a fairing such as the 5S, the 4/4D-LS, the 4/4D-LC, the 5/4D and the 5S-H fairing. But in case of using these fairings, performance will degrade mainly according to the fairing mass. Information related to the payload fairing mass is given in the § 4.4.

C

2.2.3 Launch vehicle performance confidence levels

The H-IIA launch system is designed with 99.7 % performance confidence level to meet the requirements of each user with flexibility. Performance confidence levels can be set based on each mission's requirements.

2.3 Geostationary Transfer Orbit (GTO) Mission

2.3.1 Payload capability for single launch

Payload mass for GTO mission using 4S fairing , with the LE-7A lower nozzle skirt is about 4,100kg (not including standard payload adapter) based on parameters in § 2.3.3.

C

And payload mass for GTO mission using 4S fairing , without the LE-7A lower nozzle skirt is about 3,700kg (not including standard payload adapter) based on parameters in § 2.3.3.

Payload mass using model 5S fairing is about 350 kg less, with the LE-7A lower nozzle skirt.

And payload mass using model 5S fairing is about 300 kg less, without the LE-7A lower nozzle skirt.

Figure2.3.9 ~ 2.3.12 show payload capability of H2A202, H2A2022 and H2A2024 / 4S fairing, H2A212 / 5S for GTO mission for configuration with the LE-7A lower nozzle skirt .

Figure2.3.13~2.3.15 show payload capability of H2A202, H2A2022 and H2A2024 / 4S fairing for GTO mission for configuration without the LE-7A lower nozzle skirt.

2.3.2 Payload capability for dual (GTO and GTO) launch

The combined mass of two spacecraft, upper payload adapter and the lower portion fairing equals the value of single spacecraft launch mass.

2.3.3 Typical orbital parameters

Typical orbital parameters for GTO mission are as follows :

Apogee altitude	ha	=	36,226 km *1
Perigee altitude	hp	=	250 km
Inclination	i	=	28.5°
Argument of perigee	ω	=	179.0°

Assuming the Earth's equator radius is 6,378.142 km and these values are osculating orbit parameters at spacecraft separation.

<Note>

*1 : This value includes Kepler bias (190 km) and apogee bias (250 km).

2.3.4 Injection accuracies

Typical injection accuracies for GTO mission are as follows based on parameters in § 2.3.3.

Apogee altitude	Δ ha	$= \pm$ 180 km
Perigee altitude	Δ hp	$= \pm$ 4 km
Inclination	Δ i	$= \pm$ 0.02°
Argument of perigee	Δ ω	$= \pm$ 0.40°
Longitudinal of ascend node	Δ Ω	$= \pm$ 0.40°

(These values are 3-sigma level.) A

2.3.5 Typical sequence of events

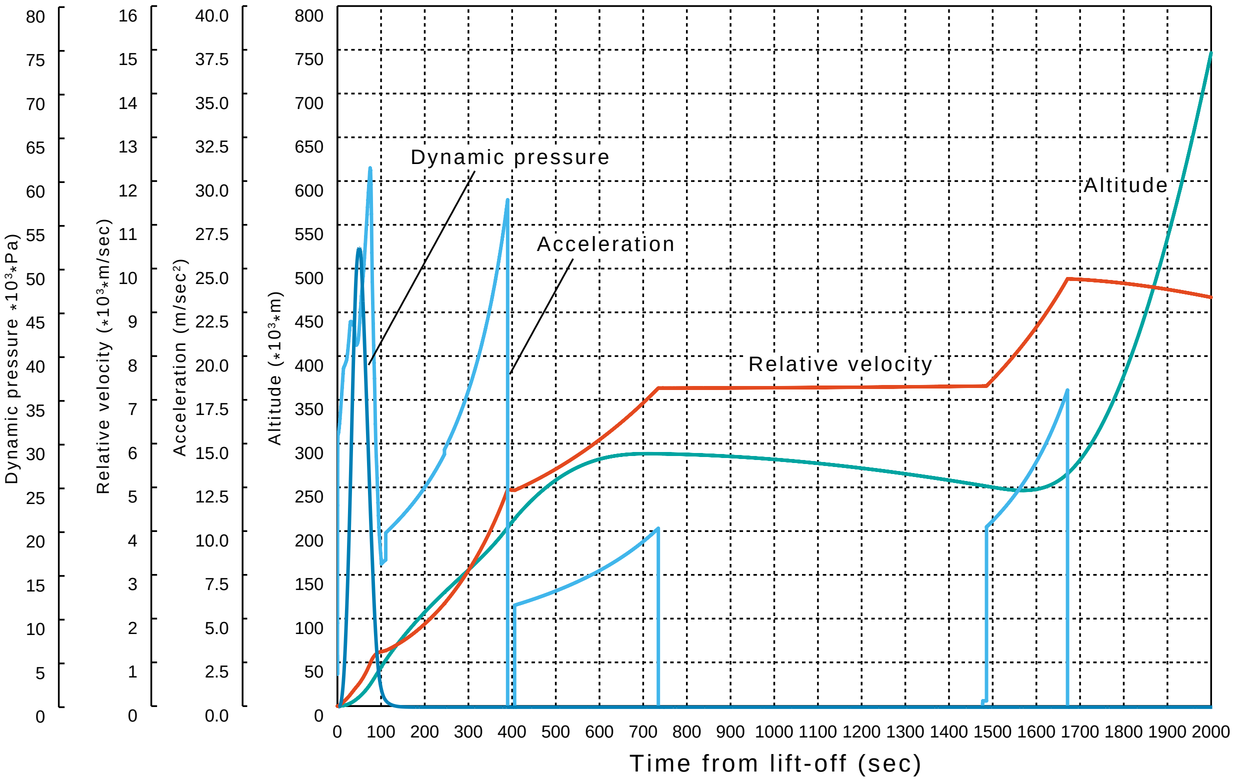
Table 2.1.2 shows a typical sequence of events for GTO mission in the case of H2A202, H2A2022 and H2A2024 with the 4S fairing. A

2.3.6 Typical trajectory

Figures 2.3.2, 2.3.4, 2.3.6, 2.3.8 show a typical flight trajectory for GTO mission in the case of H2A202, H2A2022 and H2A2024 with the 4S fairing, H2A212 with the 5S fairing respectively. A

2.3.7 Typical flight parameters

Figures 2.3.1, 2.3.3, 2.3.5, 2.3.7 show a typical flight parameters for GTO mission in the case of H2A202, H2A2022 and H2A2024 with the 4S fairing, H2A212 with the 5S fairing respectively. A



A

Figure 2.3.1 Typical flight parameters for GTO mission (H2A202 with 4S fairing)

A

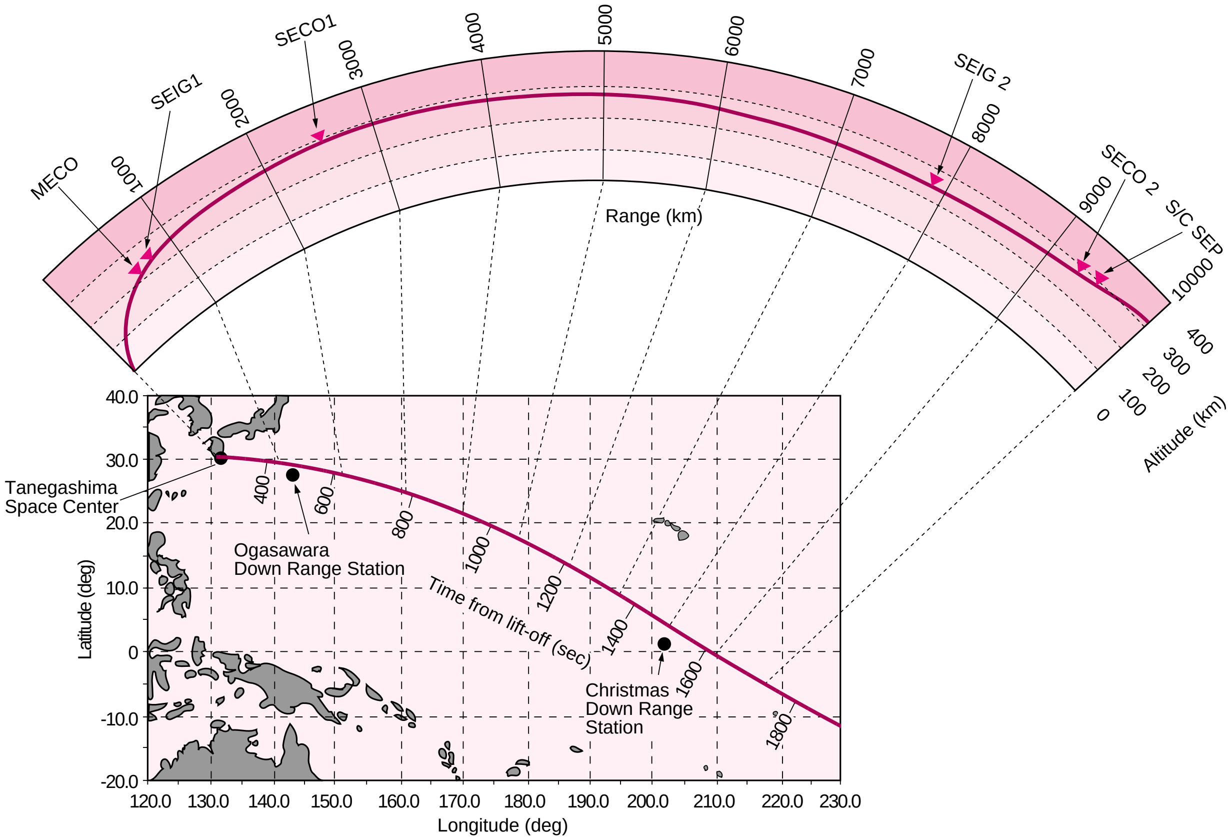


Figure 2.3.2 Typical flight trajectory for GTO mission (H2A202 with 4S fairing)

A

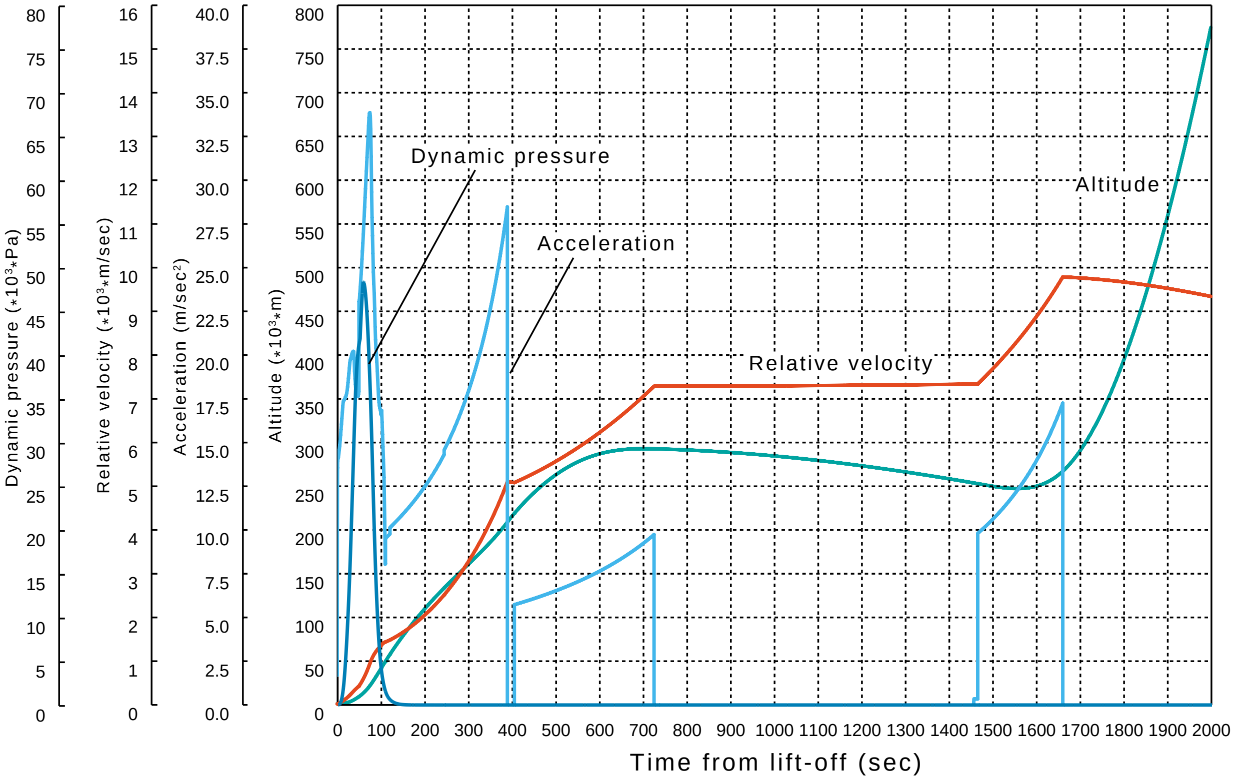


Figure 2.3.3 Typical flight parameters for GTO mission (H2A2022 with 4S fairing)

A

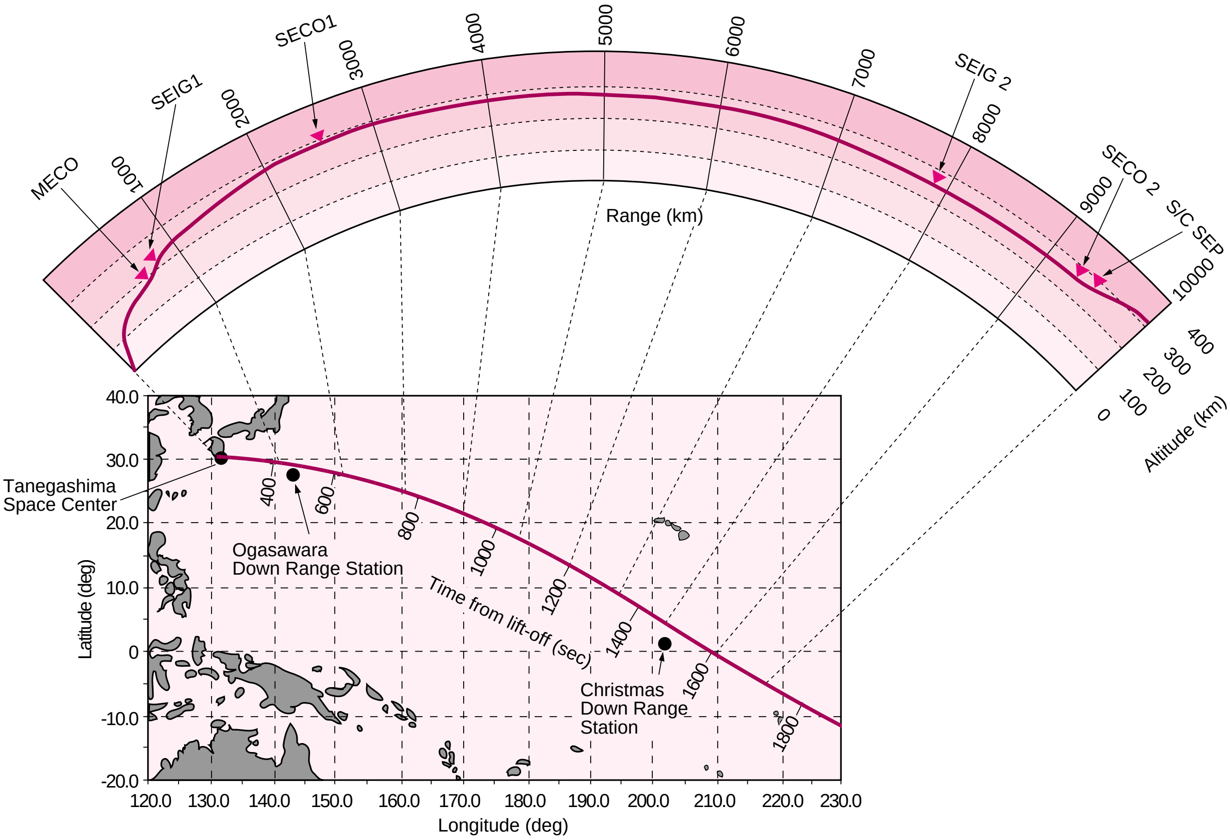


Figure 2.3.4 Typical flight trajectory for GTO mission (H2A2022 with 4S fairing)

A

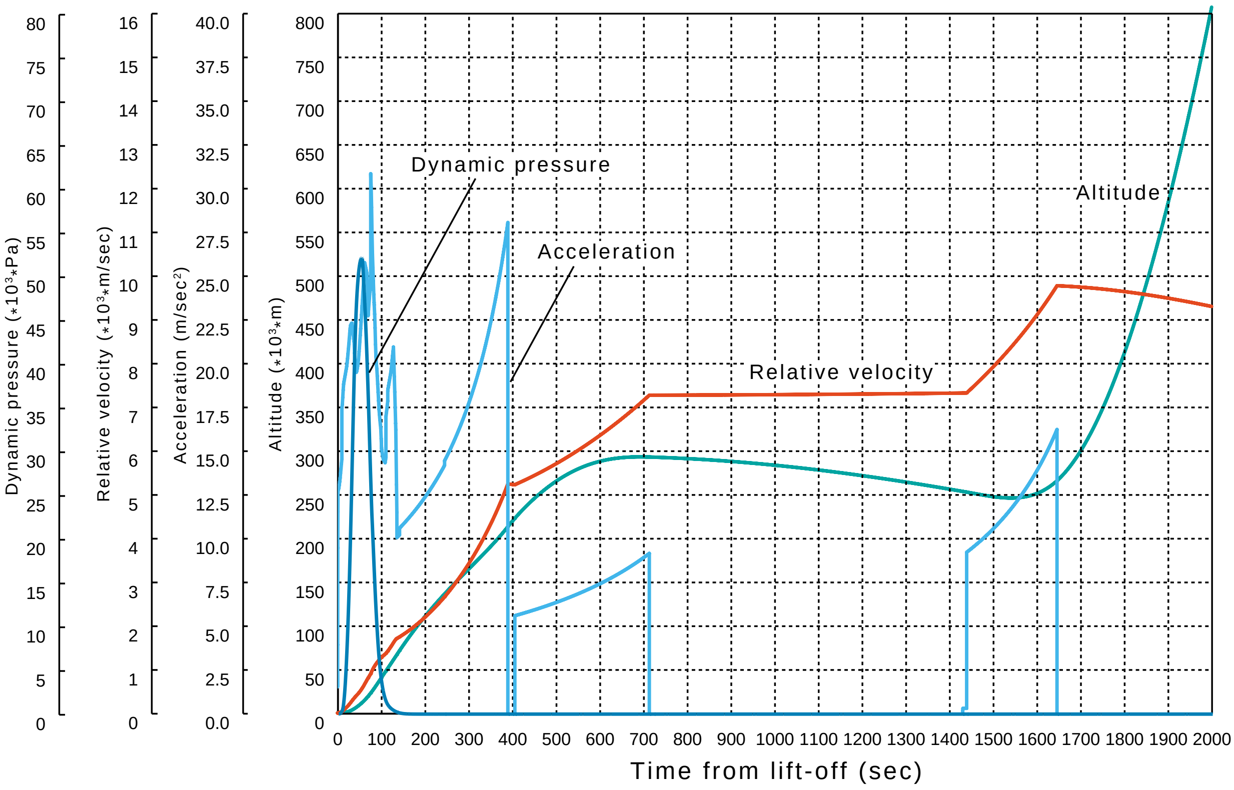


Figure 2.3.5 Typical flight parameters for GTO mission (H2A2024 with 4S fairing)

A

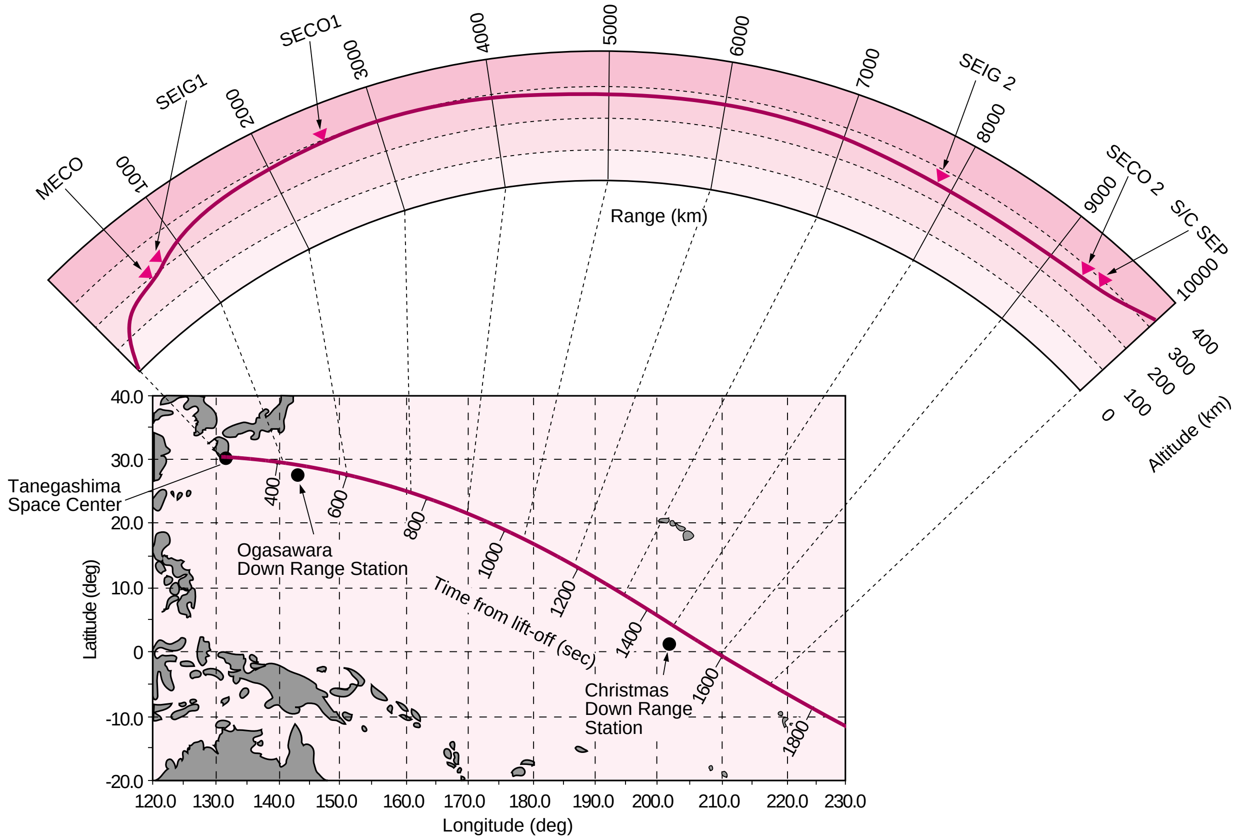
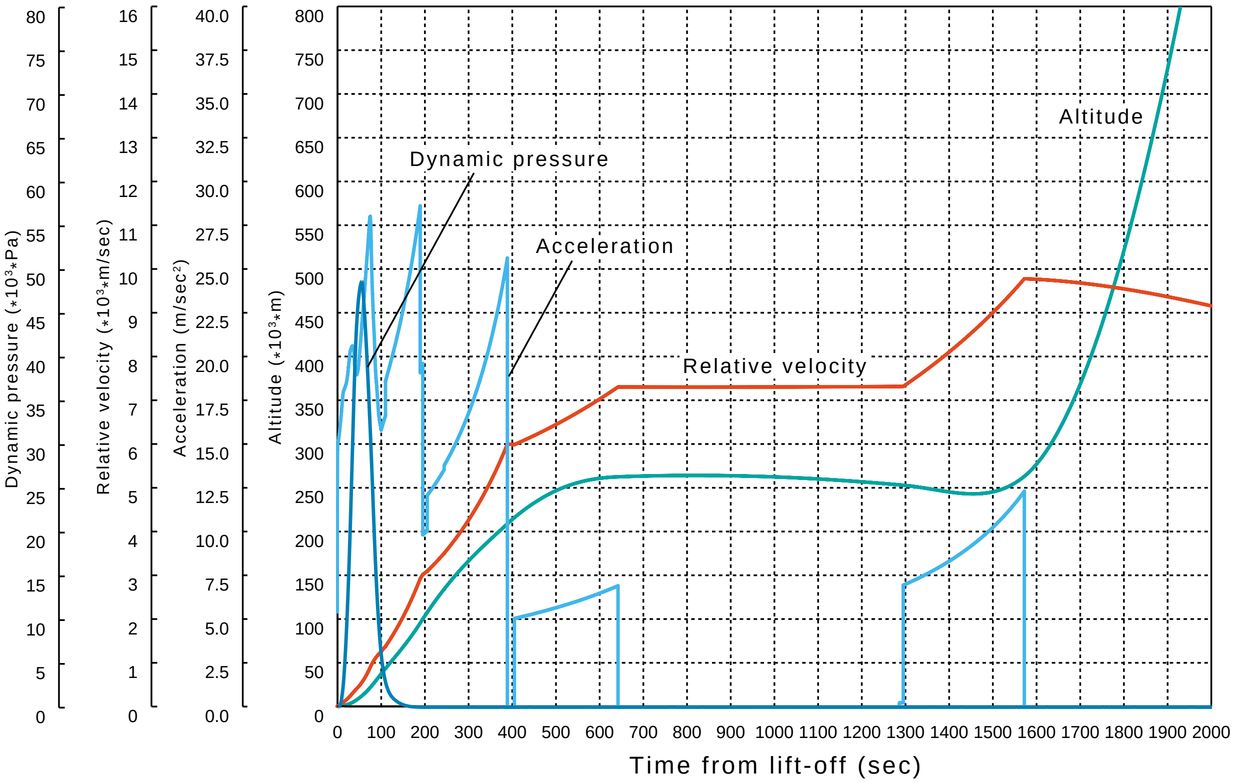


Figure 2.3.6 Typical flight trajectory for GTO mission (H2A2024 with 4S fairing)

A



A

Figure 2.3.7 Typical flight parameters for GTO mission (H2A212 with 5S fairing)

A

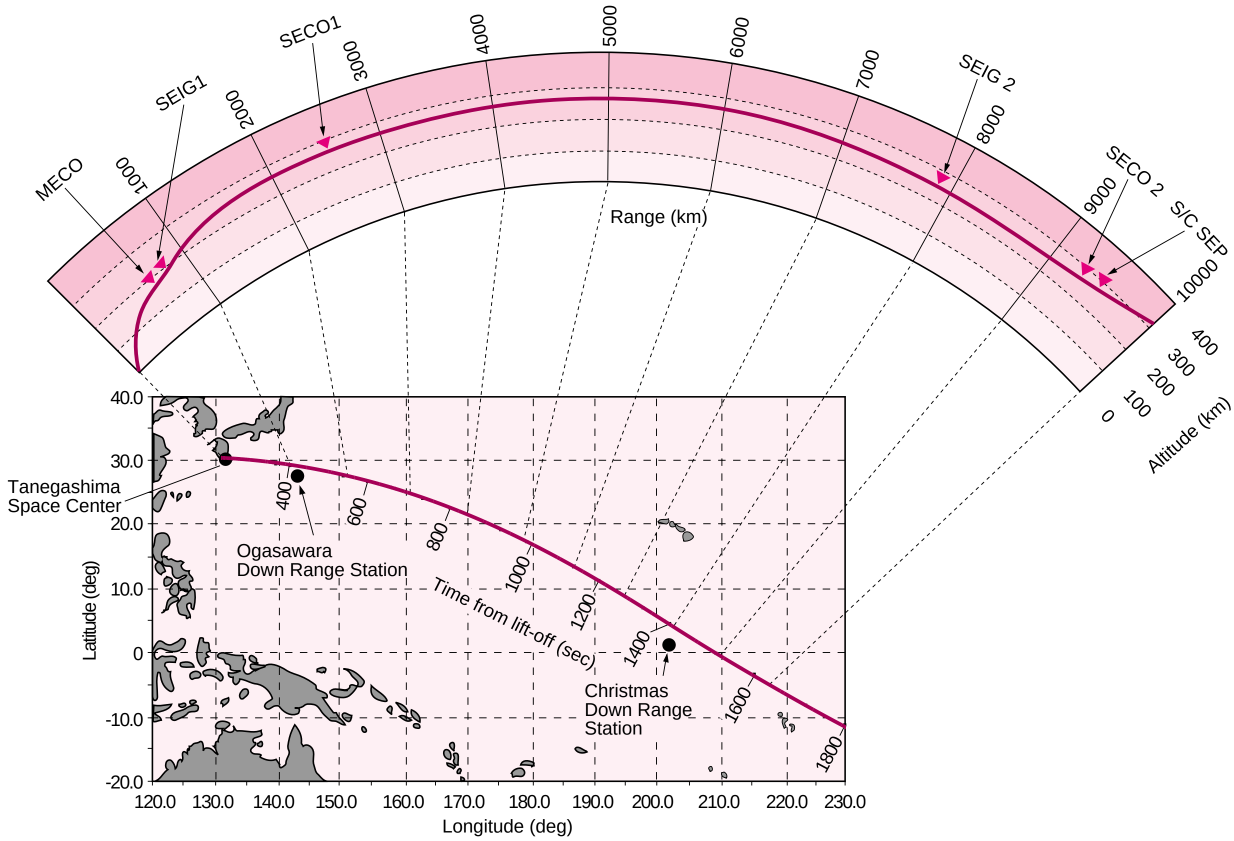


Figure 2.3.8 Flight trajectory for GTO mission (H2A212 with 5S fairing)

A

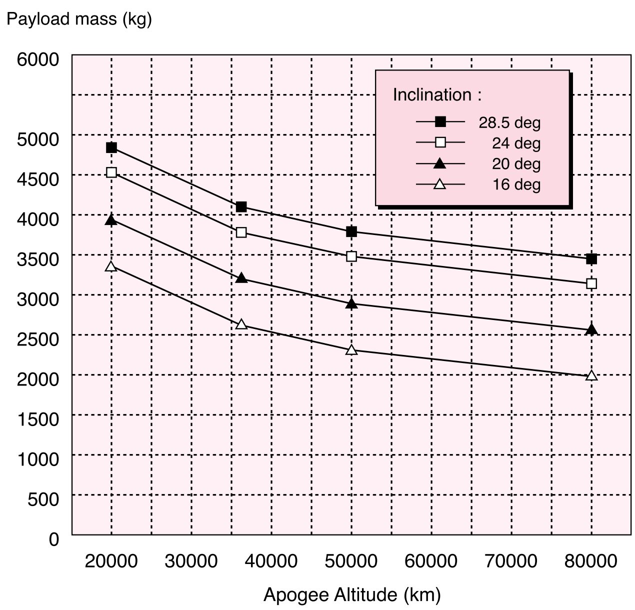


Figure 2.3.9 Payload capability for GTO mission (H2A202 with 4S fairing)

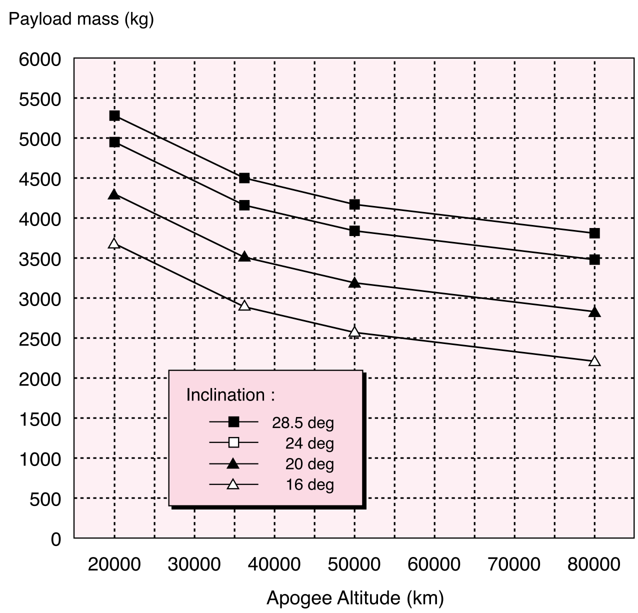


Figure 2.3.10 Payload capability for GTO mission (H2A2022 with 4S fairing)

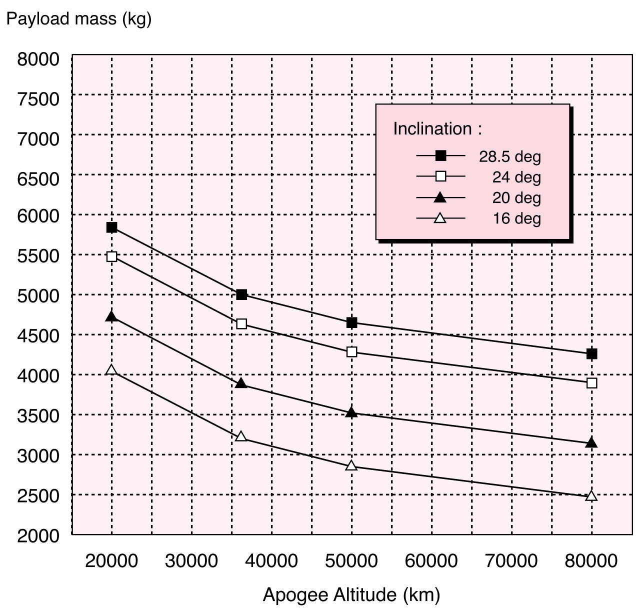


Figure 2.3.11 Payload capability for GTO mission (H2A2024 with 4S fairing)

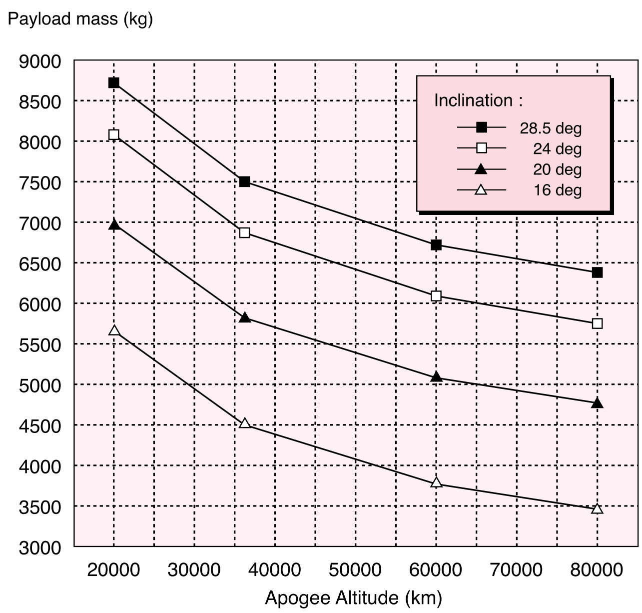


Figure 2.3.12 Payload capability for GTO mission (H2A212 with 5S fairing)

<With the LE-7A lower nozzle skirt>

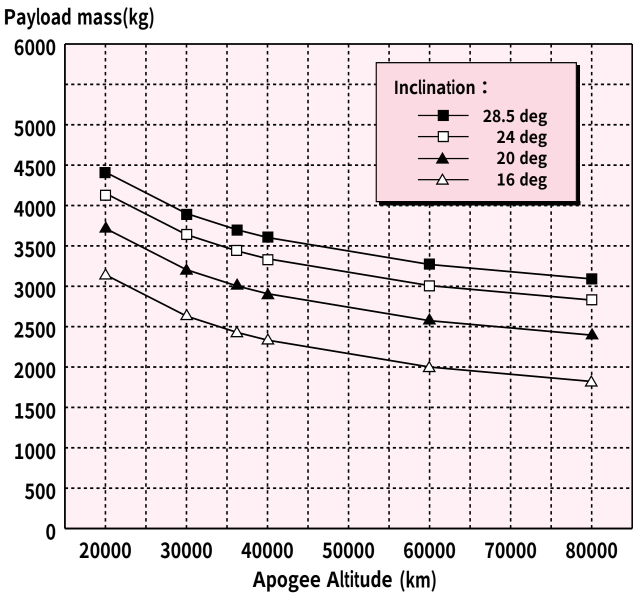


Figure 2.3.13 Payload capability for GTO mission (H2A202 with 4S fairing)

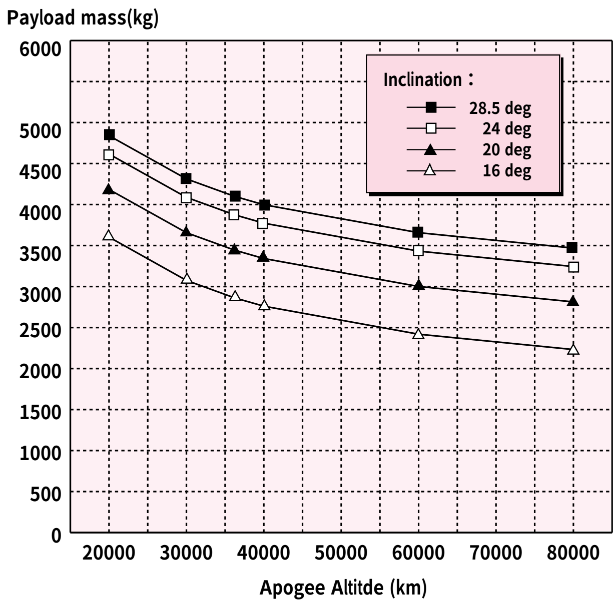


Figure 2.3.14 Payload capability for GTO mission (H2A2022 with 4S fairing)

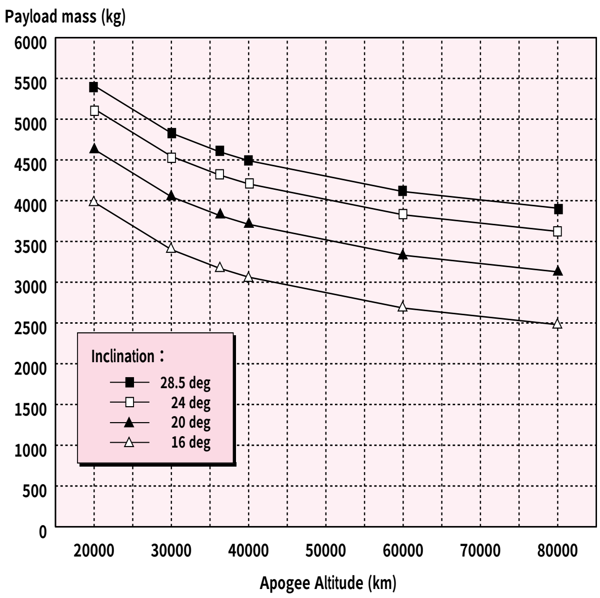


Figure 2.3.15 Payload capability for GTO mission (H2A2024 with 4S fairing)

<Without the LE-7A lower nozzle skirt>

C

2.4 Sun-Synchronous Orbit (SSO) Mission

2.4.1 Payload capability

Payload mass for SSO mission using 5S fairing, with the LE-7A lower nozzle skirt is about 4,400 kg (except summer) based on parameters in §2.4.2. C

And payload mass for SSO mission using 5S fairing, without the LE-7A lower nozzle skirt is about 4,200kg (except summer) based on parameters in §2.4.2.

Payload mass using model 4S fairing is about 300 kg greater, with the LE-7A lower nozzle skirt.

Payload mass using model 4S fairing is about 250 kg greater, without the LE-7A lower nozzle skirt.

Figure 2.4.1 shows payload capability of H2A202 / 5S fairing for SSO mission for both with and without the LE-7A lower nozzle skirt.

2.4.2 Typical orbital parameters

Typical orbital parameters for SSO mission are as follows : A

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

B

2.4.3 Injection accuracies

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

Semi-major axis	Δh	$= \pm$	10.0 km
Inclination	Δi	$= \pm$	0.18 °
Eccentricity	ΔE	=	0 ~ 0.001

B

(These values are 3-sigma level.) A

2.4.4 Typical trajectory

Figure 2.4.2 shows a typical flight trajectory for SSO missionin the case of H2A202 with 5S fairing. A B A

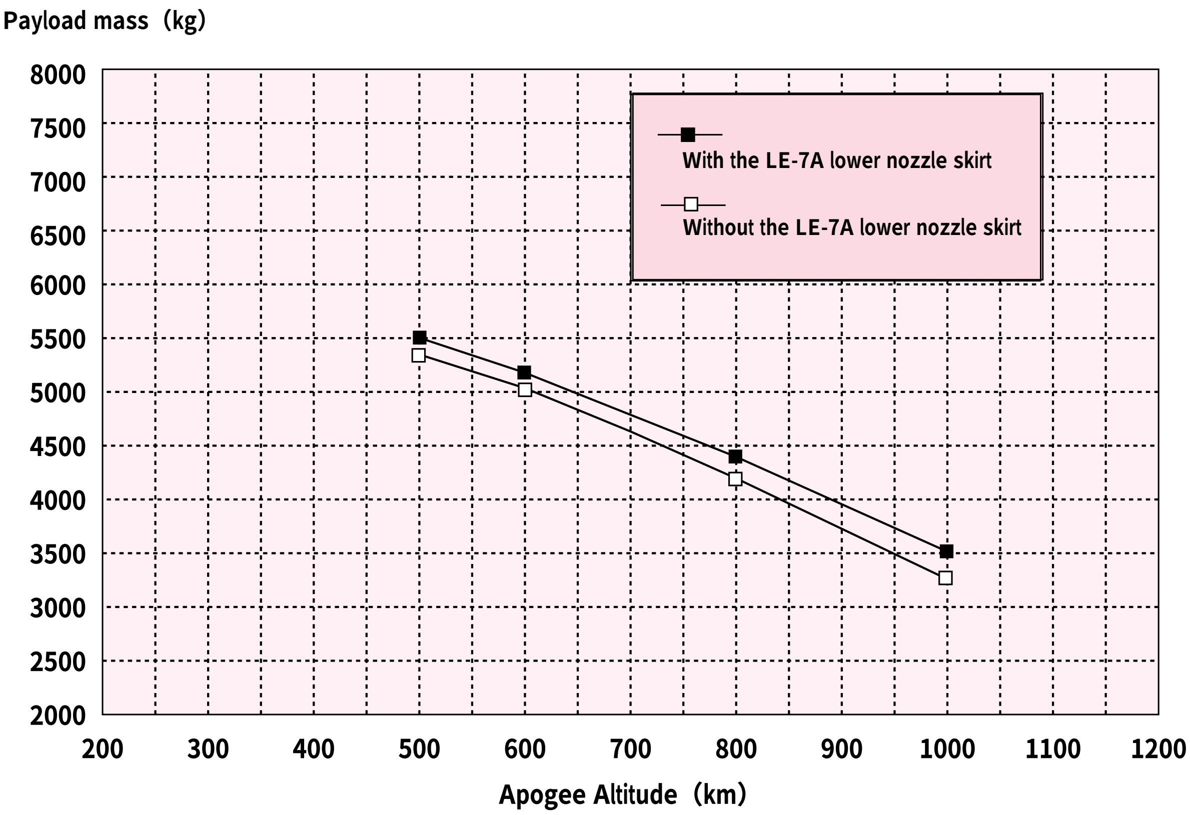


Figure 2.4.1 Payload capability for SSO mission (H2A202 with 5S fairing except summer)

C

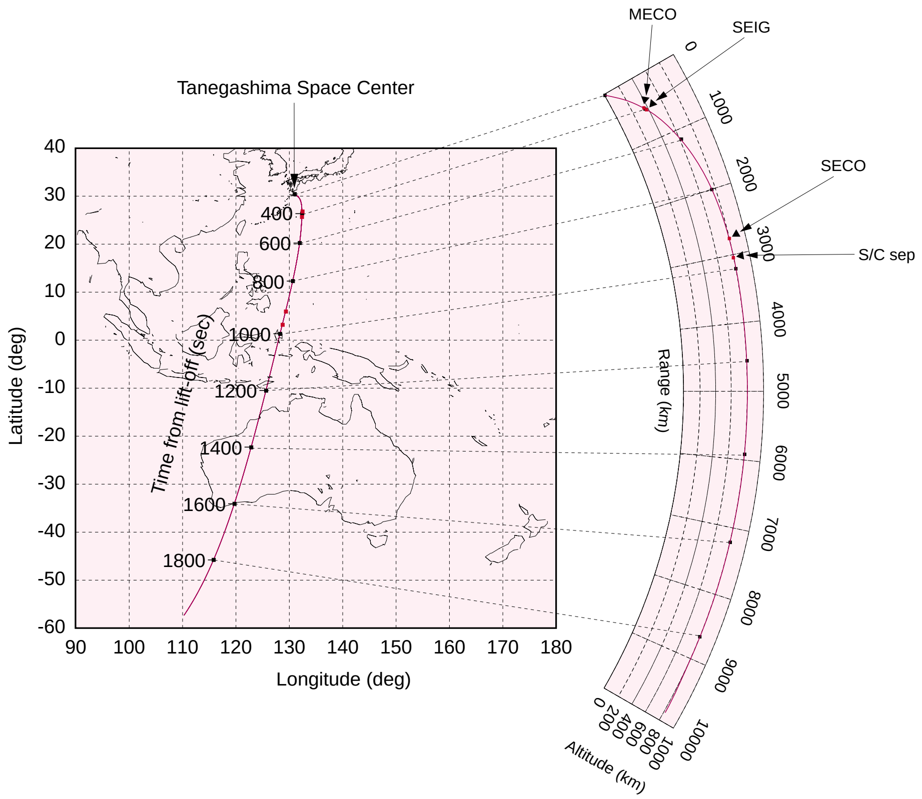


Figure 2.4.2 Typical flight trajectory for SSO mission (H2A202 with 5S fairing)

2.5 Low Earth Orbit (LEO) Mission

2.5.1 Payload capability

Figures 2.5.1 and 2.5.2 show payload capability for LEO mission using model 5S fairing for two inclinations in the case of configuration " with the LE-7A lower nozzle skirt". C

Figures2.5.3 and 2.5.4 show payload capability for LEO mission using model 5S fairing for two inclinations in the case of configuration "without the LE-7A lower nozzle skirt".

Figure 2.5.1 : Payload capability for LEO mission (H2A202 with 5S fairing) (inclination 30.4 deg) With the LE-7A lower nozzle skirt
Figure 2.5.2 : Payload capability for LEO mission (H2A202 with 5S fairing) (inclination 51.6 deg) With the LE-7A lower nozzle skirt
Figure 2.5.1 : Payload capability for LEO mission (H2A202 with 5S fairing) (inclination 30.4 deg) Without the LE-7A lower nozzle skirt
Figure 2.5.2 : Payload capability for LEO mission (H2A202 with 5S fairing) (inclination 51.6 deg) Without the LE-7A lower nozzle skirt

Payload mass using model 4S fairing is greater than using model 5S fairing but the increase of payload mass is depends on the injection orbit.

B

A

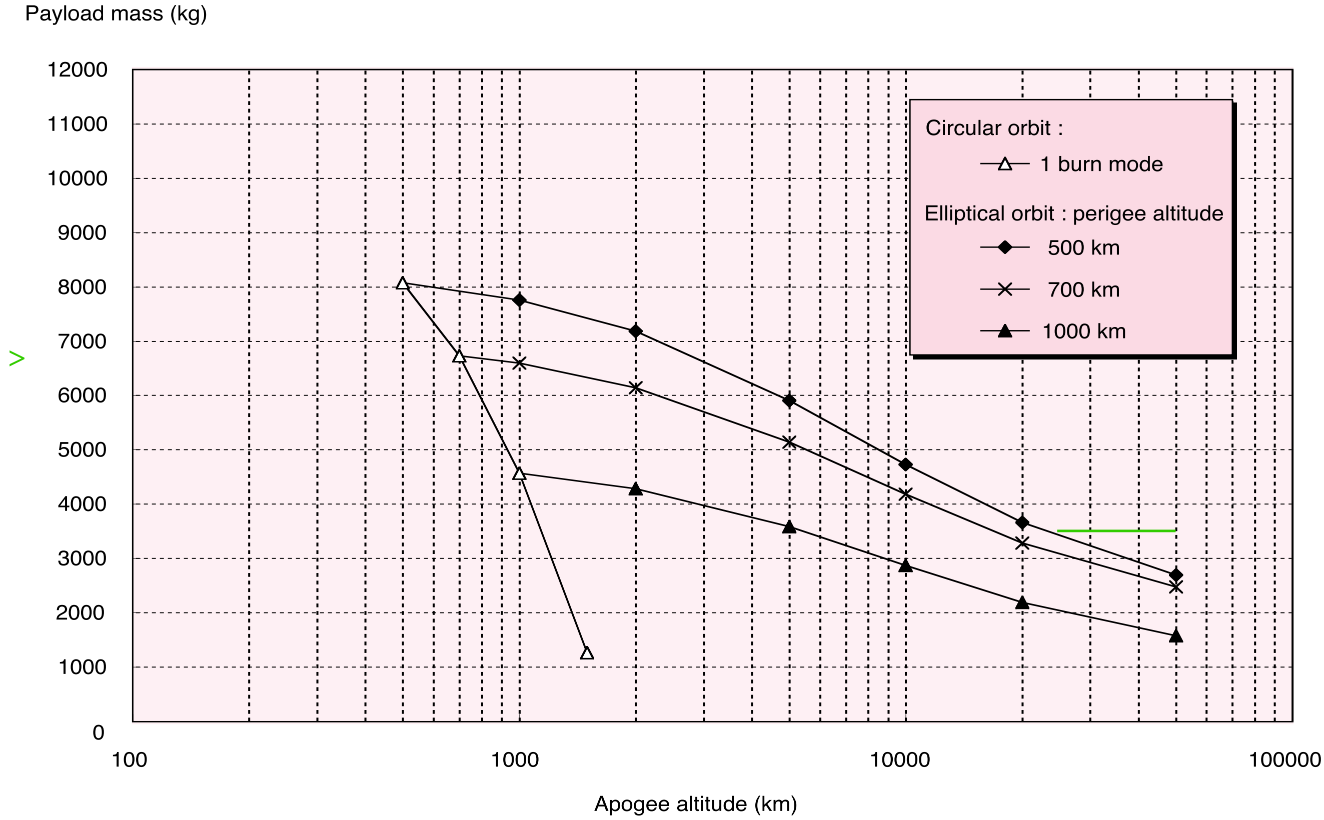


Figure 2.5.1 Payload capability for LEO mission (H2A202 with 5S fairing)
(inclination 30.4 deg)

B

A

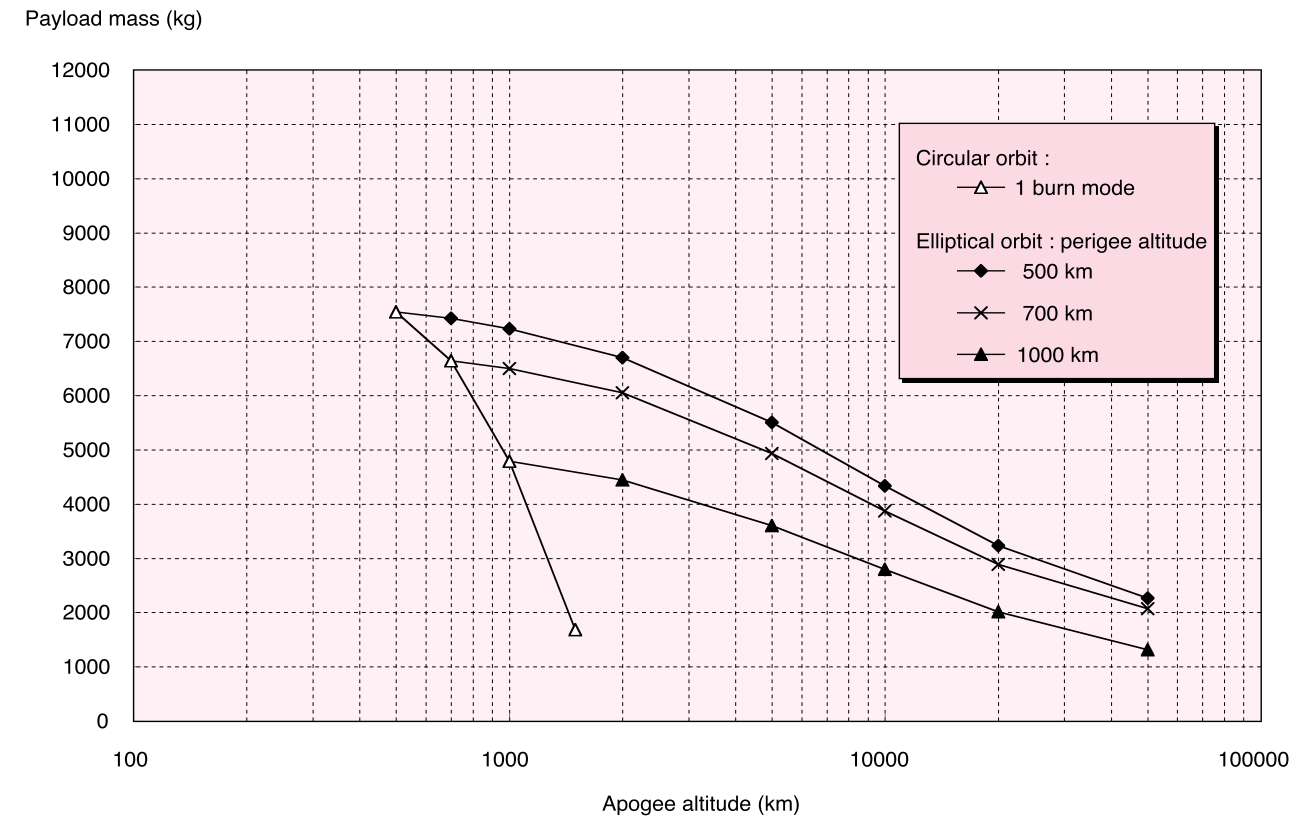


Figure 2.5.2 Payload capability for LEO mission (H2A202 with 5S fairing)
(inclination 51.6 deg)

<With the LE-7A lower nozzle skirt>

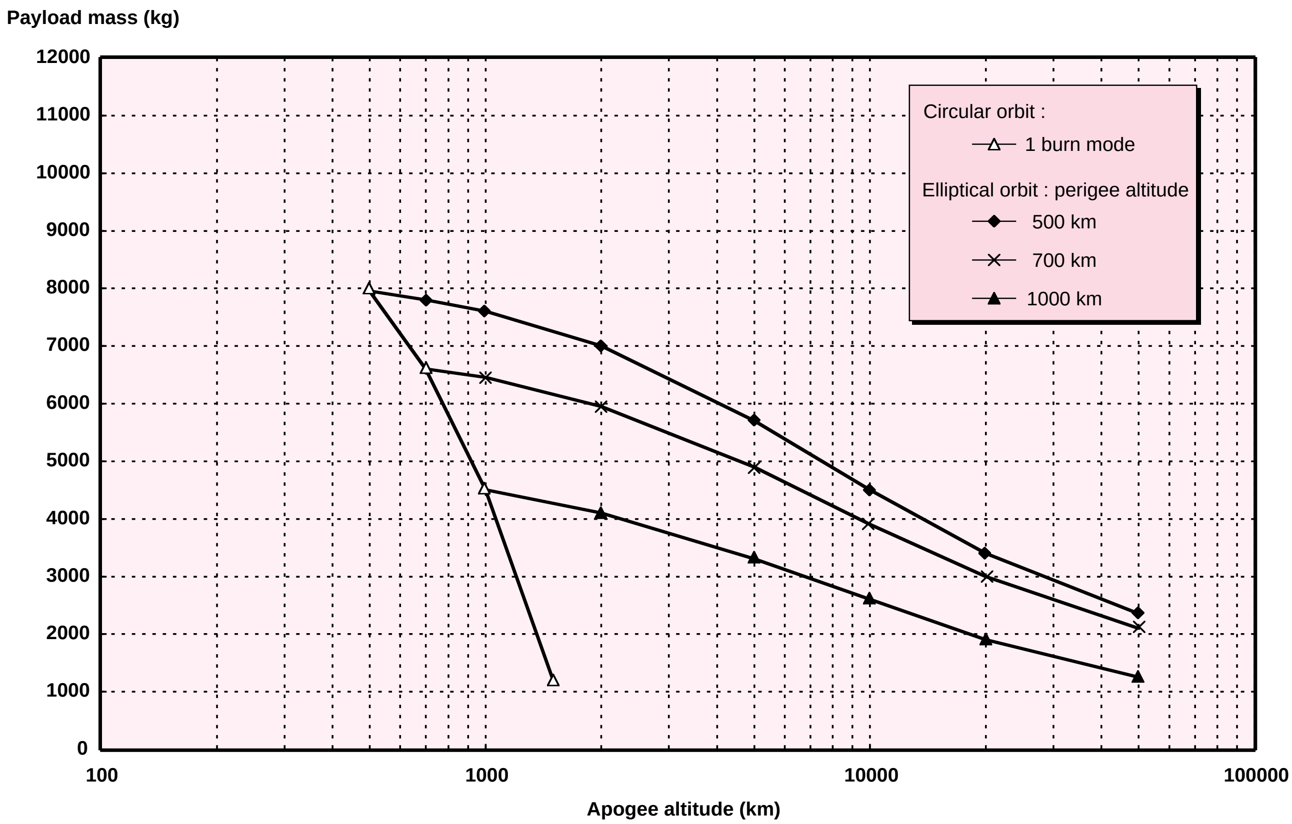


Figure 2.5.3 Payload capability for LEO mission (H2A202 with 5S fairing)
(inclination 30.4 deg)

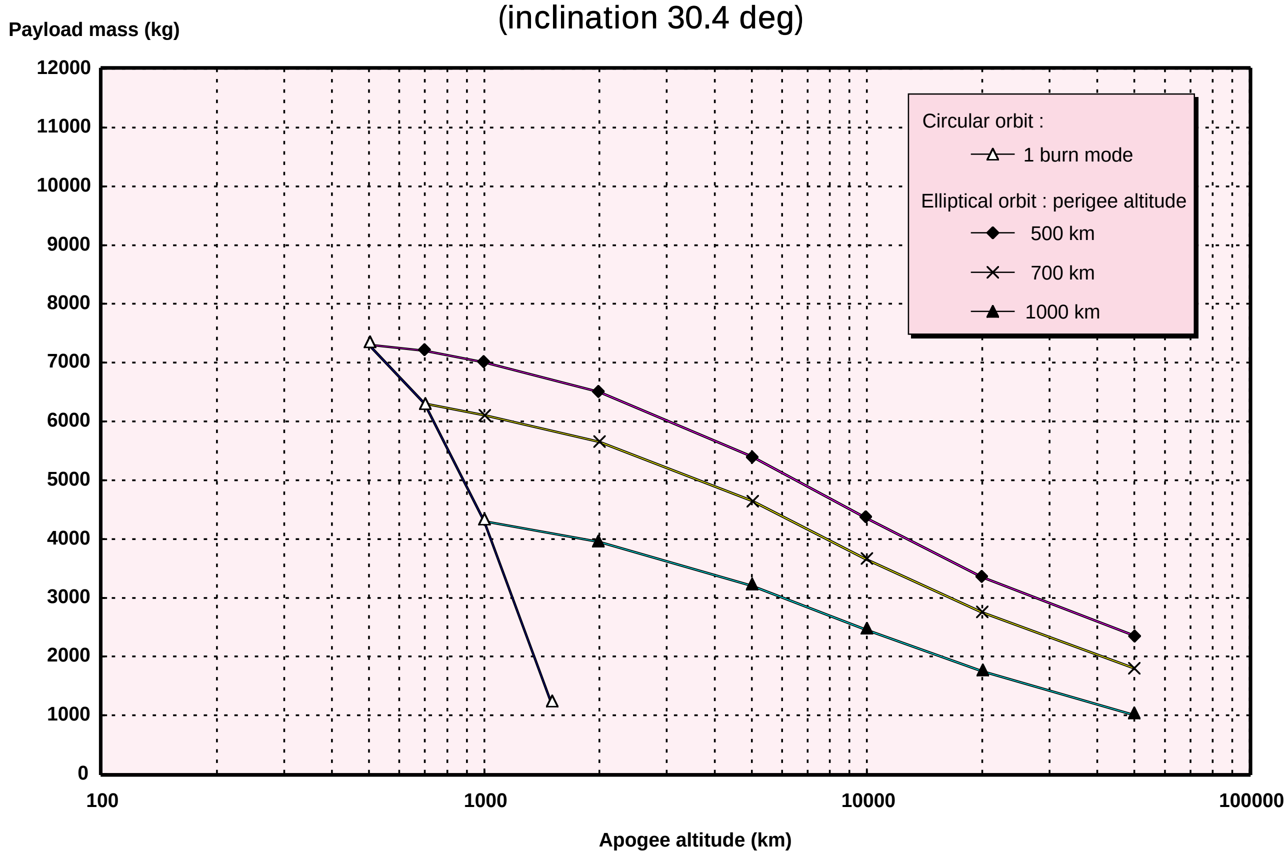


Figure 2.5.3 Payload capability for LEO mission (H2A202 with 5S fairing)
(inclination 51.6 deg)

<Without the LE-7A lower nozzle skirt>

C

2.6 Earth Escape Mission

2.6.1 Payload capability

Figure 2.6.1 shows payload capability for Earth Escape mission using model 4S and 5S fairing (See § 4.4). In this figure, the basic configuration is the two stage configuration. C

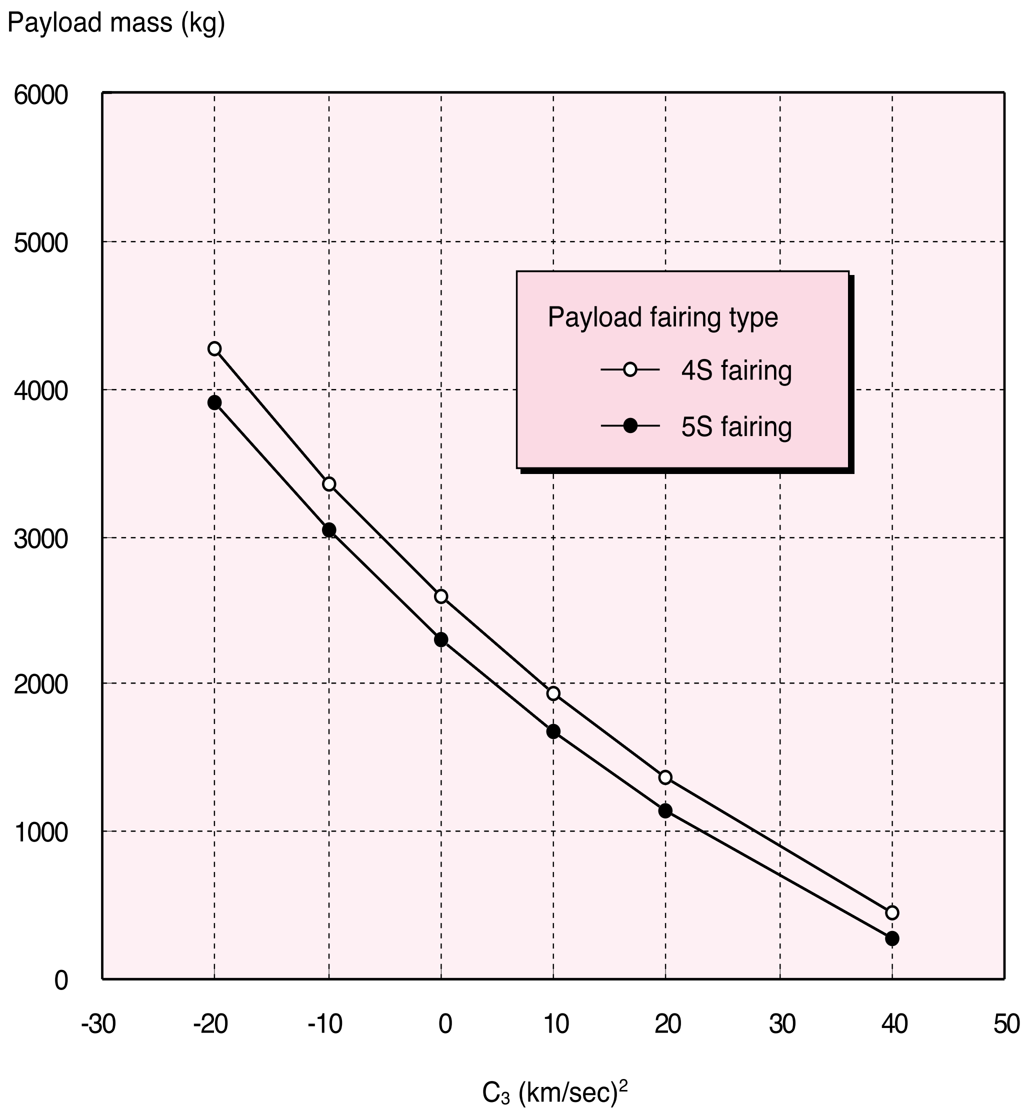


Figure 2.6.1 Payload capability for earth escape mission (H2A202)

<With the LE-7A lower nozzle skirt> C

2.7 Spacecraft Orientation and Separation

2.7.1 General description

When the second stage is injected to the planned orbit, the second stage provides the required orientation and spin (if required, see the § 2.7.3) for the spacecraft before its separation. After completion of the spacecraft separation, the second stage conducts the CAM 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, rate and sequence) with 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 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.

(3) After separation

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

2.7.3 Spin-up performance

The second stage RCS can rotate the launch vehicle body and a spacecraft with speed up to 5 rpm clockwise or counterclockwise before spacecraft separation. The user should coordinate details with NASDA if spin-up is needed.

If the user needs a spin-rate exceeding 5 rpm, 937M-spin or 937M-Spin-A adapter with a spin table attached are available.

2.7.4 Pointing accuracy

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

Following values are for roll, pitch and yaw axes.

Without pointing maneuvering : Less than ± 10 °
With pointing maneuvering : Less than ± 3 °
(When spin-up is performed, above data is available before spin-up)
Attitude rate accuracy (before separation)
: Less than 0.3 °/sec for non-spin spacecraft separation
Attitude rate accuracy (before separation)
: Less than 0.25 °/sec for spin spacecraft separation

2.7.5 Relative separation velocity

Relative velocity at spacecraft separation is nominal about 0.5 to 1.0 m/sec. | A

If the user needs a different velocity, the launch vehicle can provide it as long as the new velocity does not inhibit launch vehicle avoidance maneuvers. For upper spacecraft separation, the relative velocity will be set automatically by mission requirements.

For all missions, NASDA will require a spacecraft attitude control and mission control plan in order to analyze and modify launch vehicle movement after spacecraft separation.

2.7.6 Separation tip-off rate

Separation tip-off rate depends on the separation mechanism.

With the spring ejection mechanism, springs installed in adapter push the spacecraft at separation.

With the retro thruster mechanism, the second stage of launch vehicle moves backwards using retro thrusters at separation; there are no springs in the adapter.

Separation spring mechanism
Tip-off rate : Less than 2.0 °/sec
Retro thruster mechanism
Tip-off rate : Less than 1.0 °/sec

2.7.7 Dual launch sequence

Figure 2.7.1 shows a typical separation sequence for dual launch on LEO (upper)-GTO (lower) mission.

Figure 2.7.2 shows a typical separation sequence for dual launch on GTO (upper)-GTO (lower) mission.

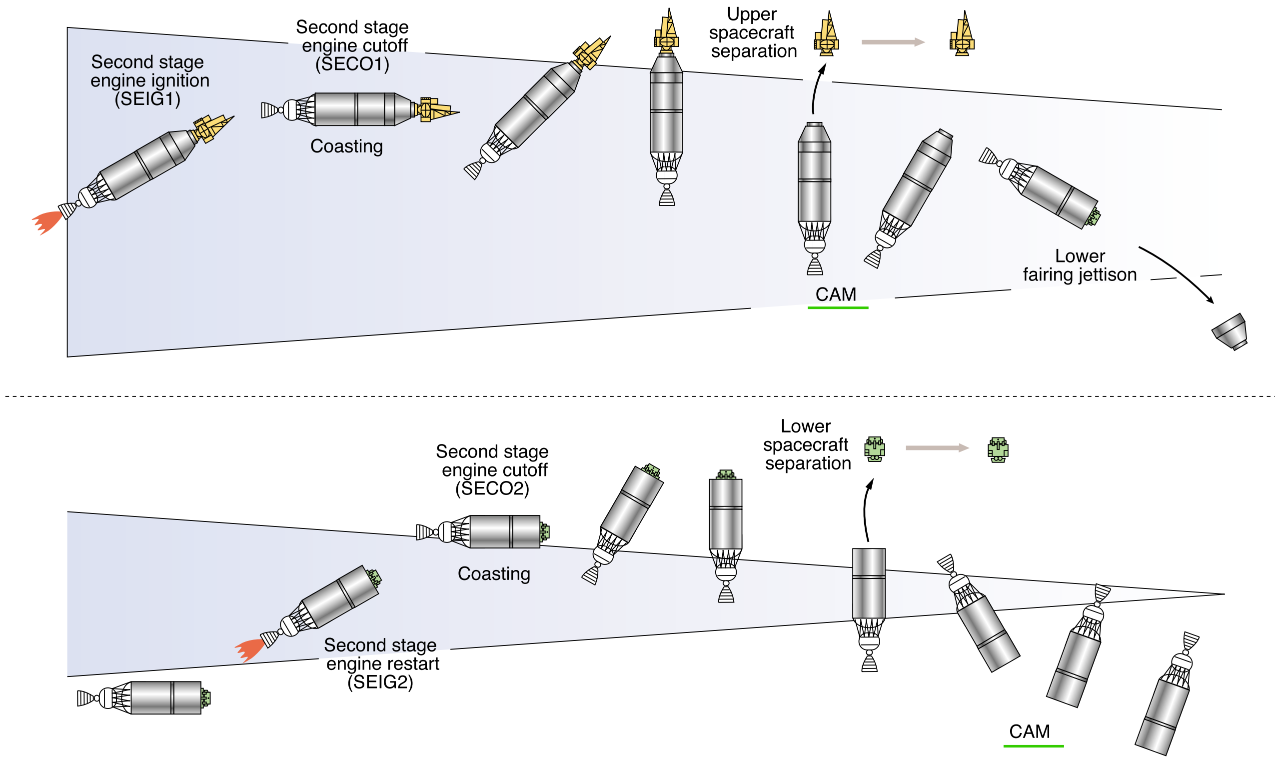


Figure 2.7.1 Typical separation sequence for dual launch on **LEO-GTO** mission

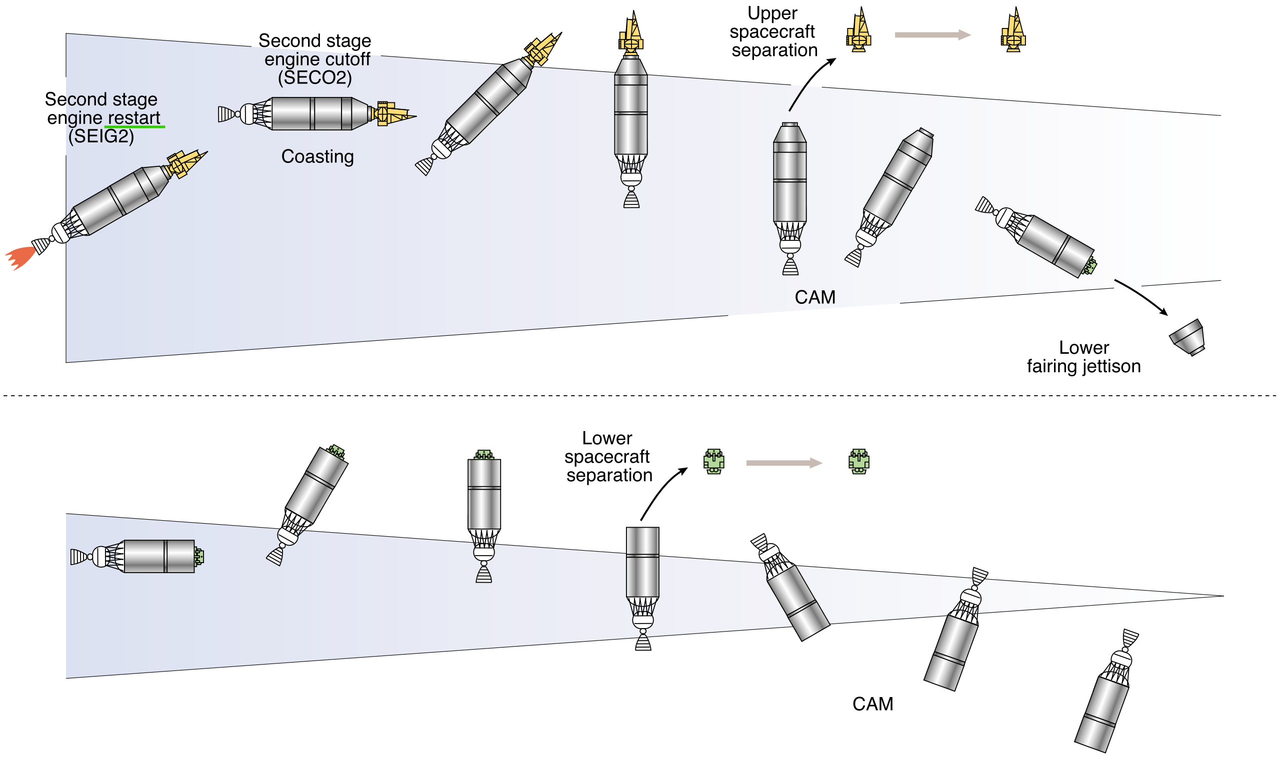


Figure 2.7.2 Typical separation sequence for dual launch on **GTO-GTO** mission

CHAPTER 3.

ENVIRONMENTS

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. So the spacecraft test requirements are also described.

3.2 Mechanical Environments

3.2.1 General

During lift-off 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 transfer to the launch pad), encapsulation of the spacecraft and mating spacecraft to the second stage. The load factors described in § 3.2.2 to 3.2.5 should be considered as limit loads applied to the spacecraft.

3.2.2 Combined load factor A

Figure 3.2.1 shows a typical longitudinal static acceleration flight time history. B

During lift-off 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 A acceleration should be considered as the design limit load factors for the spacecraft primary structure.

For the H-IIA rocket, the combined load factor at lift-off and at main engine (of A the first stage) cutoff (MECO) transient cover the maximum loads of the spacecraft primary structure during lift-off 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 CLA results. Therefore the user should discuss loads conditions on the spacecraft structures with NASDA.

3.2.3 Sinusoidal vibration

The spacecraft is exposed to vibration environments 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 § 3.2.4.

The levels shown below are enveloped levels of the low frequency vibrations which are exerted during lift-off and in-flight phases, in particular at lift-off, maximum $Q\alpha'$ (which means the maximum product of dynamic pressure and total angle of attack), MECO, the first and the second stage separation, etc. These levels are prescribed at the spacecraft interface (spacecraft separation plane).

Limit level (3-sigma high)

Longitudinal	1.0 $G_{0\text{-}P}$	for	5 to 30 Hz		A
	0.8 $G_{0\text{-}P}$	for	30 to 100 Hz		
Lateral	0.7 $G_{0\text{-}P}$	for	5 to 18 Hz		
	0.6 $G_{0\text{-}P}$	for	18 to 100 Hz	G = 9.80665 m/s^2	B

(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.)

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 are defined in detail according to the coupled loads analysis (CLA). As to the structure for which sinusoidal vibration environmental test cannot simulate the dynamic load of the flight condition sufficiently, confirmation of the strength shall be carried out by tests and analyses separately.

3.2.4 Random vibration

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

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.

Figure 3.2.2 and Figure 3.2.3 show the acoustic environment level inside the payload fairing for H2A202 series and H2A212 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 be able to endure to this level for 60 seconds. The reference point 0 dB of sound pressure level (SPL) is equivalent to 20 μPa. A

3.2.6 Shock

Table 3.2.2 shows a summary of pyrotechnic shock events during flight on all kinds of H-IIA 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 3.2.4 and Figure 3.2.5 show a typical spacecraft separation shock spectrum at the spacecraft separation plane with 1194M adapter and 2360S adapter respectively. These shock spectrums are exerted uniformly in all directions. A

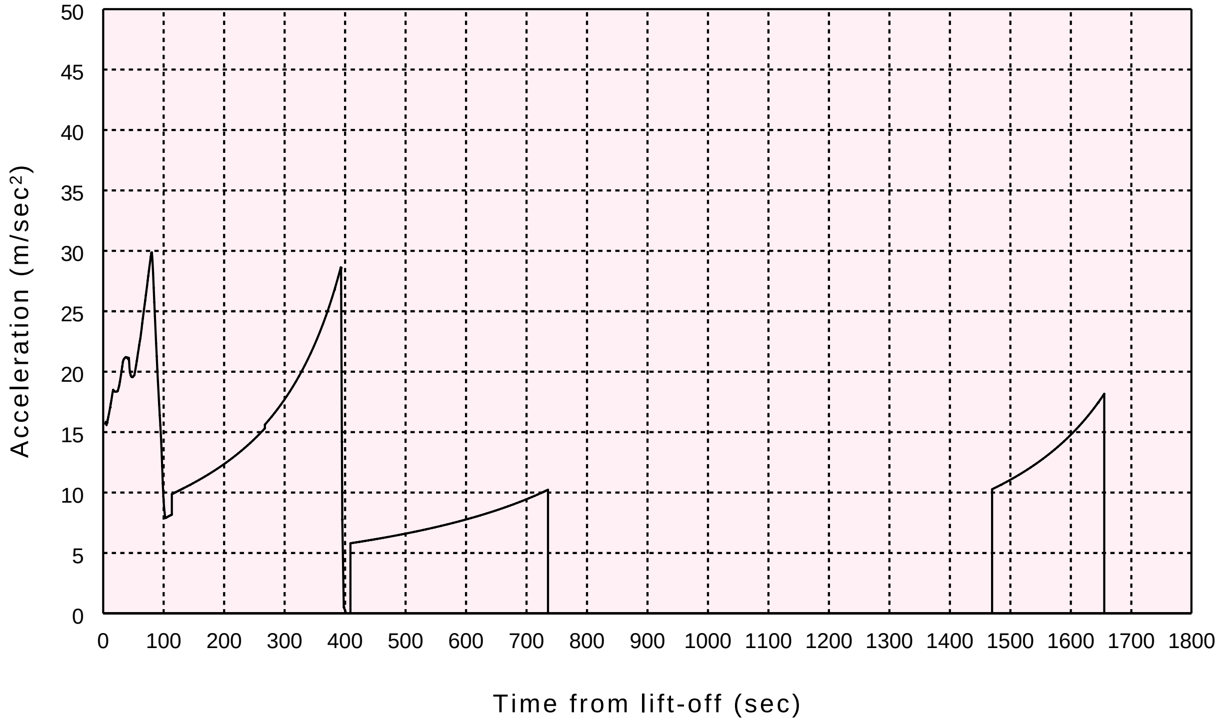


Figure 3.2.1 Typical longitudinal static acceleration for GTO (H2A202)

Table 3.2.1 Combined loads (Limit level)

Event		Acceleration	Remarks	
Lift-off	Combined loads * for compression	Longitudinal − 3.2 G Lateral ± 1.8 G	1.7 G (steady) + (1.5 G (dynamic))	B
	Combined loads * for tension	Longitudinal − 0.1 G Lateral ± 1.8 G		
At MECO	Immediately before MECO	Longitudinal − 4.0 G Lateral ± 0.5 G	H2A202, H2A2022, H2A2024	A
	MECO Transit	Longitudinal + 1.0 G Lateral ± 1.0 G		

$G = 9.80665\ m/s^2$ B
*** : Maximum load at the top of the payload adapter**
Lateral : ± may act in either direction
As for the longitudinal loads, all of them are difined with the tension loads as positive. B

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

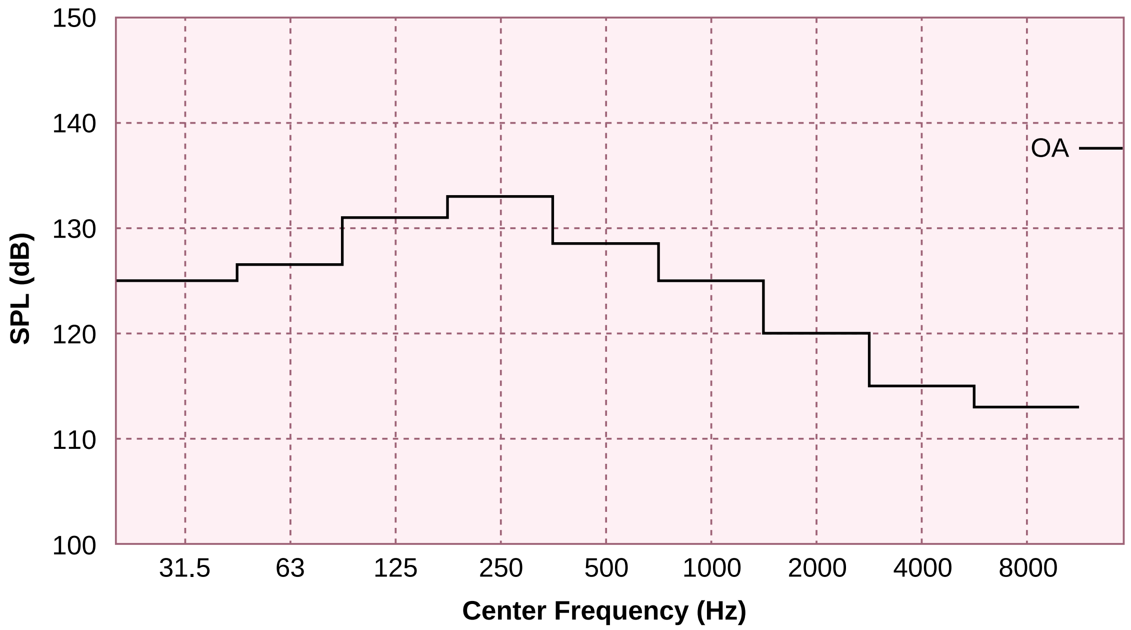


Figure 3.2.2 Sound pressure level inside fairing with acoustic blanket **(H2A202, H2A2022, H2A2024)** | A

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

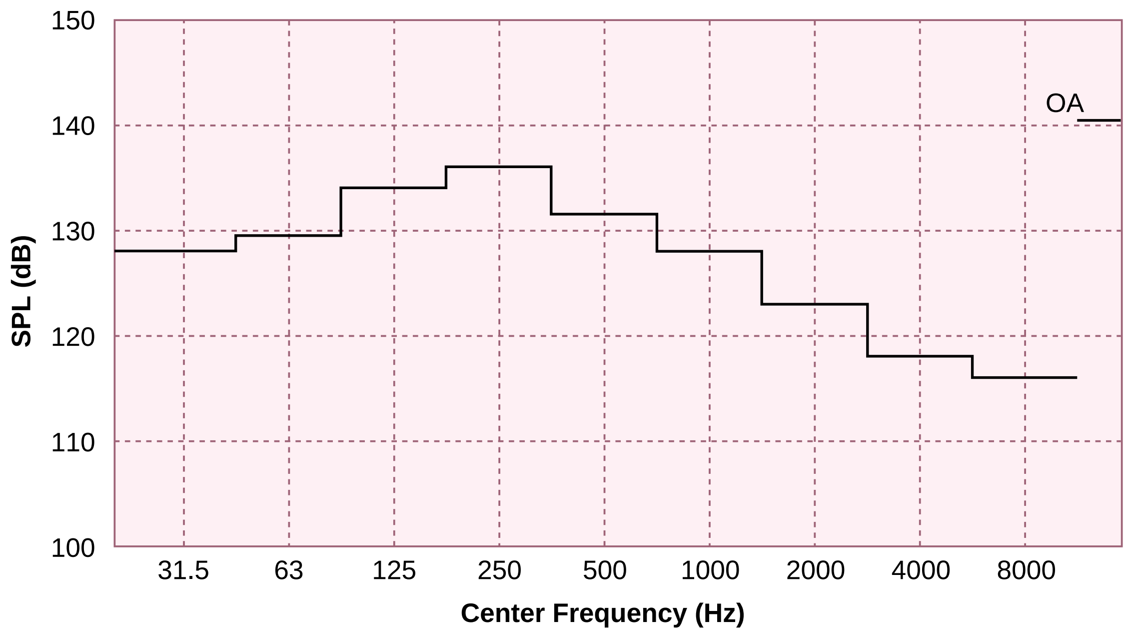


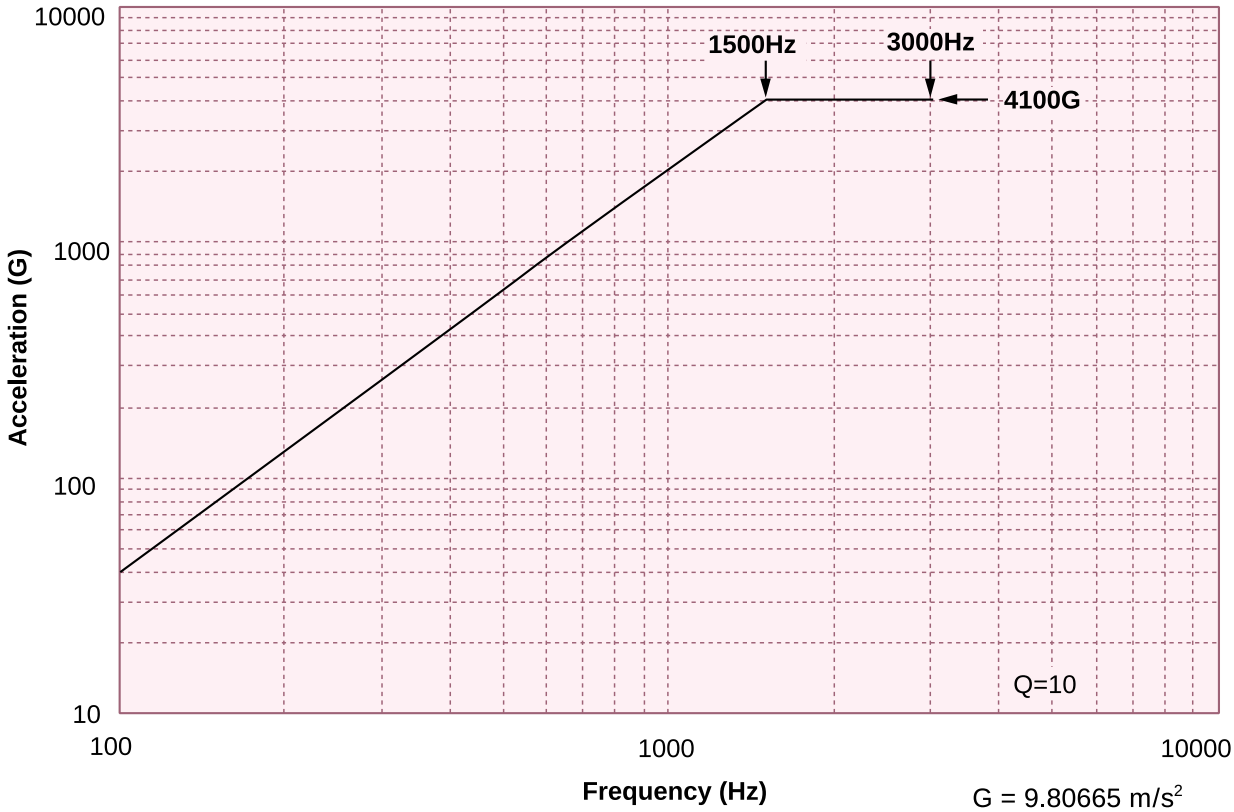
Figure 3.2.3 Sound pressure level inside fairing with acoustic blanket **(H2A212)** | A

Table 3.2.2 Summary of pyrotechnic shock events

Pyrotechnic shock events	H2A202	H2A2022	H2A2024	H2A212	H2A222 (REF)	Remarks
SRB-A separation	✓	✓	✓	✓	✓	
First SSB pair separation	N/A	✓	✓	N/A	N/A	
Second SSB pair separation	N/A	N/A	✓	N/A	N/A	
LRB separation	N/A	N/A	N/A	✓	✓	
Fairing jettison	✓	✓	✓	✓	✓	upper fairing in dual launch
First / Second stage separation	✓	✓	✓	✓	✓	
Spacecraft separation	✓	✓	✓	✓	✓	upper spacecraft in dual launch
Fairing jettison (lower)	✓	✓	✓	✓	✓	in dual launch
Spacecraft separation (lower)	✓	✓	✓	✓	✓	in dual launch

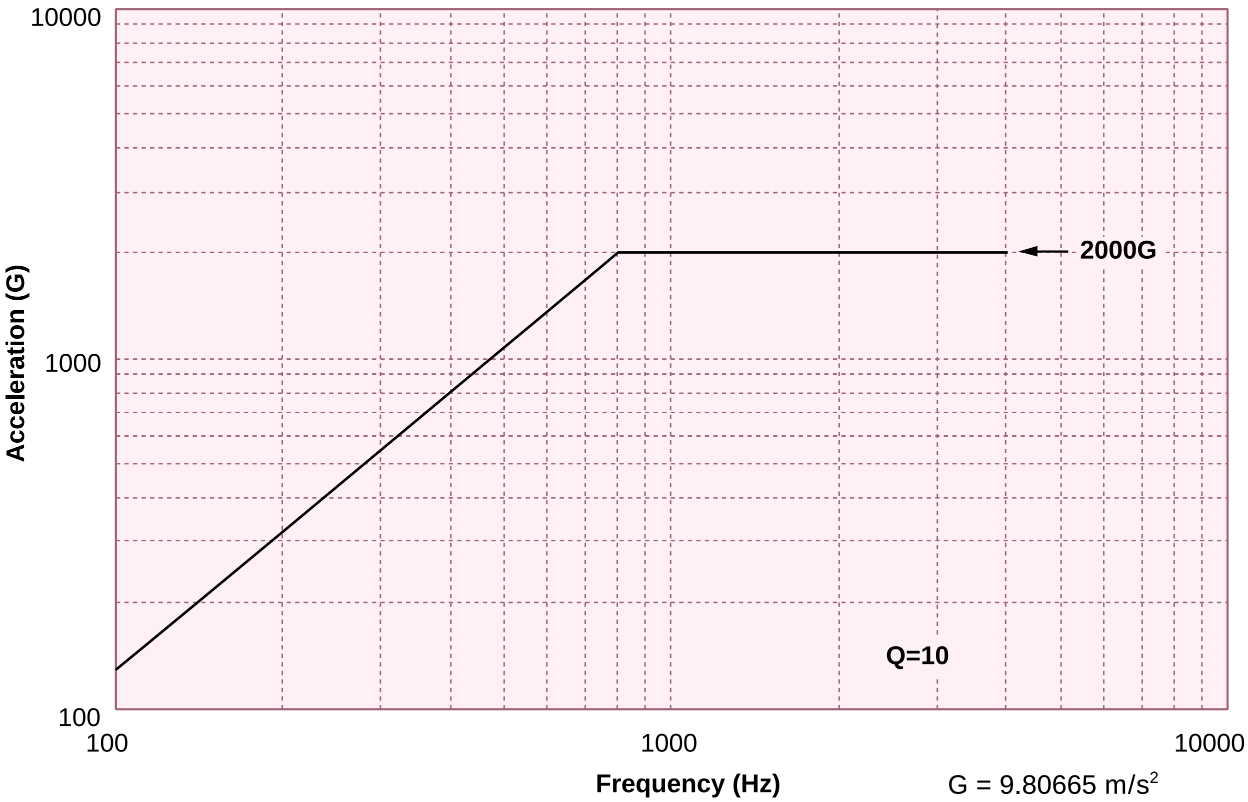
A

(lower) means adaptable in case of dual launch.
N/A means 'not applicable'.



G = 9.80665 m/s^2 | B

Figure 3.2.4 Typical spacecraft separation shock spectrum with 1194M adapter | A



| B

Figure 3.2.5 Typical spacecraft separation shock spectrum with 2360S adapter | A

3.3 Thermal Environment

3.3.1 General

Concerning thermal environments, four phases have to be considered. These are as follows :

a) the phase of spacecraft preparation in the Ground Support Equipment (GSE) buildings and transport between these buildings (Refer to the GSE Manual for details)
b) the phase of spacecraft encapsulation into the payload fairing (PLF) and mating to the launch vehicle in the Vehicle Assembly Building (VAB).
c) the phase of transfer to the launch pad and the final prelaunch
d) the in-flight phase

The phases from a) to c) are concerned with prelaunch environment and d) concerns launch and flight environment. A

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 PLF, mating to the launch vehicle, final checkout, transport to the launch pad, etc.

Environments during these activities are shown in Table 3.3.1.

Environments in the No. 2 spacecraft test and assembly building (STA2) (or No. 1 spacecraft test and assembly building (STA1)), in which spacecraft integration and functional tests are main activities, are controlled at 22 ± 3 °C (71.6 ± 5.4 °F) and 50 ± 10 % relative humidity. A

During the ground transport from the STA2 (or STA1) to the spacecraft and fairing assembly building (SFA) (or the third stage and spacecraft assembly building (TSA)), the temperature inside the PLF is not controlled because the spacecraft is loaded in the container (which is shielded completely) and the transport is carried out very quickly (about 30 minutes, not including preparation time in the STA2 or STA1).

Environments in the SFA are controlled according to the kind of each activity. In case of a normal operation, the temperature is controlled at 21 ± 3 °C (69.8 ± 5.4 °F). For each operation of battery charging and propellant loading, the temperature is controlled at 18 ± 3 °C (64.4 ± 5.4 °F) and at 21 ± 1 °C (69.8 ± 1.8 °F) respectively. The relative humidity is controlled at 45 ± 5 % during all activities. A

During the ground transport from the SFA to the VAB, the temperature inside the PLF is not usually controlled because the spacecraft is loaded in the PLF (which is shielded completely) and the transport is conducted very quickly (about one hour, not including preparation time in the SFA). If required, condition is controlled (option). A A

During operations of hoisting the encapsulated spacecraft and mating it onto the launch vehicle, air conditioning inside the PLF is not carried out for the reason that environments in the VAB are controlled at 24 ± 3 °C (75.2 ± 5.4 °F) (in Summer), 20 ± 3 °C (68.0 ± 5.4 °F) (in Winter), and 50 ± 10 % relative humidity.

After the air conditioning duct is connected to the PLF, environments inside the PLF are controlled at 10 – 25 ± 2 °C (50 – 77 ± 3.6 °F) and 40 – 50 % relative humidity until lift-off (including a transfer phase from the VAB to the launch pad (LP)).

3.3.3 Launch and flight environment

(1) Within fairing (from lift-off to fairing jettison)

The spacecraft is protected by the payload fairing during ascent to a nominal altitude of about 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 jettison.

Figure 3.3.1 shows typical internal surface temperature profiles of the 4S fairing or 4/4D upper fairing for a Geostationary Transfer Orbit (GTO) mission. And Figure 3.3.2 shows typical internal surface temperature profiles of the 5S fairing or 5/4D upper fairing for a GTO mission. And Figure 3.3.3 shows typical internal surface temparature profiles of the 5S-H fairing for a H-II Transfer Vehicle (HTV) mission. Further, Table 3.3.2 shows the maximum temperature and emittance of internal surfaces of the payload fairing during ascent flight. The maximum heat flux radiated by the fairing is less than 500 W/m^2, and the maximum temperatures remain below 140 °C at the warmest location (the nose cap).

(2) After fairing jettison

The nominal timing for jettisoning the fairing on all flights is determined so that the maximum free molecular heat flux does not exceed 1135 W/m^2. A typical free molecular heating profile for a GTO mission is shown in Figure 3.3.4. 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 jettisoning the fairing includes free molecular heating, solar heating, Earth heating, radiation to the second stage and deep space and thermal flux conducted from the forward end of the second stage through the spacecraft adapter. In addition, during the collision / contamination avoidance of the second stage after the spacecraft separation, the thruster plumes from the RCS and LH_2 / GH_2 and LOX / GOX vented through LH_2 and LOX tanks might influence the spacecraft. Solar and Earth thermal heating 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 Gas conditioning capabilities

Phase		Inside building					Inside payload fairing				
		Room temp capability °C (°F)	Flowrate capability Sm^3/min	Inlet pressure capability kPa	Relative humidity %	Cleanliness	Inlet temp capability *1 °C (°F)	Flowrate capability Sm^3/min	Inlet pressure capability kPa	Relative humidity %	Cleanliness
STA2		22 ± 3 (71.6 ± 5.4)	–	–	50 ± 10	class 100000	N / A	N / A	N / A	N / A	N / A
Transport from STA2 to SFA		N / A	N / A	N / A	N / A	N / A	N / A	N / A	N / A	N / A	N / A
SFA	Normal operation	21 ± 3 (69.8 ± 5.4)	–	–	45 ± 5	class 100000	N / A	N / A	N / A	N / A	N / A
	Battery charging	18 ± 3 (64.4 ± 5.4)	–	–	45 ± 5	class 100000					
	Propellant loading	21 ± 1 (69.8 ± 1.8)	–	–	45 ± 5	class 100000					
	After encapsulation	21 ± 3 (69.8 ± 5.4)	–	–	45 ± 5	class 100000	According to room circumstances				
Transport *2 from SFA to VAB		N / A	N / A	N / A	N / A	N / A	10 – 25 (± 2) (50 – 77 (± 3.6))	Upper : 50 Lower : 50	7	40 – 50	class 5000
VAB		24 ± 3 (in Summer) (75.2 ± 5.4) 20 ± 3 (in Winter) (68.0 ± 5.4)	–	–	50 ± 10	class 100000	10 – 25 (± 2) (50 – 77 (± 3.6))	Upper : 50 Lower : 50	7	40 – 50	class 5000
Transport from VAB to LP		N / A	N / A	N / A	N / A	N / A	10 – 25 (± 2) (50 – 77 (± 3.6))	Upper : 50 Lower : 50	7	40 – 50	class 5000

<Note> ***1 : Inlet temperature is adjustable within system capability according to Spacecraft requirements.**
***2 : Conditioning control during transporting is option.**

Table 3.3.2 Maximum temperatures and emittances of fairing internal surface

Section	Max. temp. (℃) (T.B.D.)	Emittance (T.B.D.)	
Nose cap	140	0.15	B
Nose cone	100	0.15	B
Cylinder	120	0.15	B
Acoustic blanket	34	0.85	

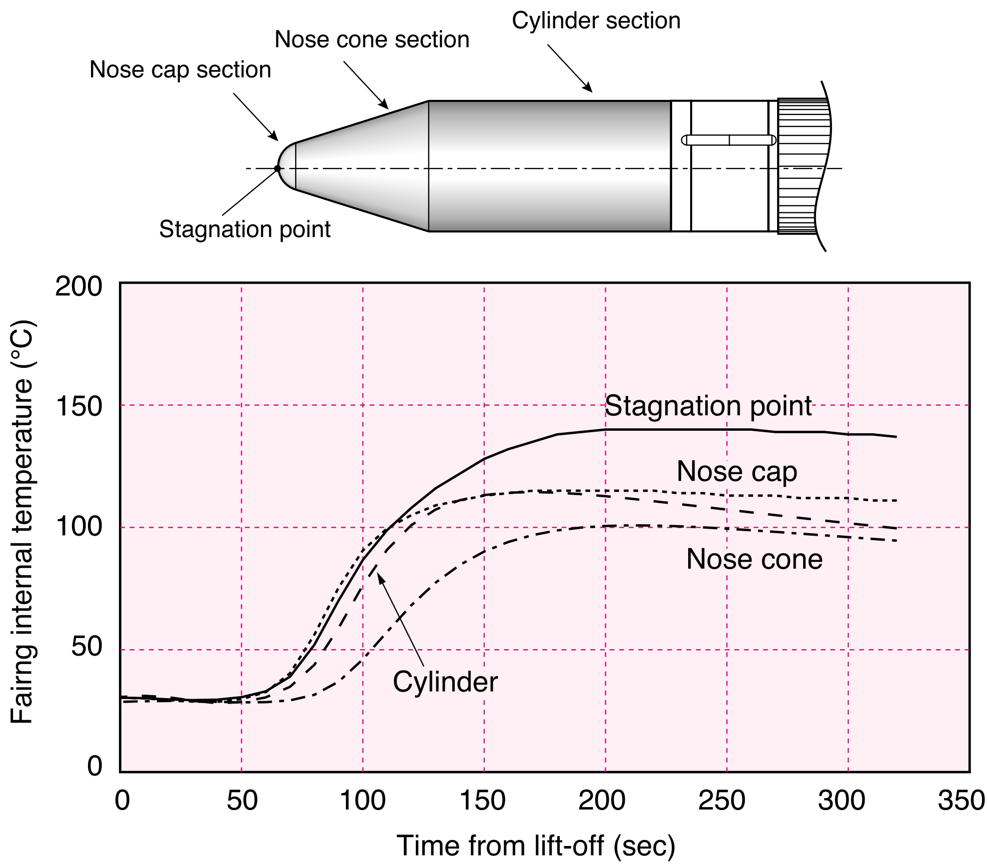


Figure 3.3.1 Internal surface temperature profiles of 4S or 4/4D upper fairing for GTO mission

A

B

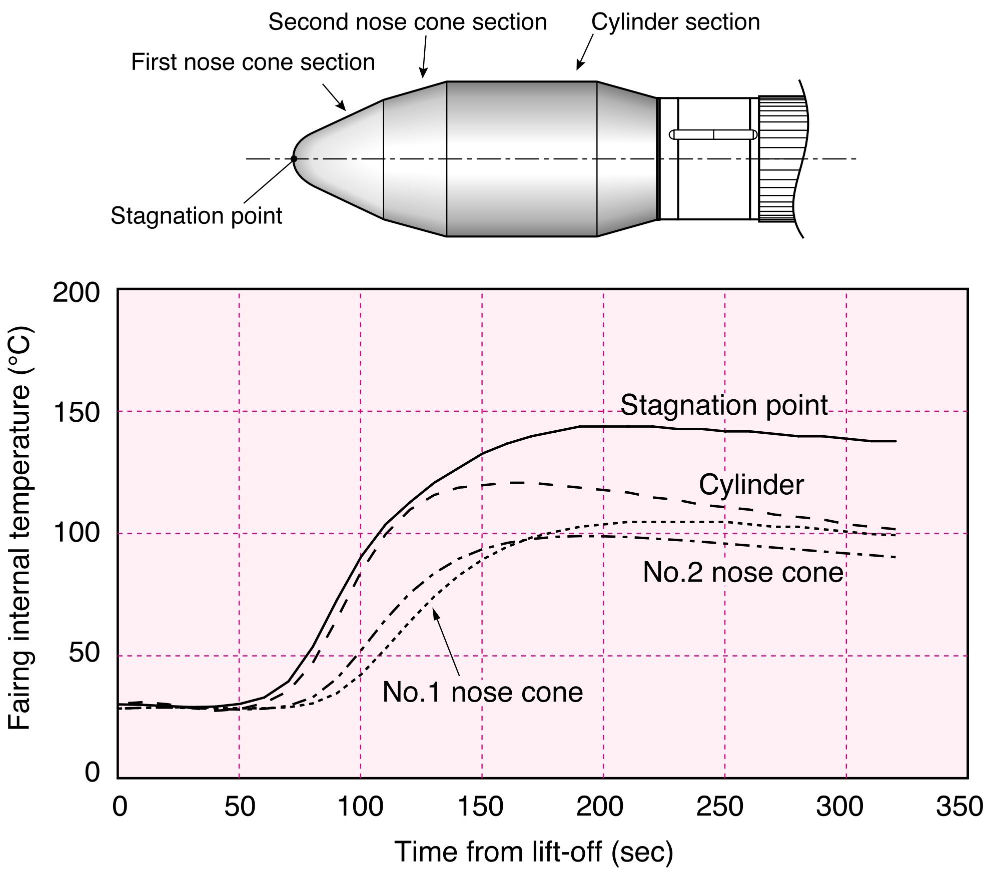


Figure 3.3.2 Internal surface temperature profiles of 5S or 5/4D upper fairing for GTO mission

A

B

A

C

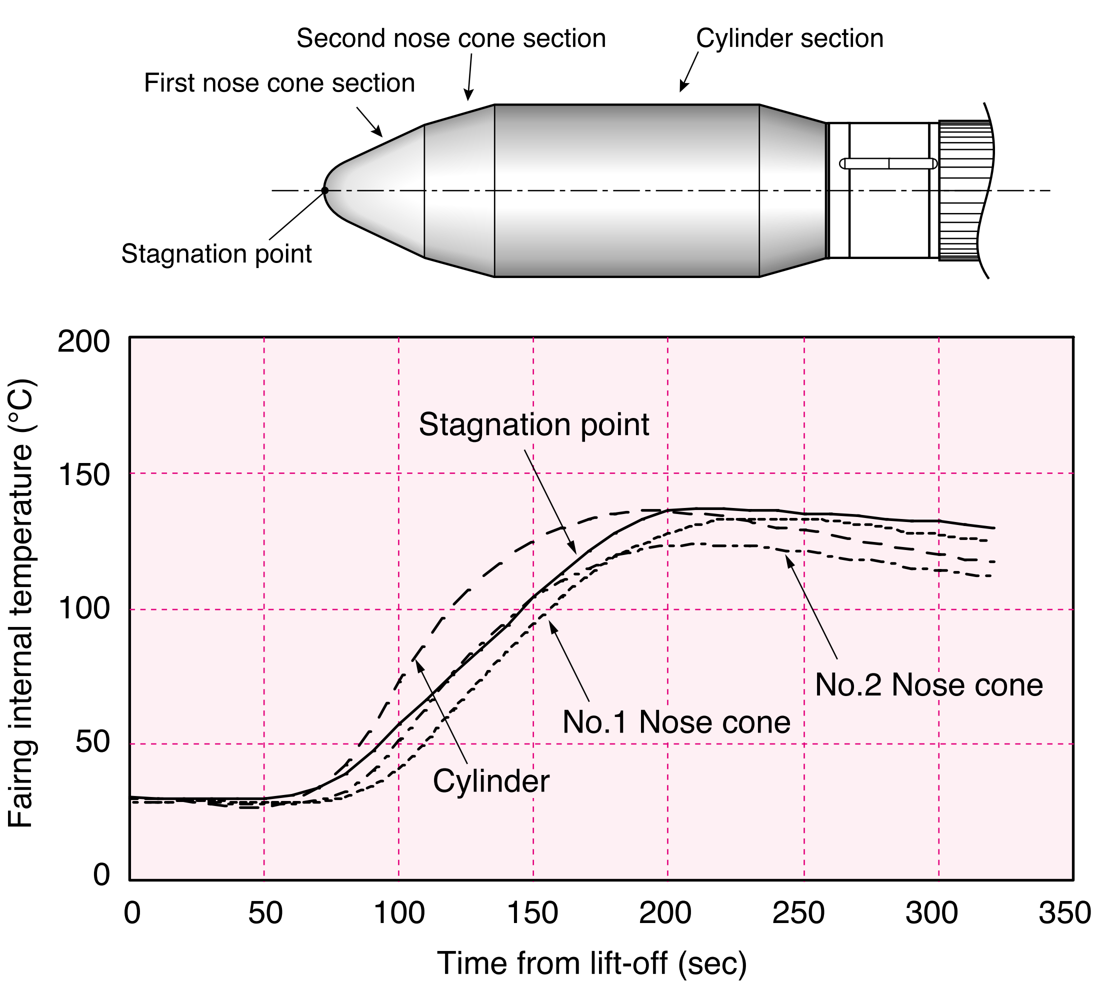
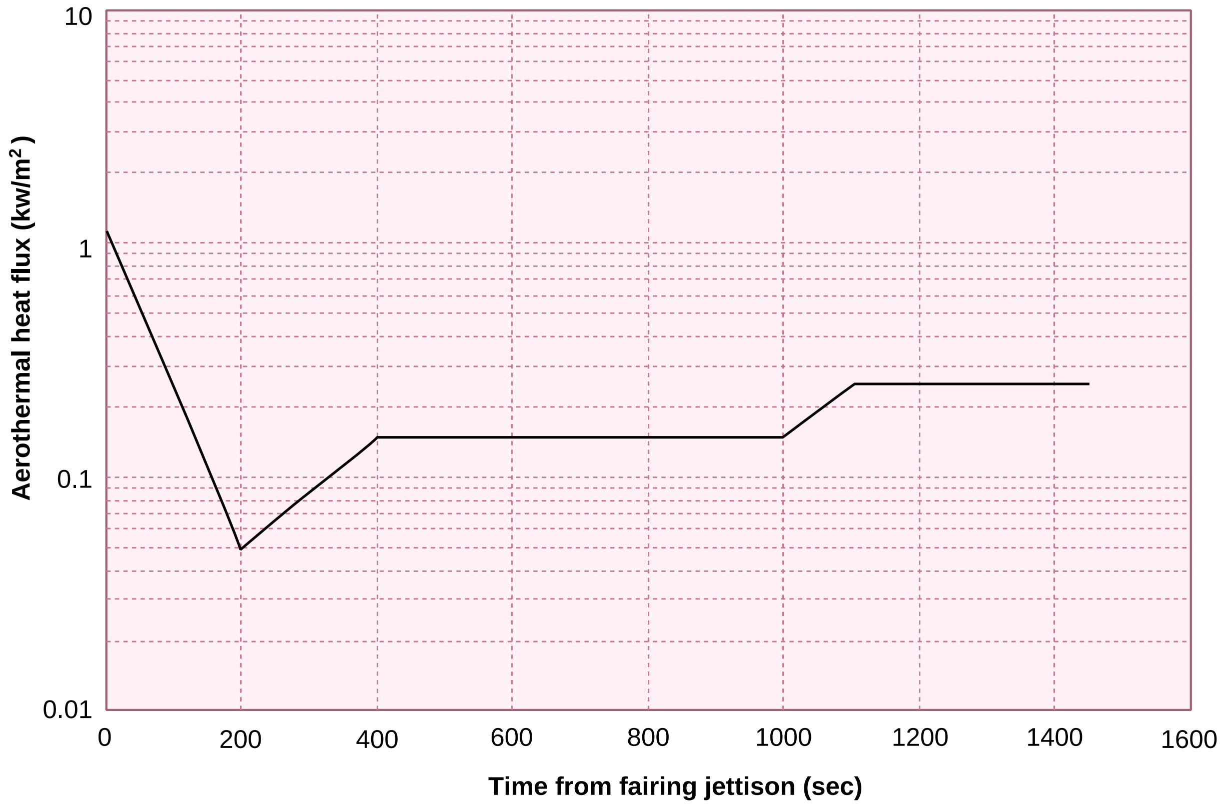


Figure 3.3.3 Internal surface temperature profiles of 5S-H fairing for HTV mission



C

Figure 3.3.4 Aerothermal heat flux of the free molecular flow (GTO mission) | C

3.4 Fairing Internal Pressure Environment

Air inside the payload fairing is vented during the ascent phase through one-way vent ports. Typical predicted internal pressure profiles and pressure decay rate profiles for the 4S fairing are shown in the following figures respectively.

a) Figure 3.4.1 : Typical internal pressure profile for GTO mission in case of the 4S fairing
b) Figure 3.4.2 : Typical pressure decay rate profile for GTO mission in case of the 4S fairing
c) Figure 3.4.3 : Typical internal pressure profile for SSO mission in case of the 4S fairing
d) Figure 3.4.4 : Typical pressure decay rate profile for SSO mission in case of the 4S fairing

As shown in these figures, the pressure decay rate typically varies while the launch vehicle is in the transonic flight phase. Maximum pressure decay rates for the 4S fairing are as follows :

a) GTO mission : 2.50 kPa/sec
b) SSO mission : 2.57 kPa/sec

C

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. If a user requires such venting, the user should check with NASDA.

In case of the 1194 adapter, it has two vent holes whose diameter is 45 mm respectively (refer to A3.3).

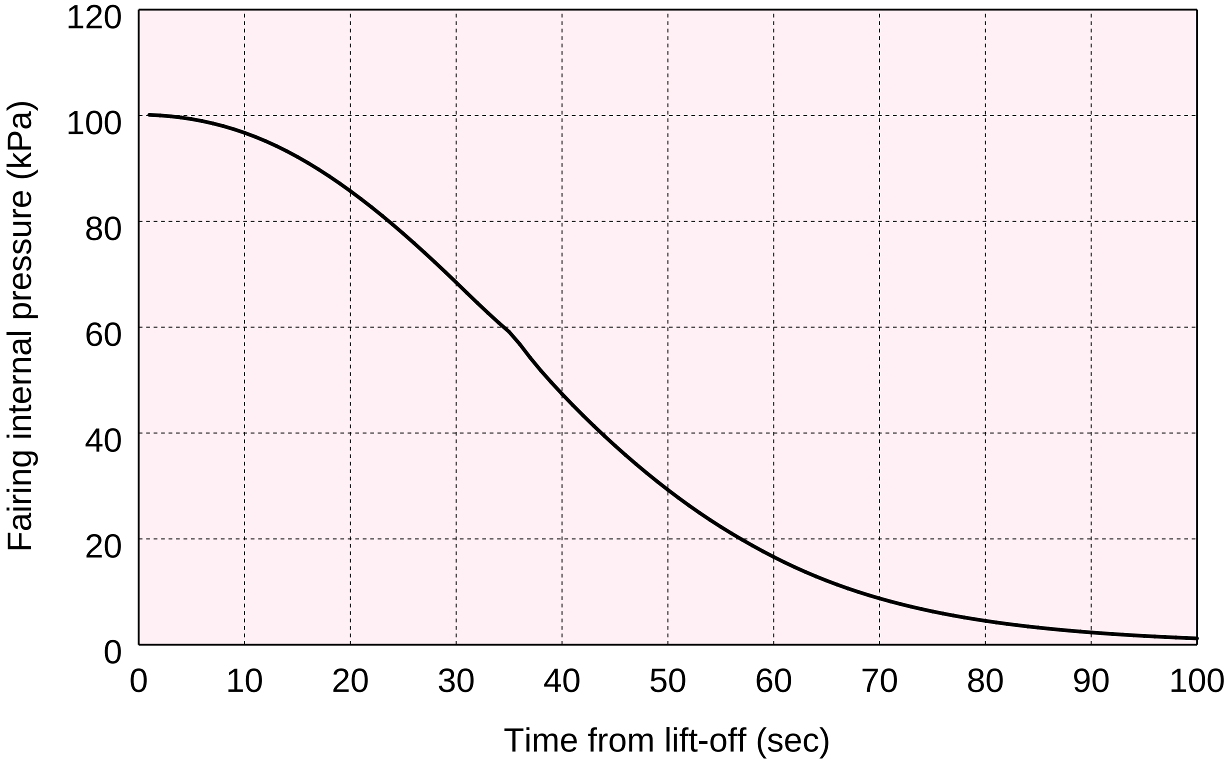


Figure 3.4.1 Typical internal pressure profile for GTO mission (Model 4S fairing)

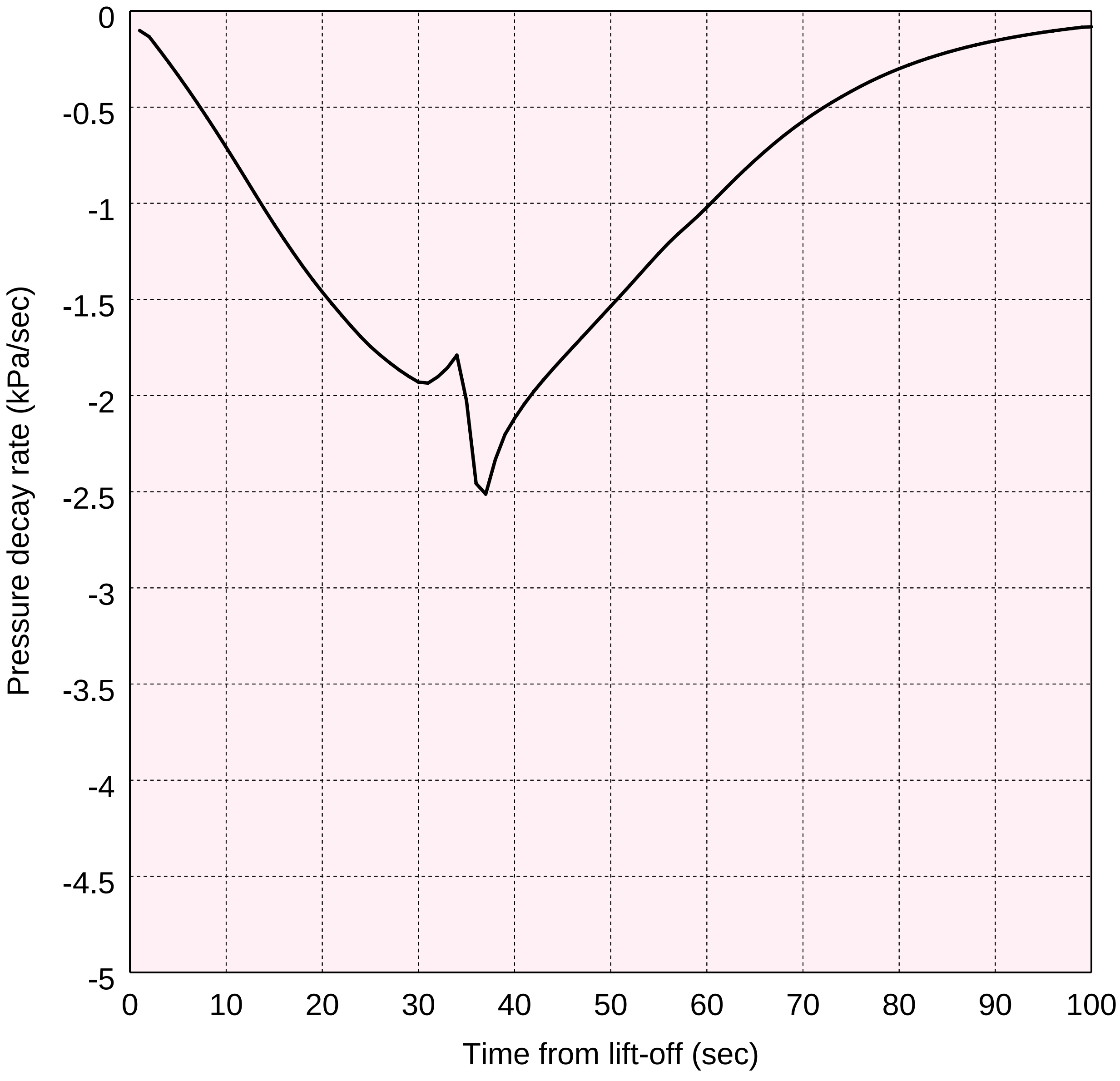
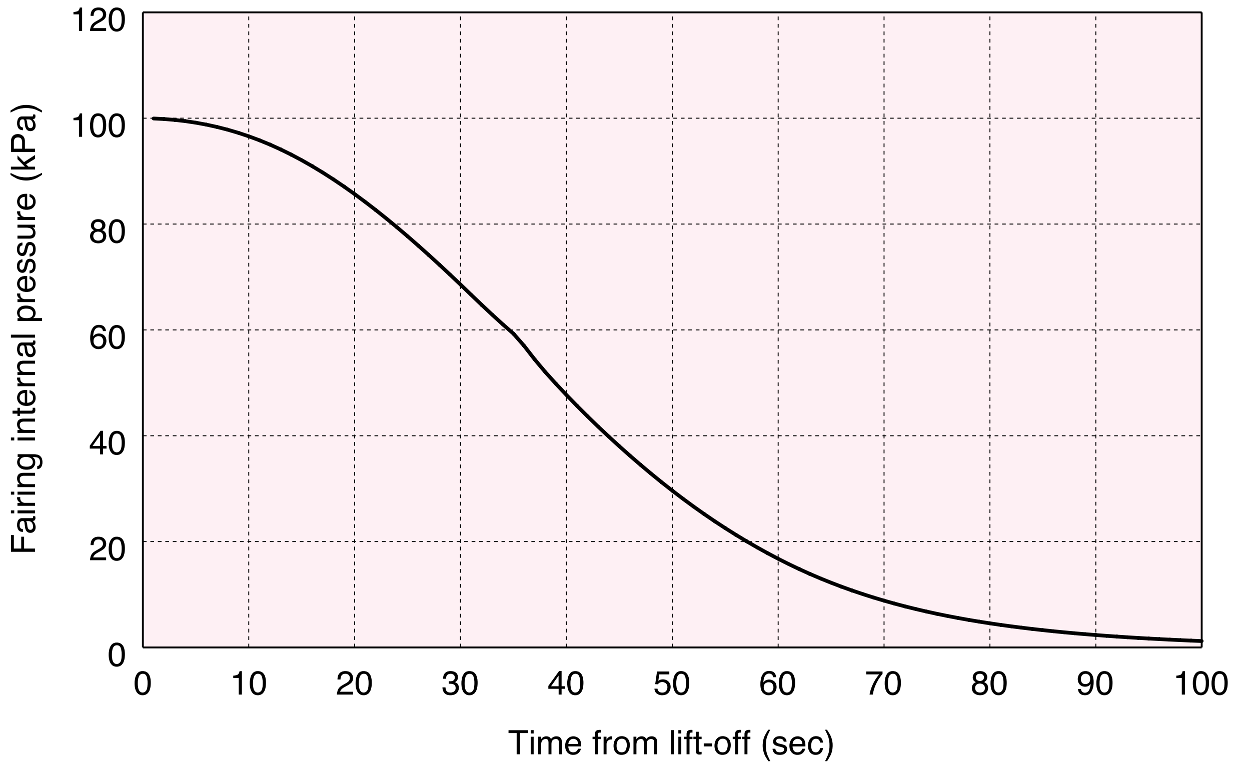
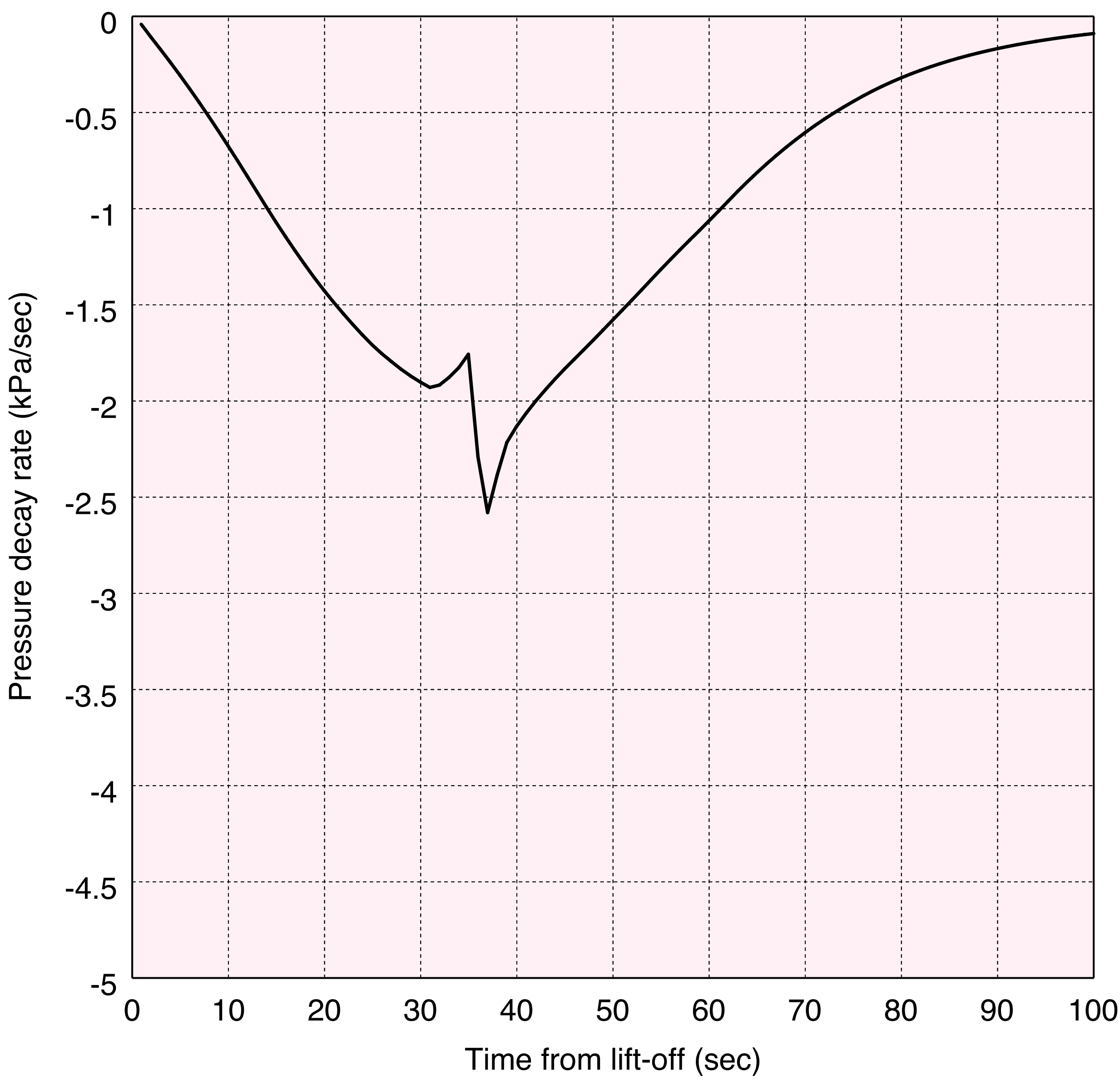


Figure 3.4.2 Typical pressure decay rate profile for GTO mission (Model 4S fairing)



C

Figure 3.4.3 Typical internal pressure profile for SSO mission (Model 4S fairing)



C

Figure 3.4.4 Typical pressure decay rate profile for SSO mission (Model 4S fairing)

A

3.5 Contamination and Cleanliness

3.5.1 Prelaunch contamination and cleanliness

The launch vehicle hardware which may affect the spacecraft environment shall be 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 (PLF). In addition, operation activities at the launch site from arrival at the STA2 (or STA1) and air supplied to the spacecraft are also controlled strictly according to the contamination control guidelines.

(1) Contamination control before spacecraft encapsulation

Surfaces of the contamination critical hardware items are cleaned and inspected to maintain cleanliness of less than Class 100,000 conditions. Supplied gas (air) is also controlled at less than Class 100,000 conditions through a filter.

(2) Contamination control after encapsulation

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

3.5.2 Flight contamination control

The sequence of the spacecraft separation and the collision / contamination avoidance maneuver is carried out as described in § 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 nonpropulsive, GOX is expelled from the LOX tank through the vent port (nonpropulsive), GH_2 is expelled from the LH_2 tank through the nonpropulsive 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 noncontaminating to the spacecraft. If necessary, a contamination analysis is carried out. Outgases from A
inside the fairing, which are measured per ASTM E595-77[*1], are consistent with NASA regulation.

<Note>
***1 : Standard test method for total mass loss and collected volatile condensable materials from outgassing in a vacuum environment.**

Table 3.5.1 Source of gaseous contaminant to spacecraft and its installation location

Gas producing 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	Aft end of LH_2 tank : about 90 deg	
< 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

To ensure that electromagnetic compatibility (EMC) is achieved for each launch, the electromagnetic environment is thoroughly evaluated. The spacecraft and the launch vehicle system must 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.

3.6.1 Launch vehicle generated radio environment

Launch vehicle intentional radiations are limited to the UHF-band telemetry transmitters at 8 W (nominal), the VHF-band telemetry transmitter at 3 W (nominal), the radar transponder at 400 W peak (nominal) and the SHF-band telemetry transmitter at 2 W (nominal). C

Launch vehicle RF systems and their frequencies are as follows:

VHF telemetry system	295	to	297	MHz (standard)	C
UHF telemetry system	2.289	to	2.291	GHz (standard)	
Radar transponder system	5.23	to	5.786	GHz (standard)	
SHF telemetry system	14.855	to	14.865	GHz (option)	C

3.6.2 LV generated electromagnetic environment

When the launch vehicle is not transmitting any radio signals, the H-IIA radiated emission level is below that shown in Figure 4.6.8. Detailed information related to RF requirements is presented in § 4.6.8.2. C

This level is defined at the spacecraft separation plane (at the lower spacecraft separation plane in a dual launch). A

3.7 Spacecraft Compatibility Test Requirements

NASDA 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 protoflight 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 protoflight article.

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

Flight hardware fit checks 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 2 minutes	Limit + 3 dB 1 minutes	Limit Level 1 minutes
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

Table 3.7.2 Spacecraft qualification and acceptance tests requirement

	Acoustic	Shock	Sine Vib	EMI / EMC	Modal Survey	Static Loads	Fit Check
Qualification	✓	✓	✓	✓	✓	✓	
Acceptance	✓		✓		✓		✓

CHAPTER 4.

INTERFACE REQUIREMENTS

CHAPTER 4. INTERFACE REQUIREMENTS

4.1 General

This chapter describes interface requirements between the spacecraft and the H-IIA launch vehicle. In this chapter, frequency requirements, balance requirements, mechanical requirements, electrical and radio requirements are included. All interface information given is a baseline, so slight modifications or tailoring might be permitted according to negotiation with NASDA.

4.2 Frequency Requirements

4.2.1 General

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.

Figure 4.2.1 shows the H-IIA launch vehicle axes used in this document for reference. A

4.2.2 Fundamental frequencies

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

a) Lateral direction $\geq$ 10 Hz
b) Longitudinal direction $\geq$ 30 Hz

If the spacecraft can not satisfy the above conditions, the spacecraft organization should discuss loads, environmental conditions and usable volume, etc., with NASDA using a result of coupled loads analysis at the interface meetings, and confirm that there are no problems. A

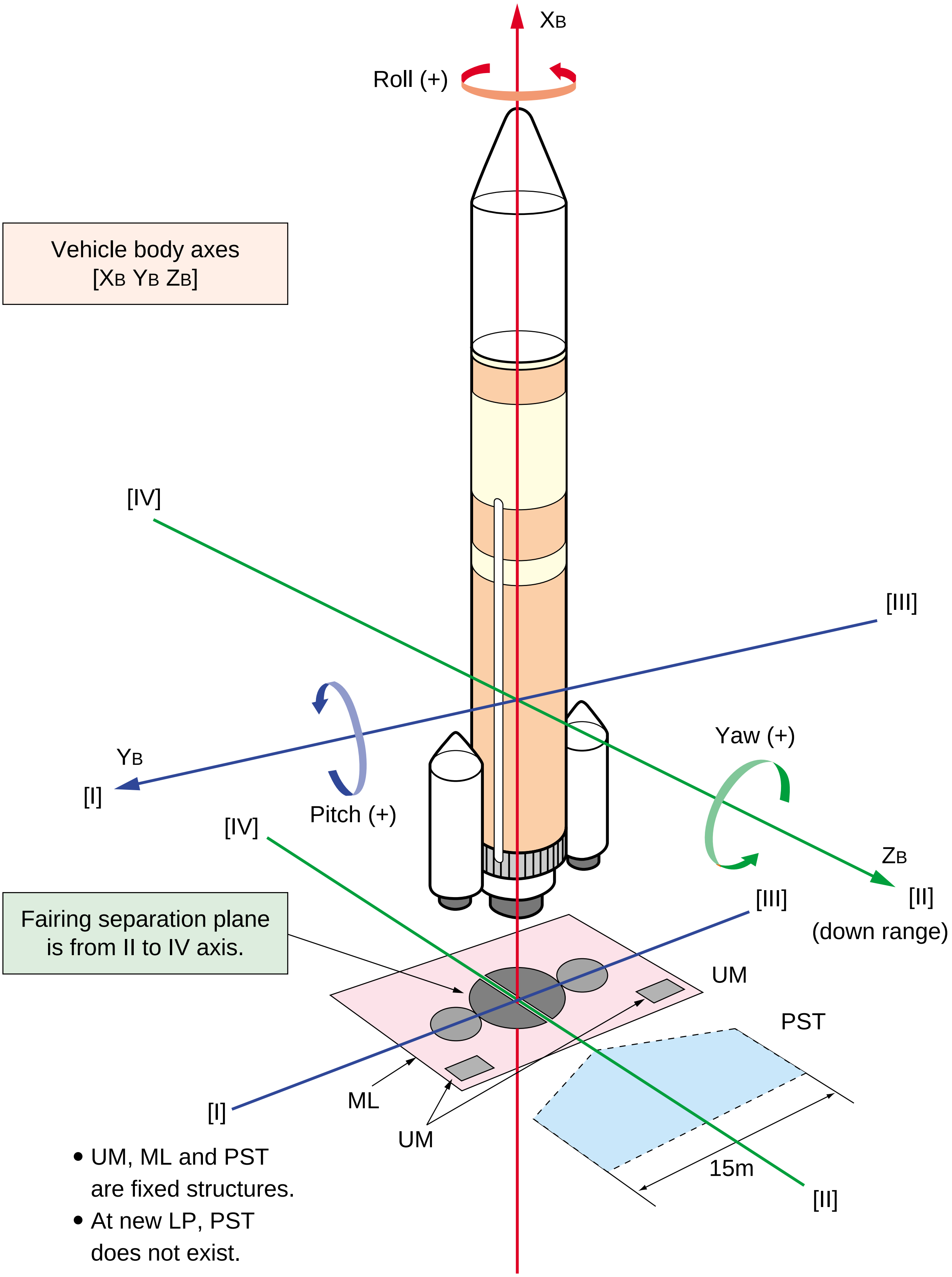


Figure 4.2.1 H-IIA launch vehicle axes

A

4.3 Balance Requirements

4.3.1 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 concerns a disturbance at the spacecraft separation and spin-up unbalance for the spun-up spacecraft.

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

4.3.2 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.

4.3.3 Balance requirements

4.3.3.1 Static balance

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

a) Separation without spin-up
   Less than 25 mm from the launch vehicle center axis in radial direction

b) Separation with spin-up (less than 5 rpm)
   Less than 25 mm from the launch vehicle center axis in radial direction | B

If the separation mechanism with springs is not used, the spacecraft organization should coordinate with NASDA about static balance requirements.

4.3.3.2 Dynamic balance

For the spacecraft which requires to be spun-up, the angle between the principal roll inertia axis of the spacecraft and the launch vehicle roll axis (or the spacecraft spin axis) shall be less than 3°. | B

4.4 Payload Fairing

This section describes the H-IIA launch vehicle payload fairing, especially the configuration, usable volume, and mission modification of five fairing types.

The spacecraft is located at the 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 the top of the lower fairing using an adapter for the upper spacecraft.

4.4.1 Fairing types

There are 6 standard payload fairing models. C

a) 4 m diameter single launch model (Model 4S)

b) 5 m diameter single launch model (Model 5S)

c) 4 m diameter dual launch model (Model 4/4D-LS)
  — upper : long type / lower : short type

d) 4 m diameter dual launch model (Model 4/4D-LC)
  — upper : long type (same as Model 4/4D-LS) / lower : clamshell type

e) 5 m / 4 m diameter dual launch model (Model 5/4D)

f) 5 m diameter single launch model (Model 5S-H) C

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

Table 4.4.1 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 § 4.2.

4.4.1.1 Model 4S

This payload fairing is available for a single launch of 3.7 m diameter spacecraft.

Figure 4.4.1 shows the model 4S.

Figure 4.4.2 shows the usable volume of the model 4S fairing.

4.4.1.2 Model 5S

This payload fairing is available for a single launch of 4.6 m diameter spacecraft.

Figure 4.4.3 shows the model 5S.

Figure 4.4.4 shows the usable volume of the model 5S fairing.

4.4.1.3 Model 4/4D-LS

This payload fairing is available for a dual launch of 3.7 m diameter spacecraft.

Figure 4.4.5 shows the model 4/4D-LS.

Figure 4.4.6 shows the usable volume of the 4/4D long upper fairing.

Figure 4.4.7 shows the usable volume of the 4/4D short lower fairing.

4.4.1.4 Model 4/4D-LC

This payload fairing is available for a dual launch of 3.7 m diameter spacecraft.

Figure 4.4.8 shows the model 4/4D-LC.

Figure 4.4.9 shows the usable volume of the 4/4D clamshell lower fairing.

4.4.1.5 Model 5/4D

This payload fairing is available for a dual launch of 4.6 m diameter and 3.7 m diameter spacecraft.

Figure 4.4.10 shows the model 5/4D.

Figure 4.4.11 shows the usable volume of the model 5/4D upper fairing.

Figure 4.4.12 shows the usable volume of the model 5/4D lower fairing.

4.4.1.6 Model 5S-H

C

This payload fairing is available for a single launch of 4.6 m diameter spacecraft. This payload fairing is developed for the launch of HTV.

Figure 4.4.13 shows the model 5S-H.

Figure 4.4.14 shows the usable volume of the model 5S-H fairing.

4.4.2 Stay out zone around the payload adapter

Figure 4.4.15 (1/3) shows the general configuration of the 1194M adapter, Figures 4.4.15 (2/3) and (3/3) show the stay-out zone around the 1194M adapter. C

4.4.3 Large door

After encapsulation of the spacecraft in the fairing, the user can enter the fairing through a large door prepared for spacecraft operations, but this door is optional and the user should discuss with NASDA about the number and locations of this door.

Standard size of large door : 600 mm x 600 mm

Figure 4.4.16 shows the large door.

Aside from above large door, two large doors for launch operations of the H-IIA vehicle are located at the bottom of the cylindrical part for each fairing model. C After encapsulation of the spacecraft in the fairing, the user can enter the fairing through these doors, but the user should discuss with NASDA about using these doors.

4.4.4 Mission modification

4.4.4.1 Access door

After encapsulation of the spacecraft in the fairing, the user can access the spacecraft through access doors.

Standard size of access door : 450 mm in diameter (ø 450)
Optional size of access door : 600 mm in diameter (ø 600)
Standard number of the access doors of ø 450
Four for model 4S, 5S, 5S-H, 4/4D and 5/4D upper fairings
Two for model 4/4D and 5/4D lower fairings

Figure 4.4.17 shows the access door of ø 450
Figure 4.4.18 shows the access door of ø 600
Allowable Access door areas of ø 450 in case of several model fairing are shown in Figure 4.4.19 to Figure 4.4.26
Allowable Access door areas of ø 600 in case of several model fairing are shown in Figure 4.4.27 to Figure 4.4.34 C

The distance between centers of access doors shall be more than 1,000 mm.
Access doors shall normally be located at the cylindrical section of the payload fairing.

4.4.4.2 Umbilical connectors

Restrictions of locating the umbilical connectors are as follows:

a) Center angle from I and III axes of launch vehicle axes shall be less than 14°. A
b) Standard number of umbilical connectors :
2 for model 4S, 5S, 5S-H and upper fairing of model 4/4D and 5/4D C

Details of umbilical connectors are described in § 4.6.6.
If the spacecraft provider requires it, the umbilical connectors can be mounted on the interface plane of the payload adapter.

4.4.4.3 Transparent window

After encapsulation of the spacecraft in the fairing, the user can link the spacecraft and the ground stations with the radio signals through a transparent window.

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

Figure 4.4.35 shows the transparent window.

It may be installed in the same area as the access door.

Instead of using a transparent window, users can link the spacecraft and the ground stations with the radio signals directly with an internal / external antenna connected through a coaxial cable. It can transmit and receive radio signals before and after lift-off as a substitute for transparent windows.

Standard number of internal / external antennas : zero

4.4.4.4 Internal antenna

After encapsulation of the spacecraft in the fairing, the user can link the spacecraft and the ground stations using the antenna inside the fairing. It can receive radio signals from the spacecraft and can transmit radio signals from the ground stations to the spacecraft.

Standard number of internal antennas : one
Provided model 5S-H : two

Figure 4.4.36 shows typical installation of an internal antenna. The internal antenna is connected to Ground Support Equipment (GSE) through coaxial lines.

4.4.4.5 Separate air conditioning

If fairings will be used for a dual launch, an umbilical ventilation inlet, bulkhead and relief valves can be installed in the lower fairing to provide separate air conditioning between the upper and lower fairings.

Size of ventilation inlet : 400 mm diameter

Standard number of ventilation inlets :

- one for model 4S, 5S, 5S-H and upper fairing of model 4/4D and 5/4D
- one for lower fairing of model 4/4D and 5/4D

Size of relief valve : 120 mm (in case of 45 ° type of the payload support structure)
160 mm (in case of 25 ° type of the payload support structure)

Number of relief valves : two

4.4.4.6 Acoustic blankets

Acoustic blankets can be 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 a blanket are determined according to user’s requirements.

The standard blanket is 10 mm thick.

Figure 4.4.37 shows a typical configuration of an acoustic blanket.

Table 4.4.1 Characteristics of payload fairings

Items / Model	Launch	External		Portion of fairing	Usable volume		Application	
		Height (m)	Diameter (m)		Height (m)	Diameter (m)		
4S	Single	12.0	4.07	–	10.23	3.7	NASDA ETS-VI, COMETS	
5S	Single	12.0	5.1	–	9.12	4.6	NASDA ADEOS	
4/4D-LS	Dual	14.5	4.07	upper	8.23	3.7	NASA TRMM	
				lower	3.80	3.7	NASDA ETS-VII	
4/4D-LC	Dual	16.0	4.07	upper	8.23	3.7	None	
				lower	5.36	3.7	None	A
5/4D	Dual	14.1	5.1 / 4.07	upper	6.70	4.6	ISAS SFU	
				lower	4.68	3.7	NASDA GMS-5	
5S-H	Single	15.0	5.1	–	12.9	4.6	None	C

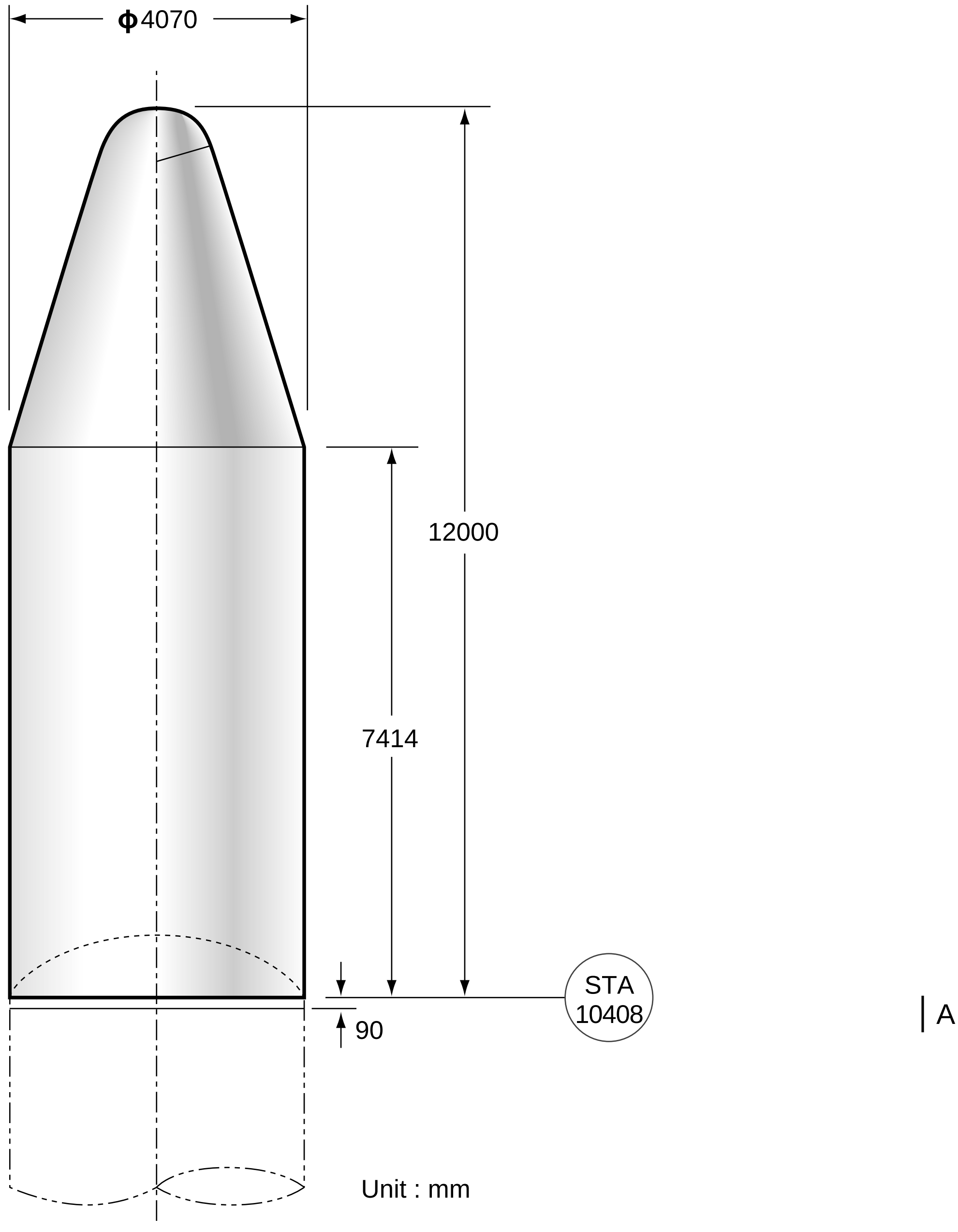
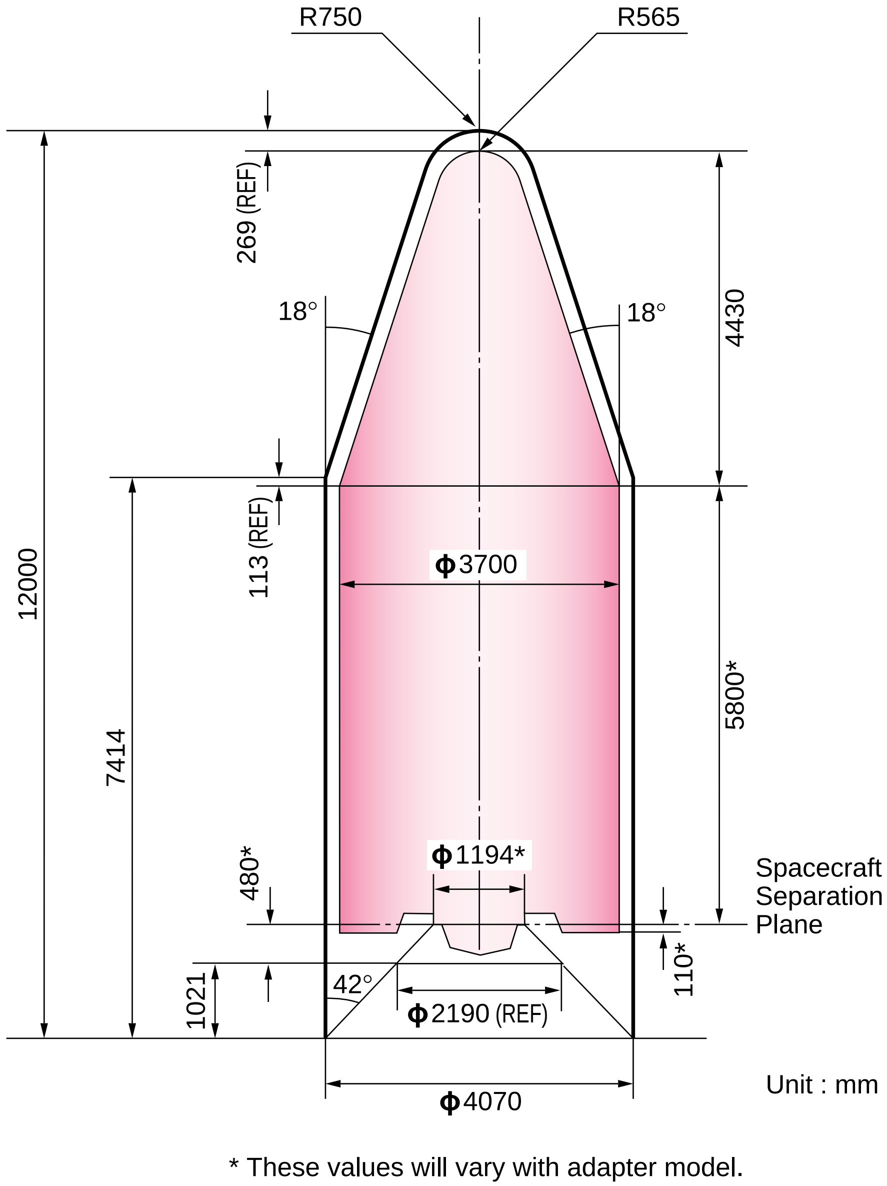


Figure 4.4.1 Model 4S



A

Figure 4.4.2 Usable volume of model 4S with 1194M adapter

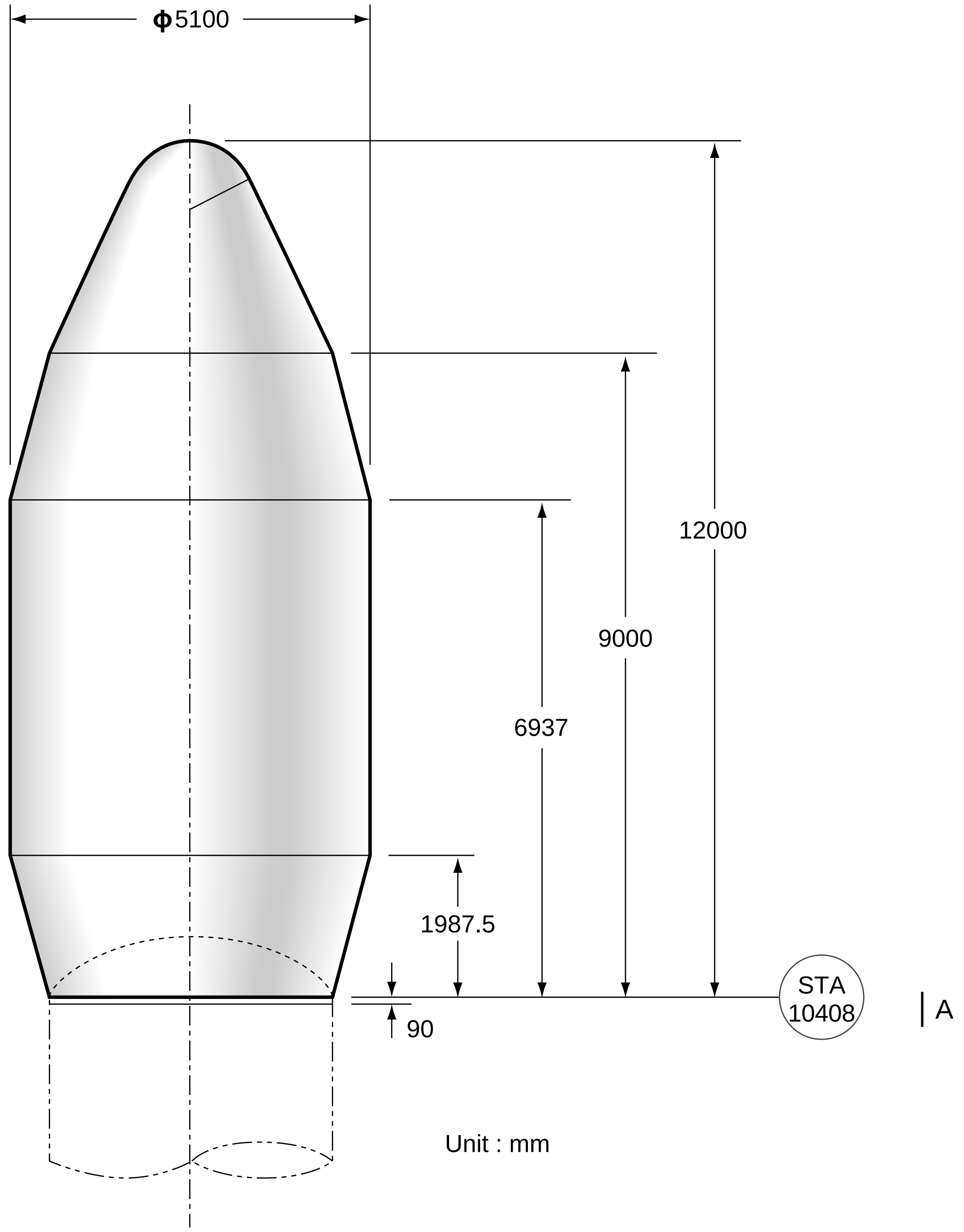
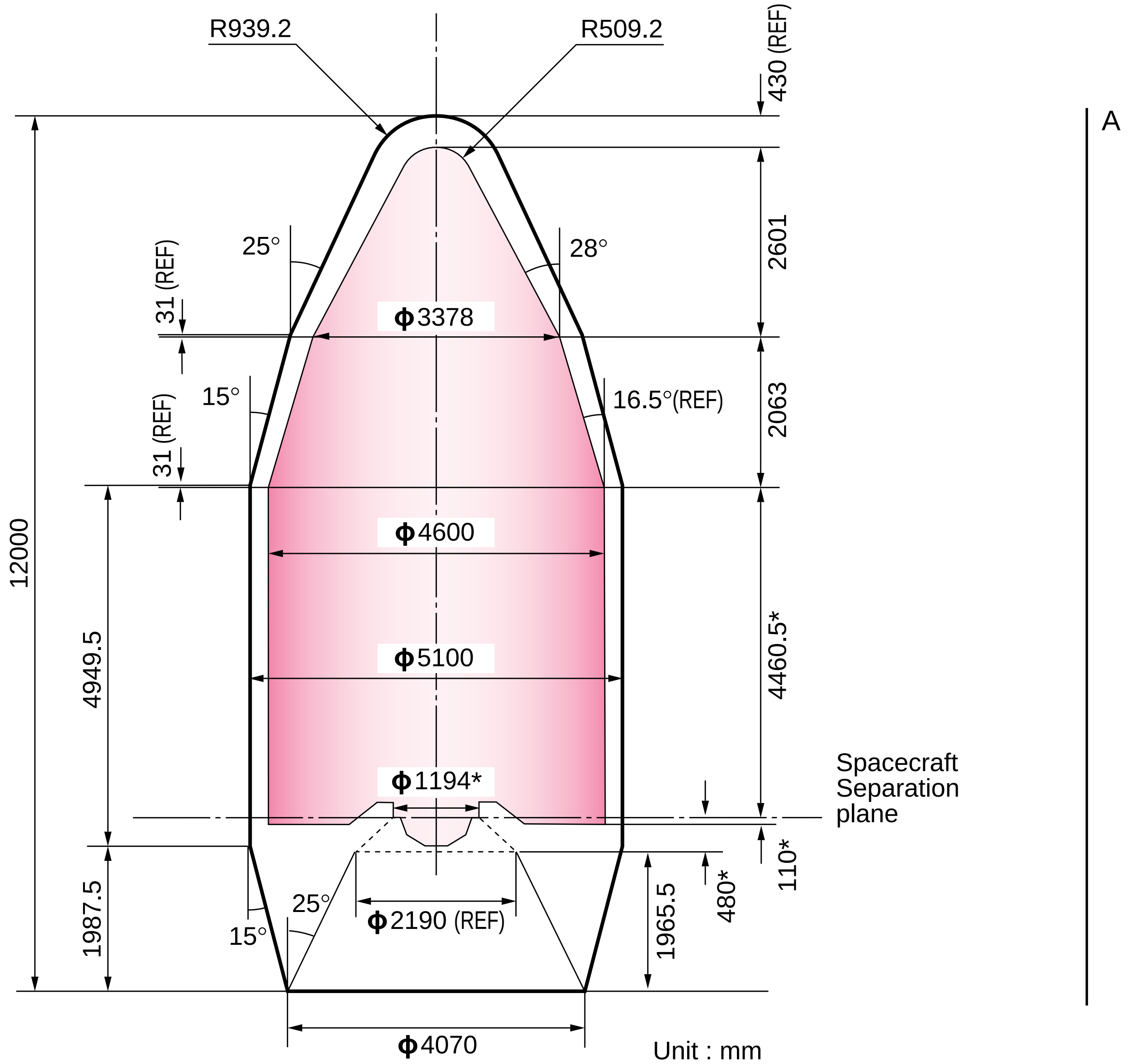


Figure 4.4.3 Model 5S



* These values will vary with adapter model.

Figure 4.4.4 Usable volume of model 5S with 1194M adapter

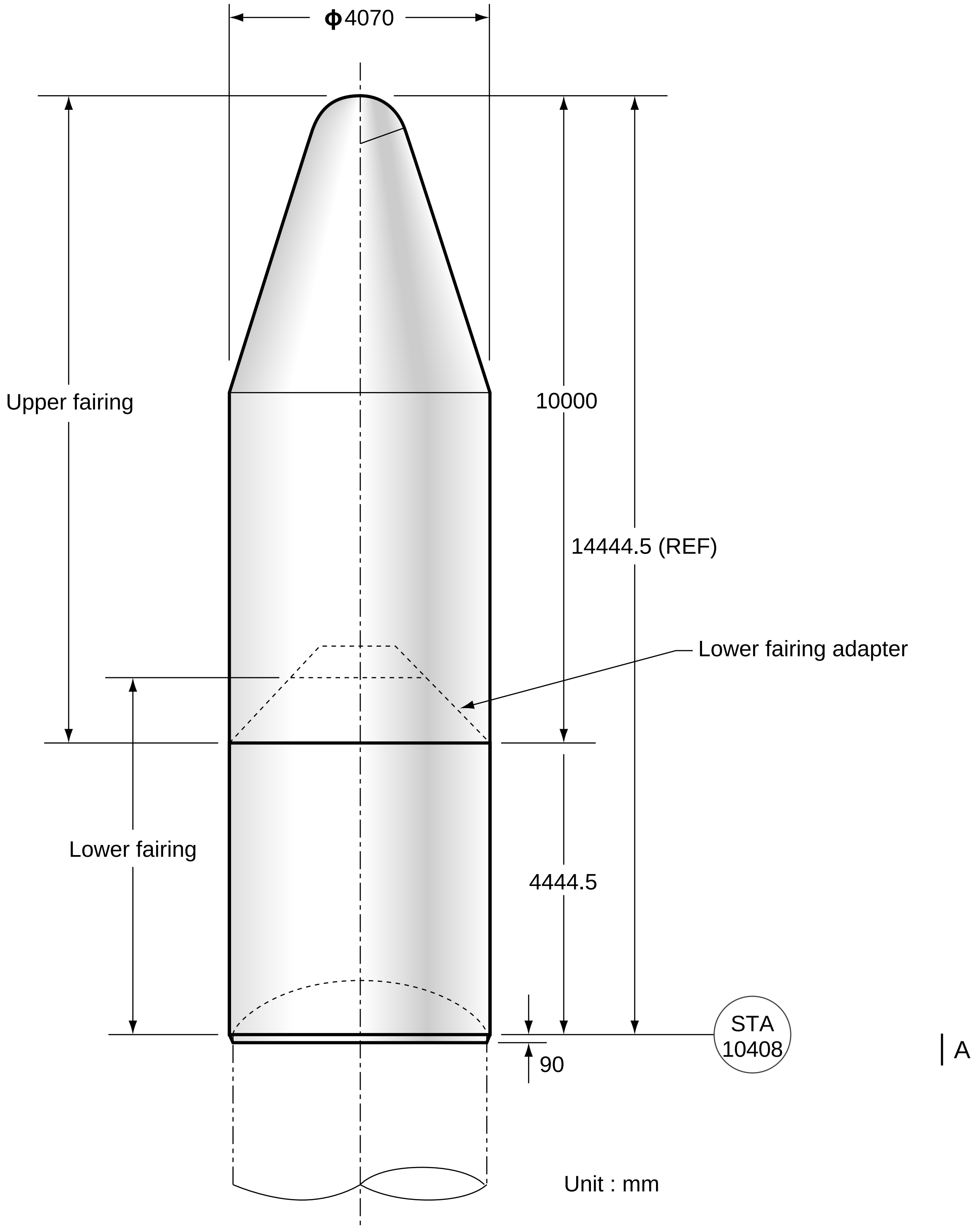
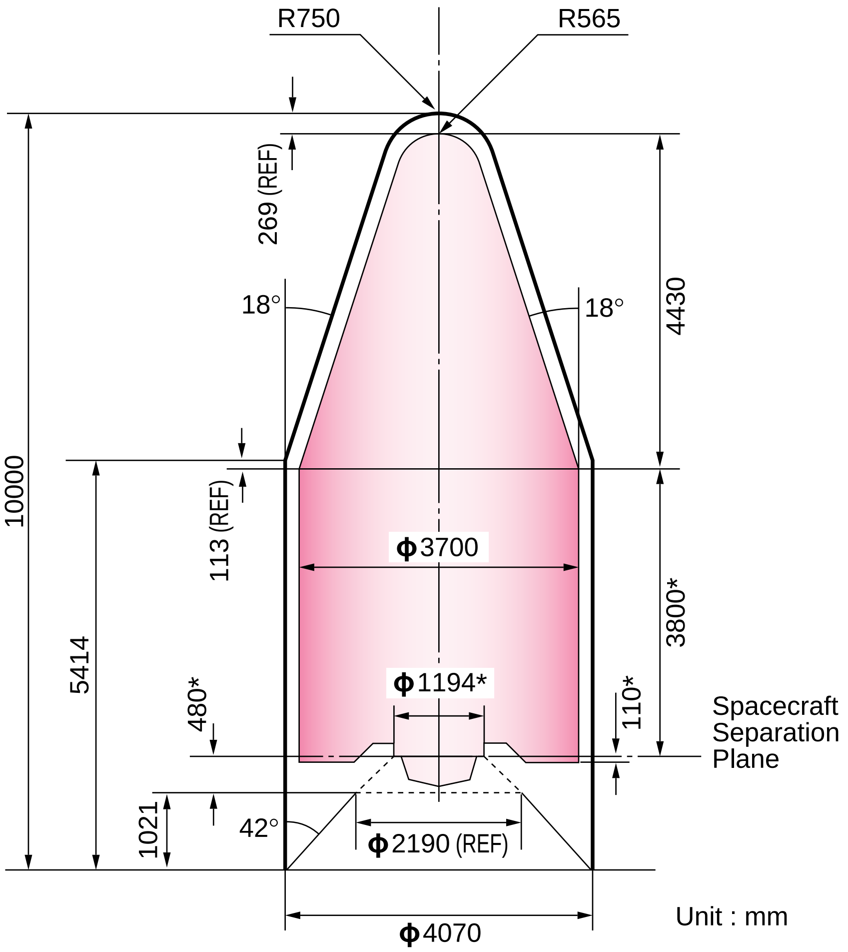


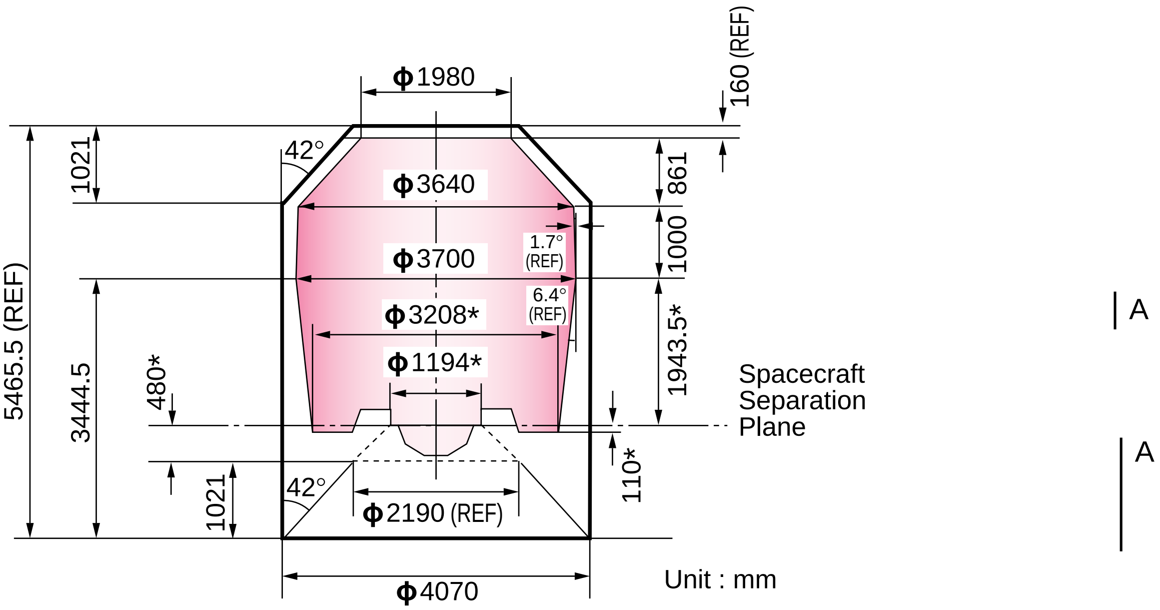
Figure 4.4.5 Model 4/4D-LS



* These values will vary with adapter model.

A

Figure 4.4.6 Usable volume of model 4/4D long upper fairing with 1194M adapter



* These values will vary with adapter model.

Figure 4.4.7 Usable volume of model 4/4D short lower fairing with 1194M adapter

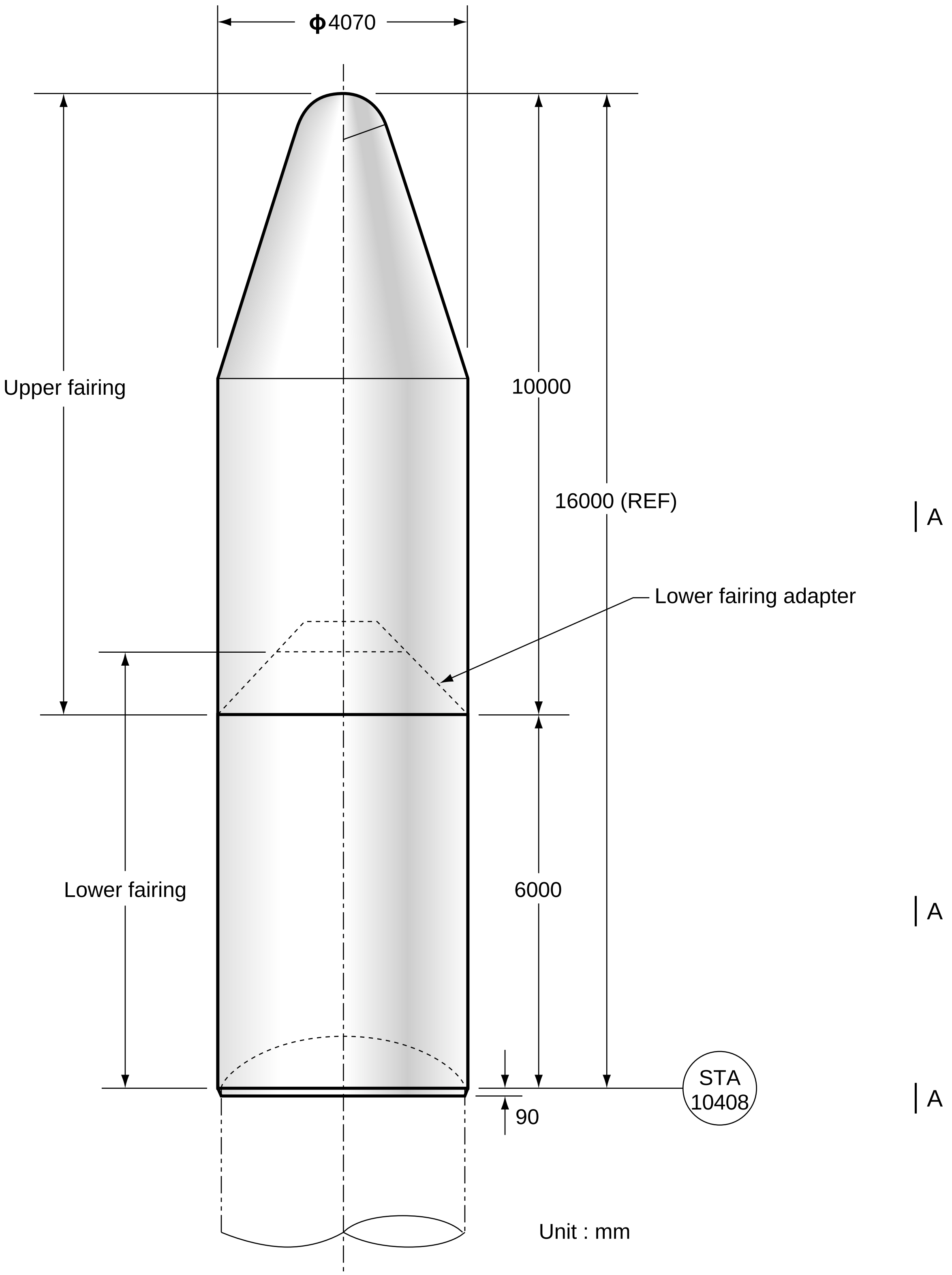
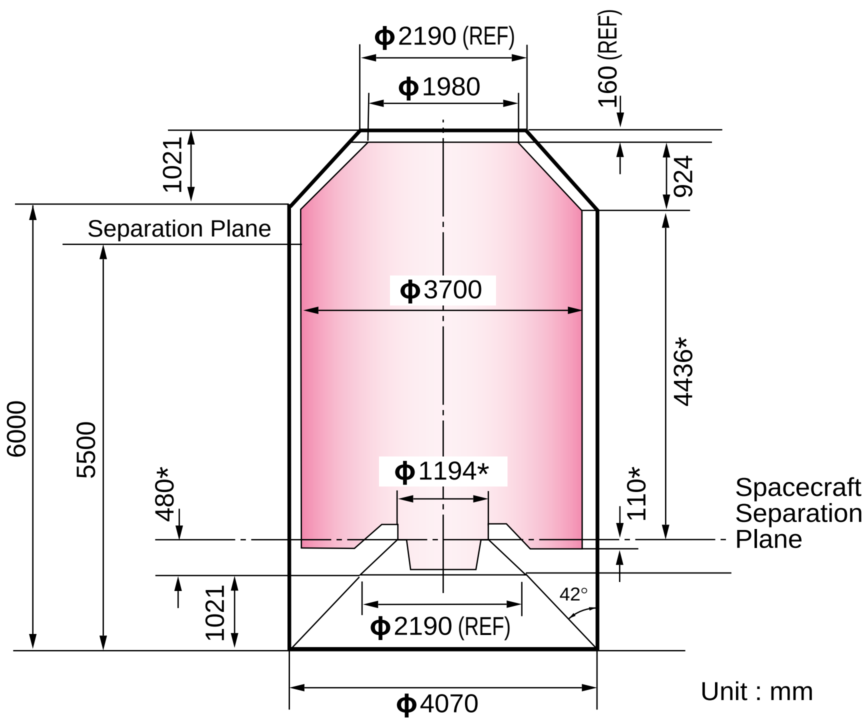


Figure 4.4.8 Model 4/4D-LC



* These values will vary with adapter model.

Figure 4.4.9 Usable volume of model 4/4D clamshell lower fairing with 1194M adapter

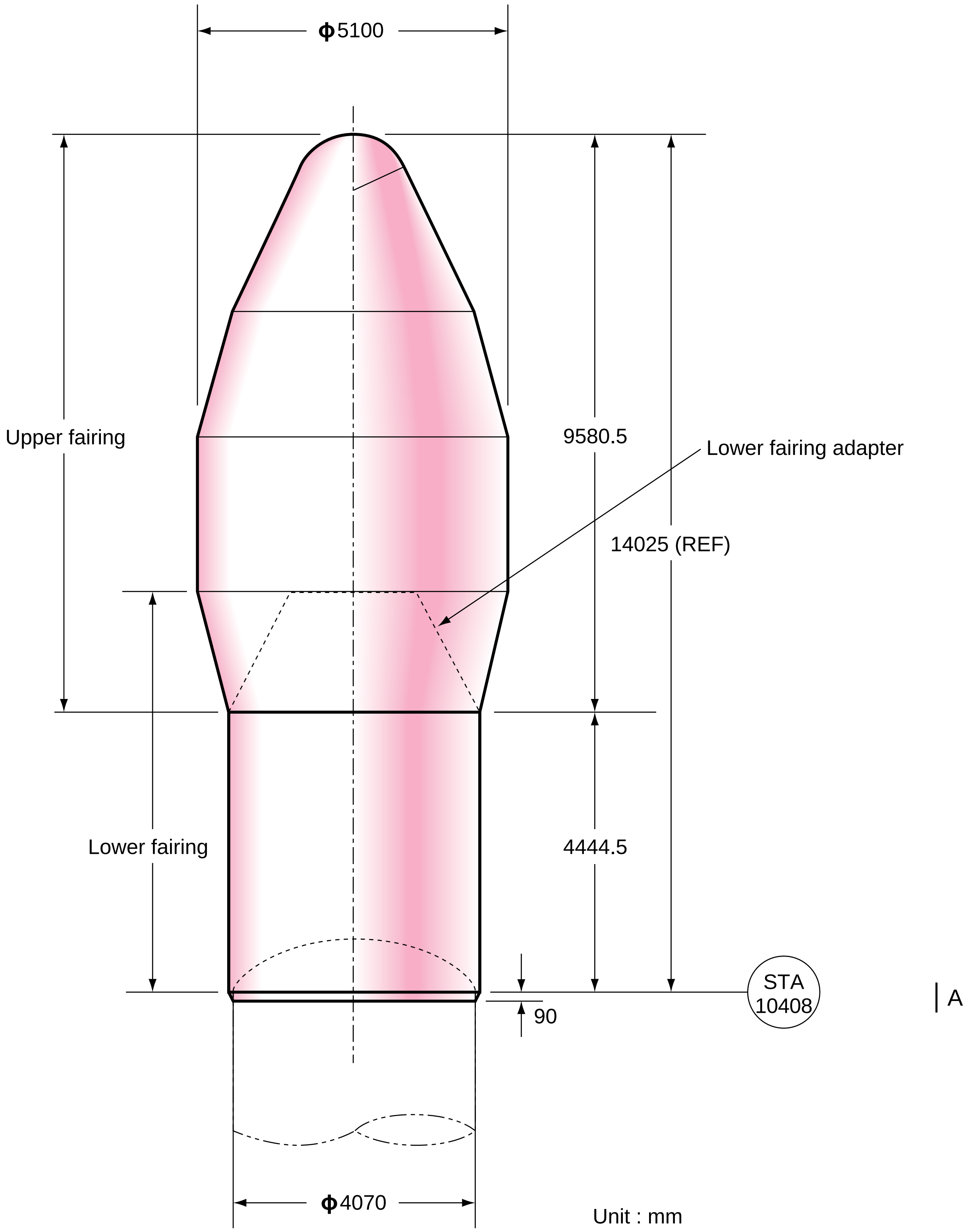
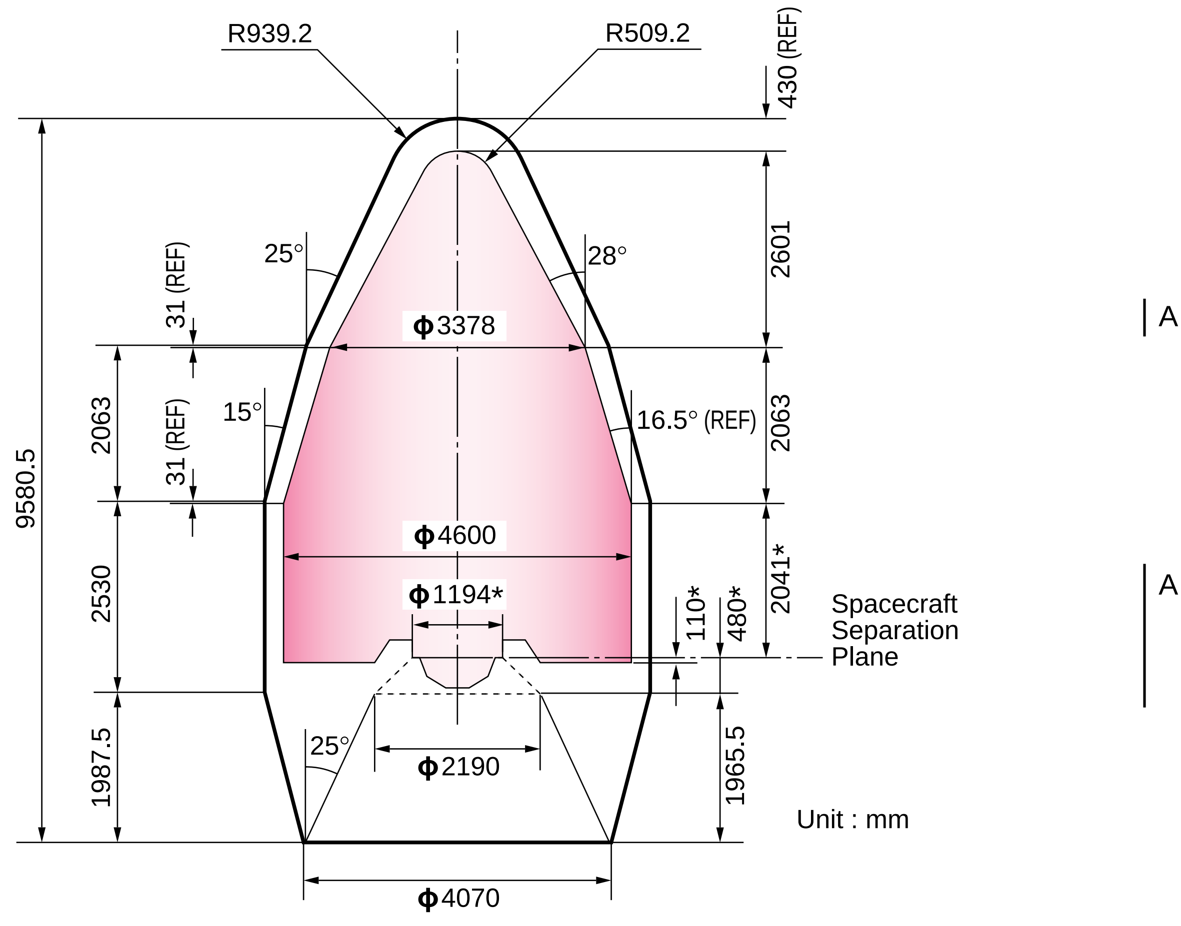
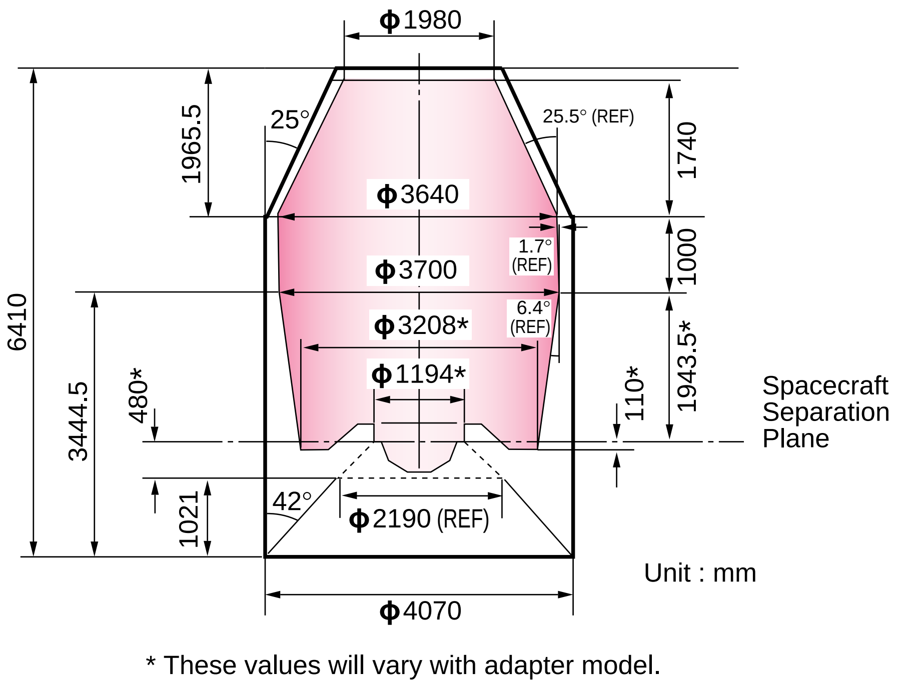


Figure 4.4.10 Model 5/4D



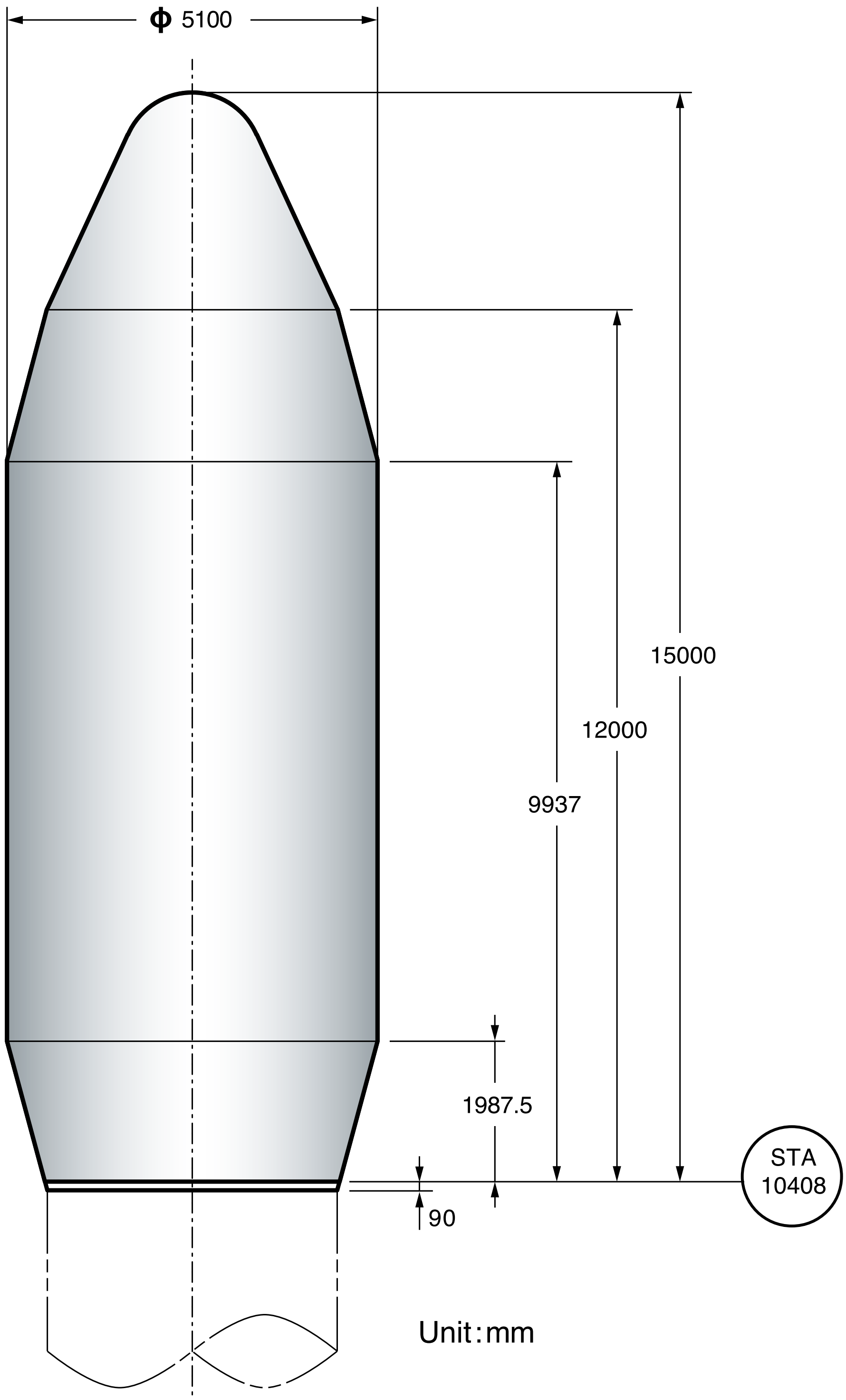
* These values will vary with adapter model.

Figure 4.4.11 Usable volume of model 5/4D upper fairing with 1194M adapter



A

Figure 4.4.12 Usable volume of model 5/4D lower fairing with 1194M adapter



C

Figure 4.4.13 Model 5S-H

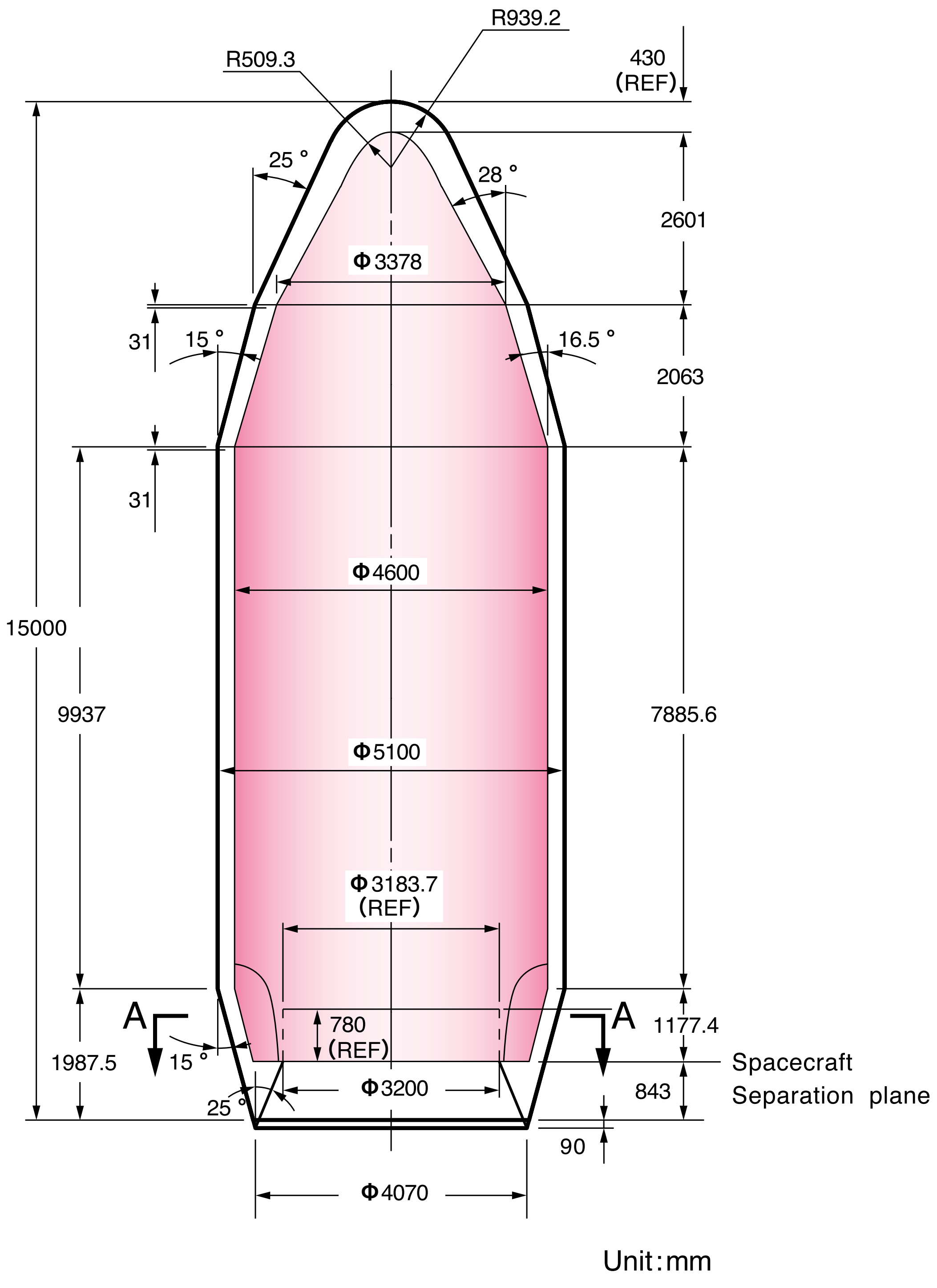


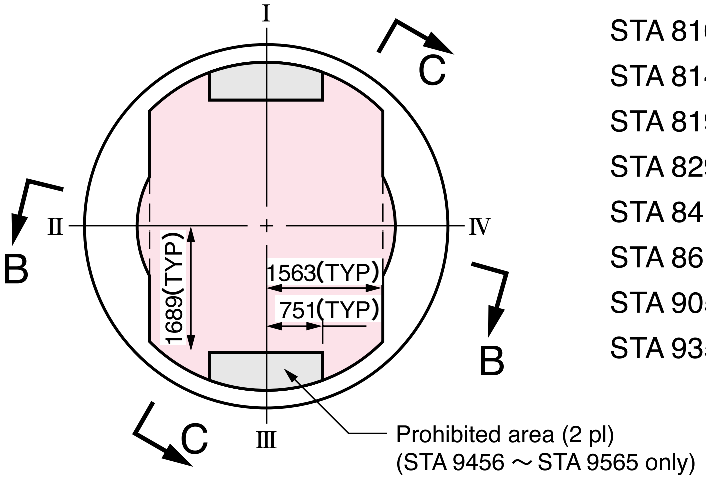
Figure 4.4.14 (1/2) Usable volume of the model 5S-H #1

C

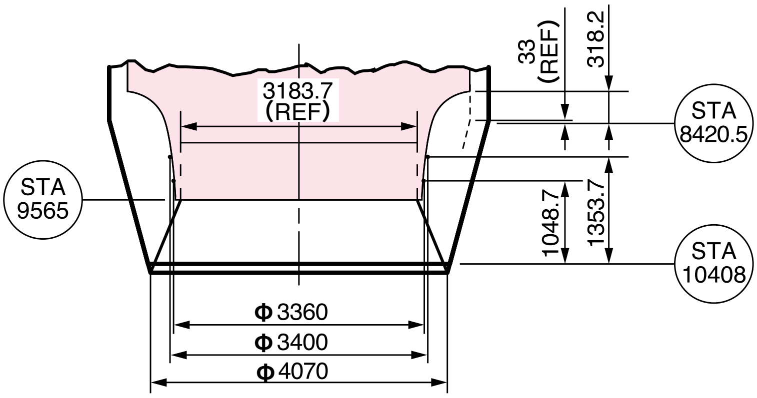
Diameter of usable volume (ϕD)

STA 8102.3 : ϕ4600 mm
STA 8140.8 : ϕ4400 mm
STA 8195.5 : ϕ4200 mm
STA 8290.9 : ϕ4000 mm
STA 8410.2 : ϕ3800 mm
STA 8610.5 : ϕ3600 mm
STA 9054.3 : ϕ3400 mm
STA 9359.3～STA 9565.0 : ϕ3360 mm

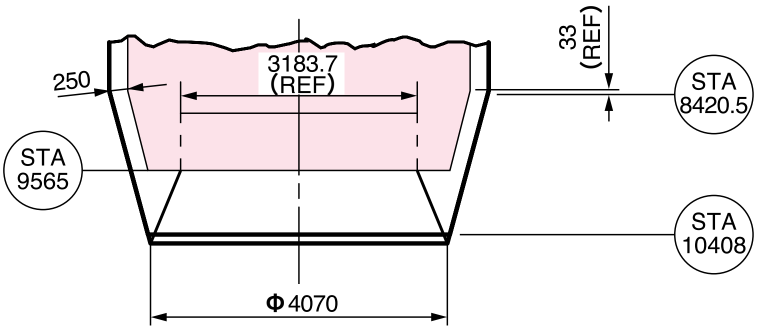
C



SECT A-A



SECT B-B



SECT C-C

Unit : mm

Note : The constraints by PAF are not included.
This usable volume is under development.

Figure 4.4.14 (2/2) Usable volume of the model 5S-H #2

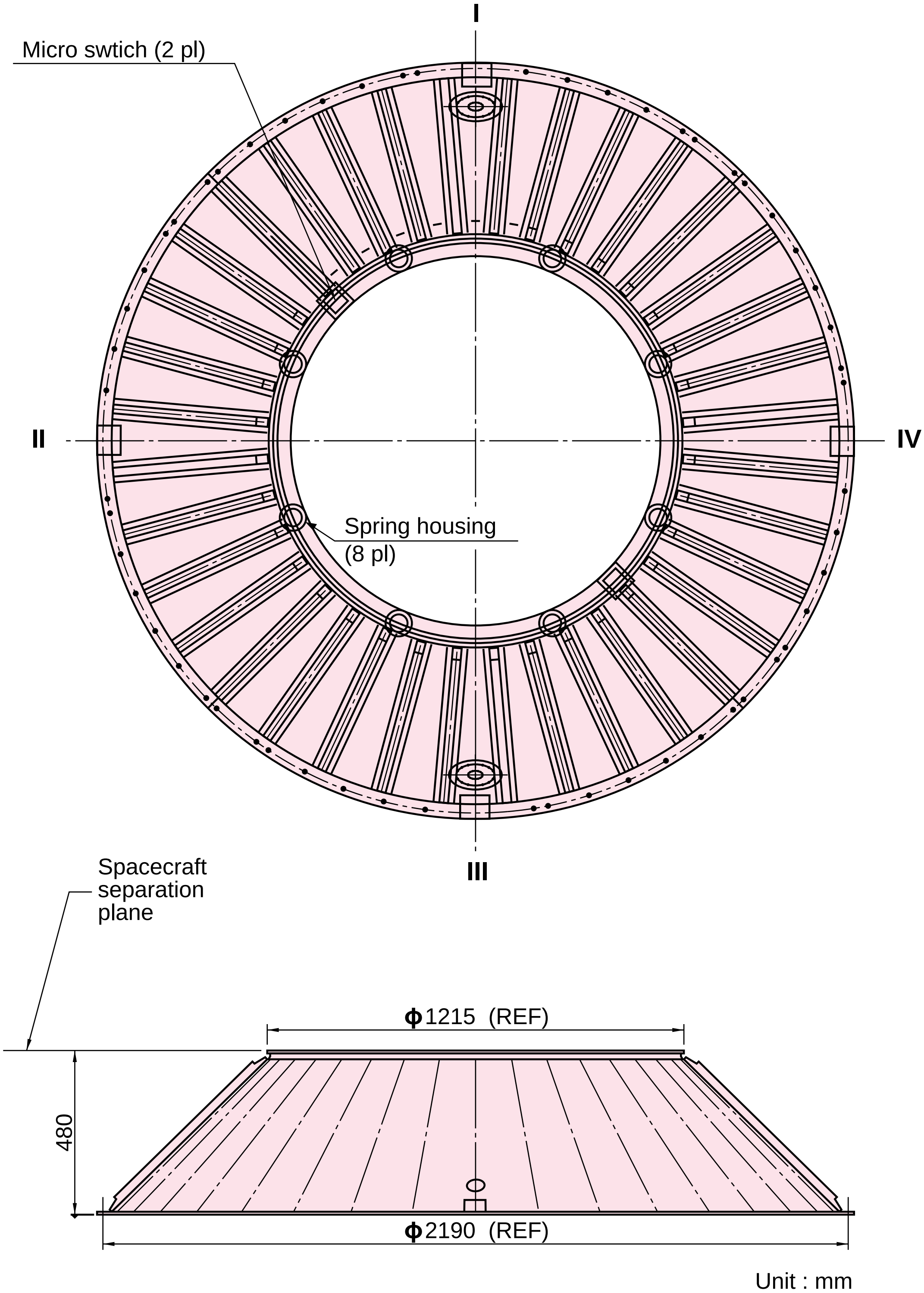


Figure 4.4.15 (1/3) General configuration of the 1194M adapter

| C

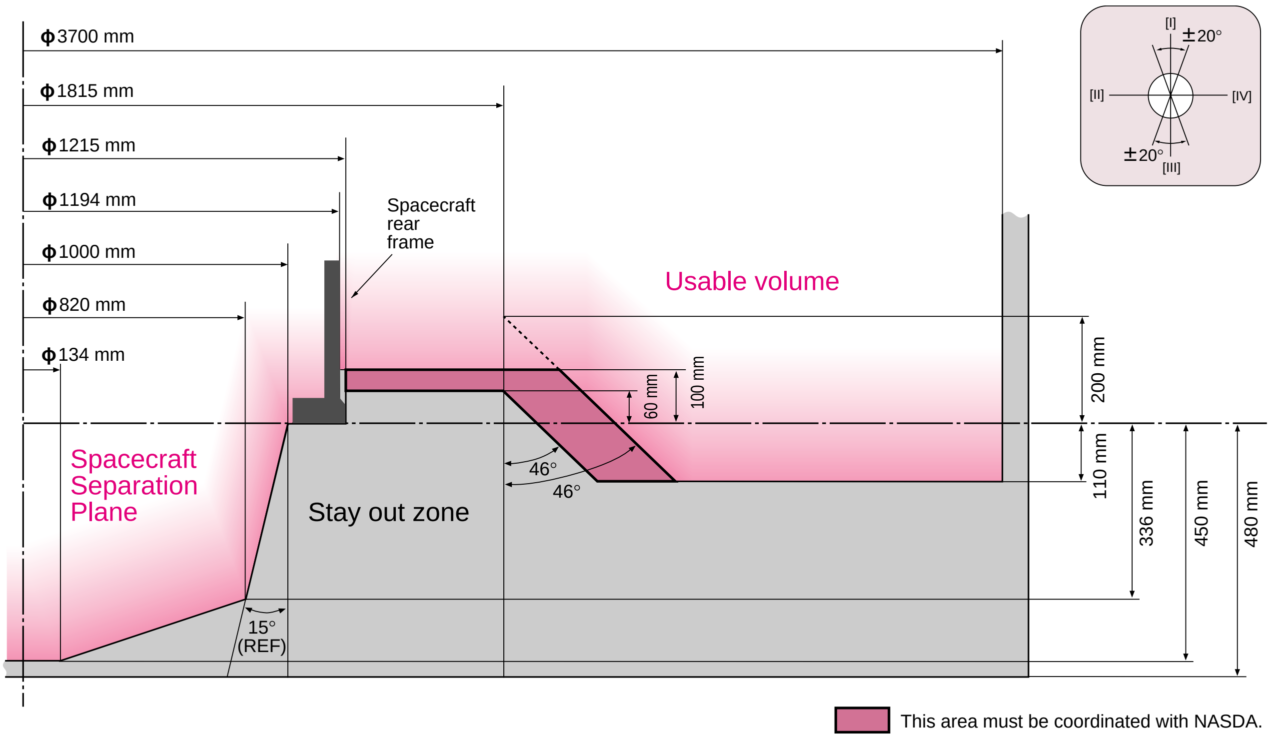


Figure 4.4.15 (2/3) Stay-out zone around the 1194M adapter (I / III -axis ±20°)

| C

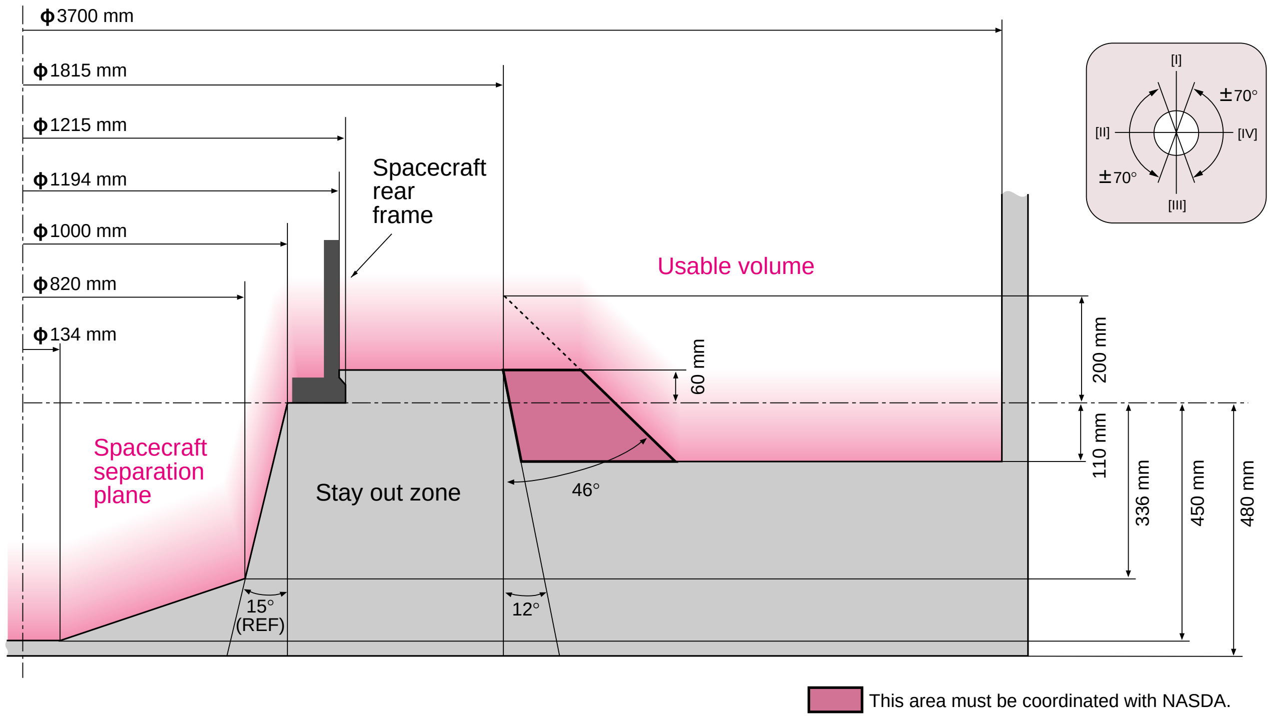
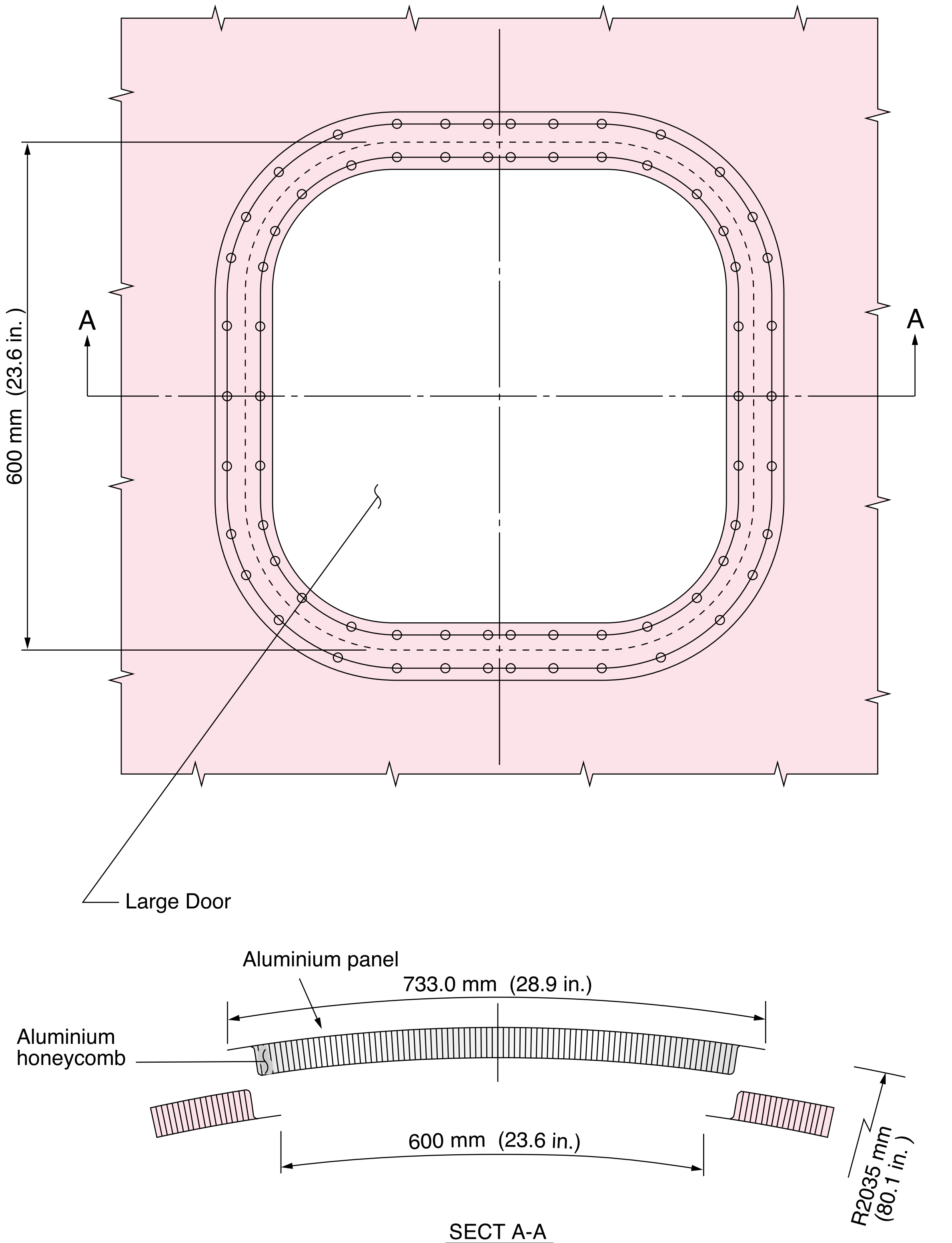
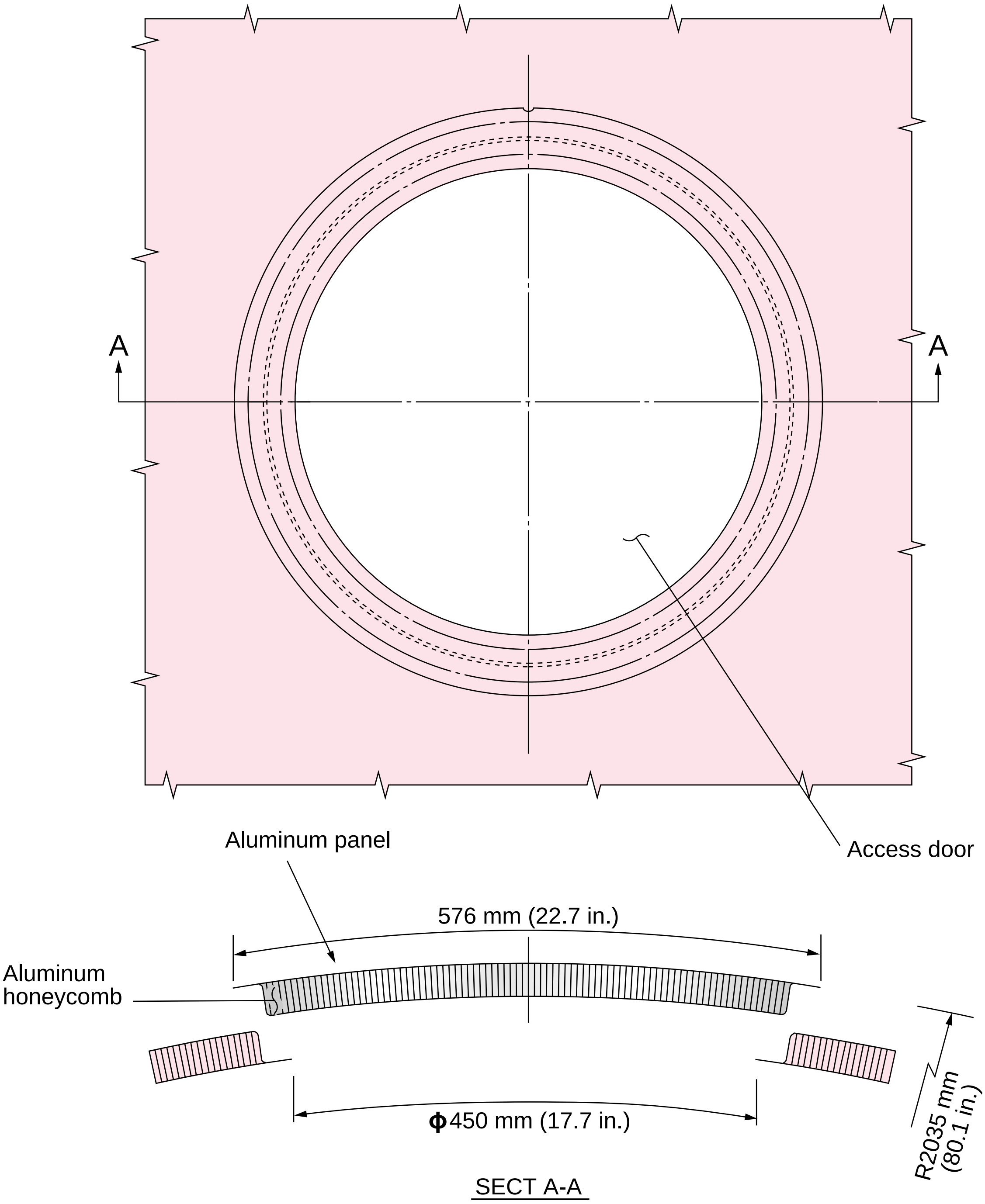


Figure 4.4.15 (3/3) Stay-out zone around the 1194M adapter (II / IV -axis ±70°) | C



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

Figure 4.4.16 Large door



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

Figure 4.4.17 ø 450 access door

| C

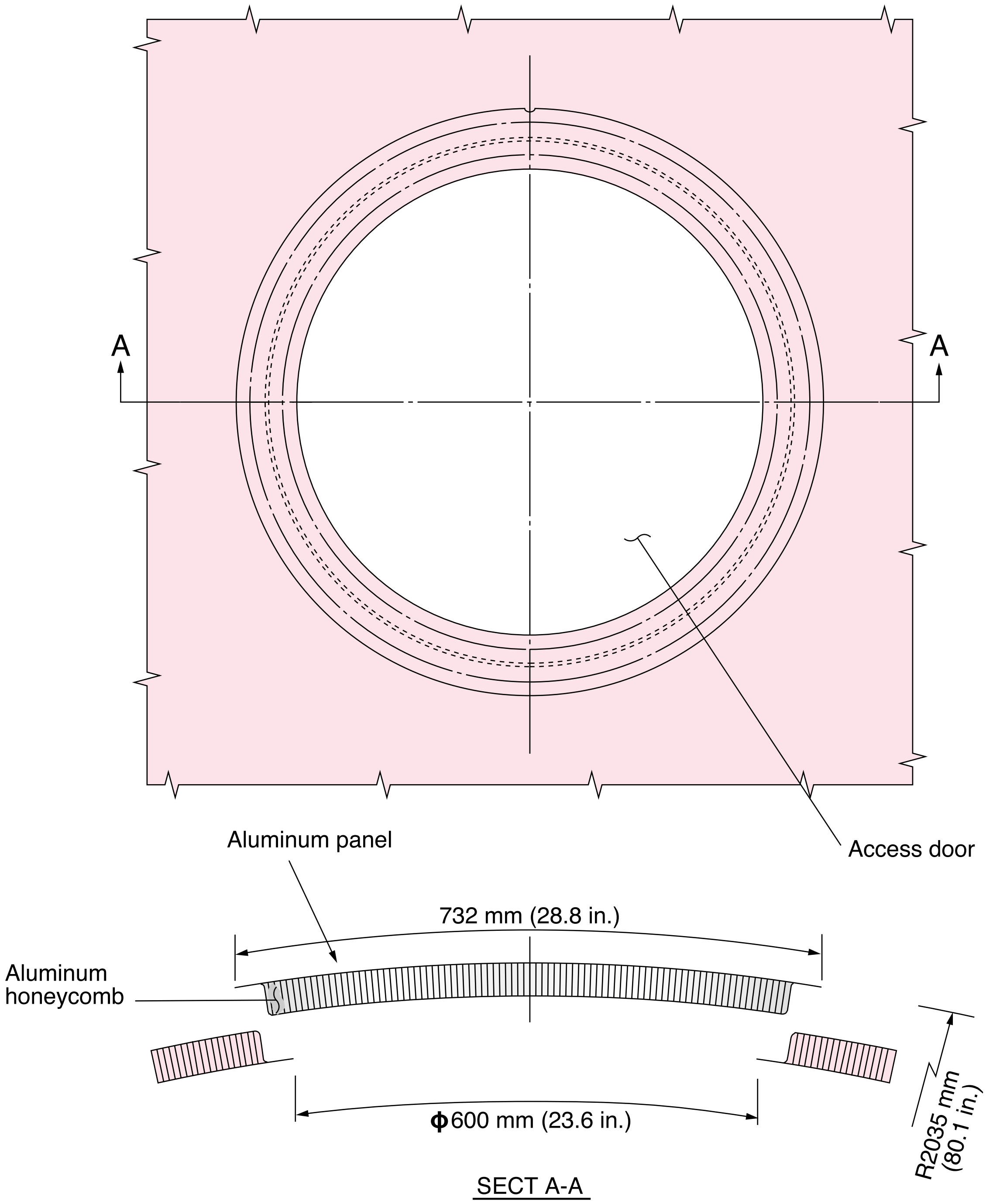
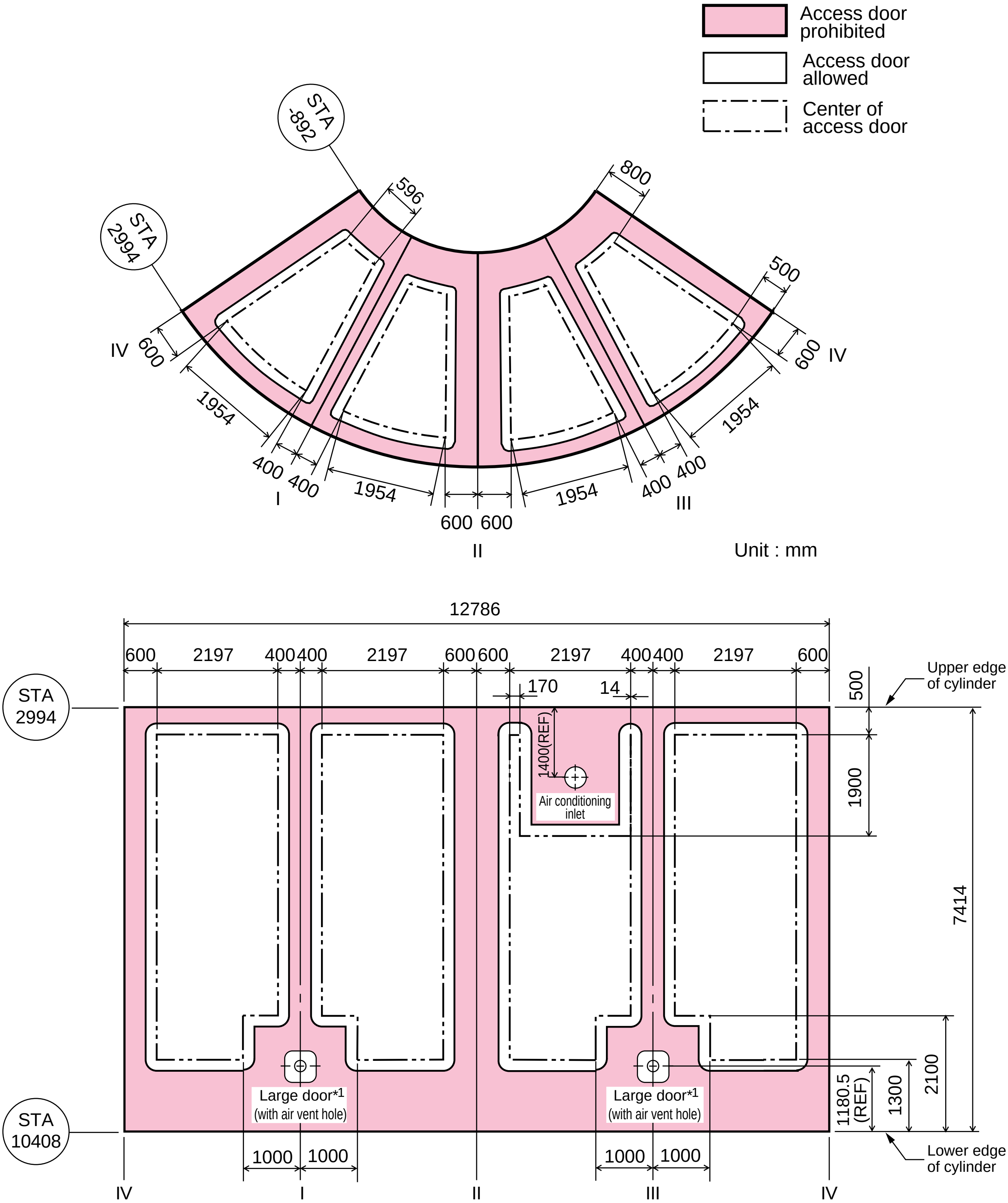


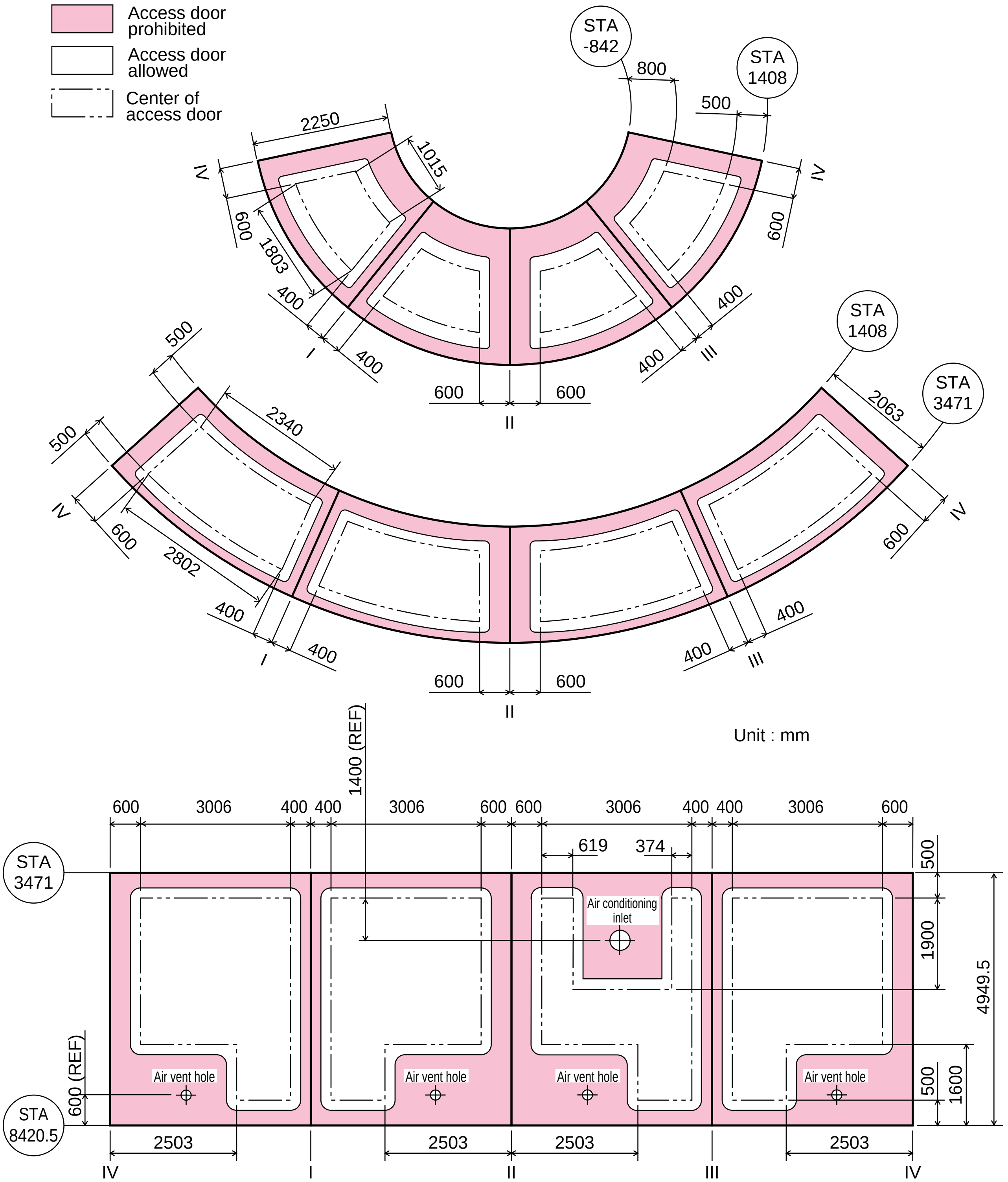
Figure 4.4.18 ø 600 access door

C



<Note>
*1 : This door is fixed and for launch vehicle operations.
*2 : Refer to the figure 4.4.2 for the payload fairing angle of conical section.

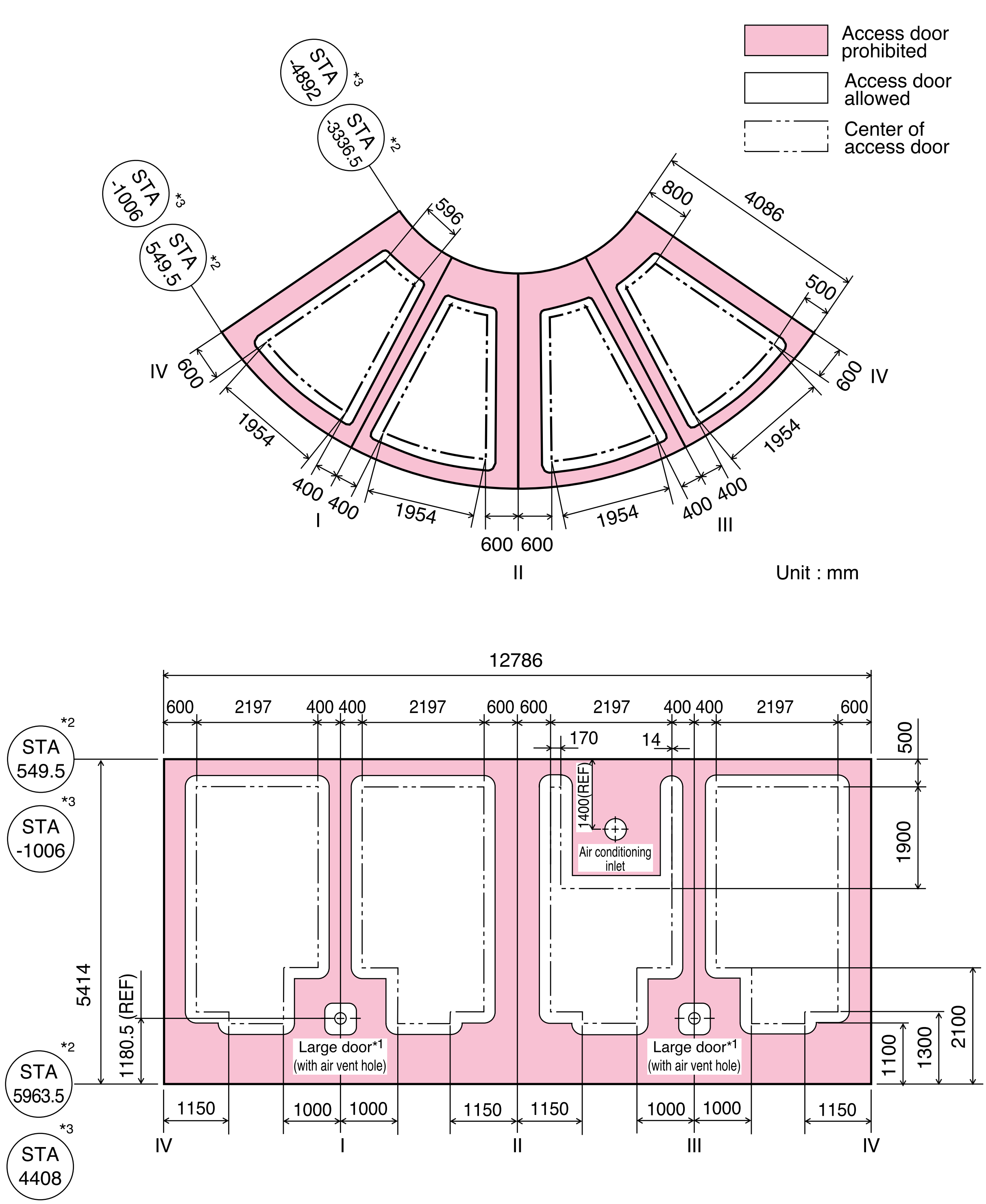
Figure 4.4.19 Allowable areas of ø 450 access door on model 4S fairing



<Note>

*1 : Refer to the figure 4.4.4 for the payload fairing angle of conical section.

Figure 4.4.20 Allowable areas of ø 450 access door on model 5S fairing



<Note>

*1 : This door is fixed and for launch vehicle operations.

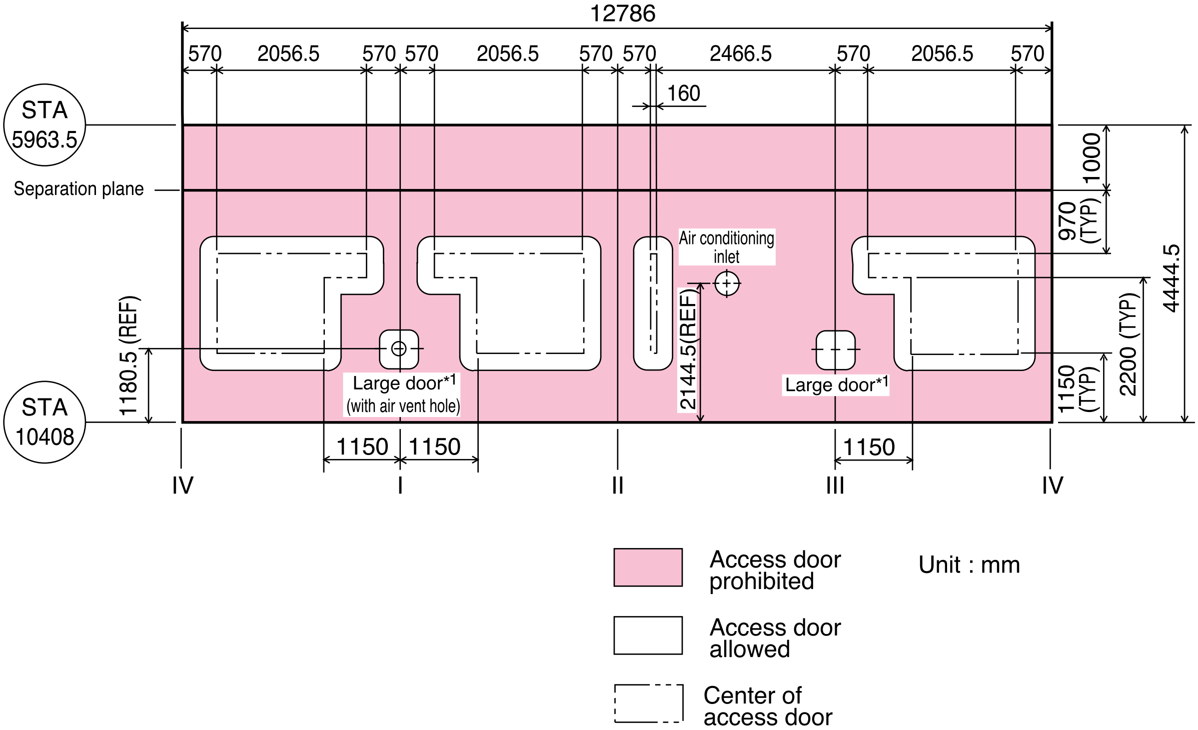
*2 : This station for the 4/4D-LS

*3 : This station for the 4/4D-LC

*4 : Refer to the figure 4.4.6 for the payload fairing angle of conical section.

Figure 4.4.21 Allowable areas of ø 450 access door on model 4/4D long upper fairing

A C A C A C

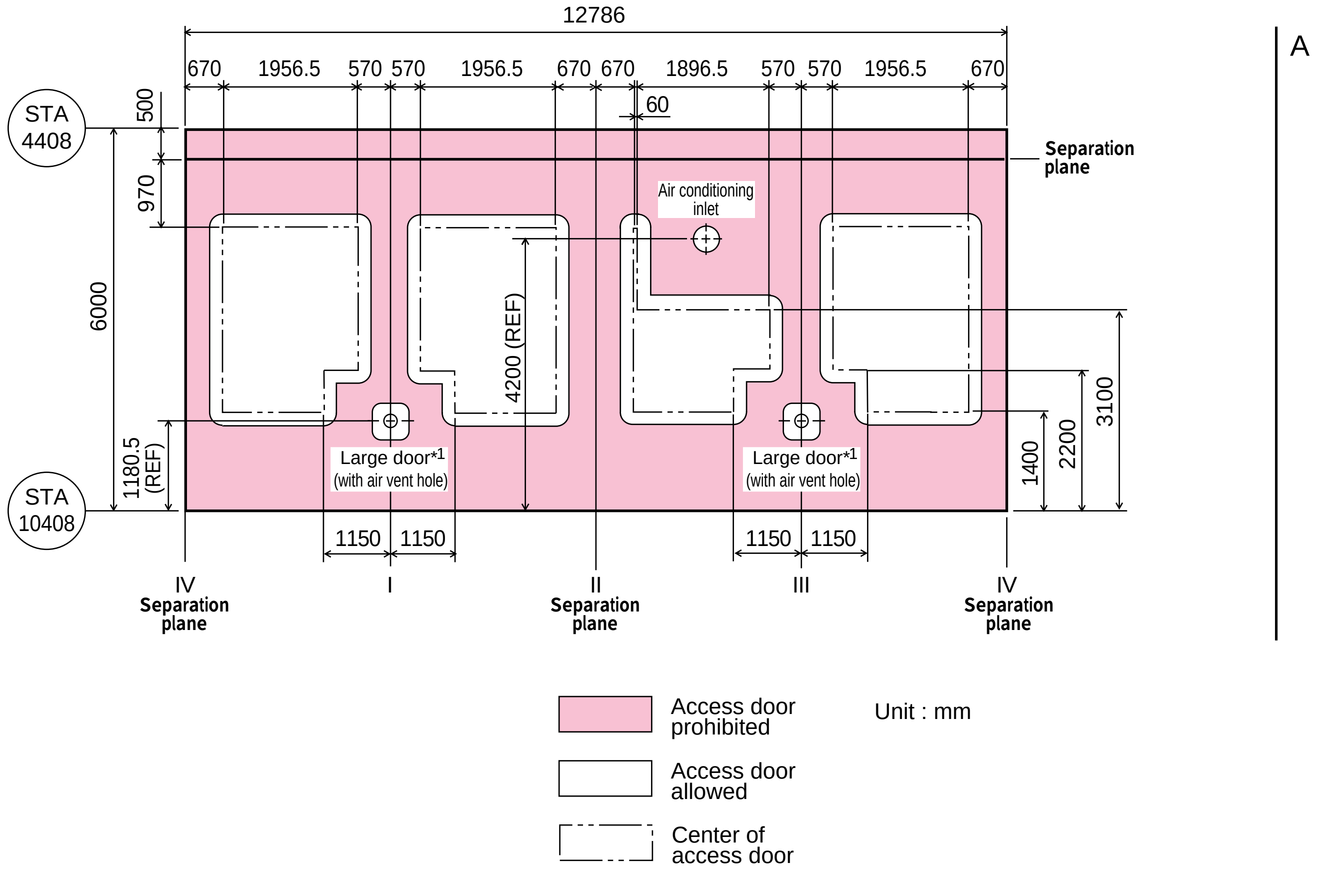


A

<Note>

*1 : This door is fixed and for launch vehicle operations.

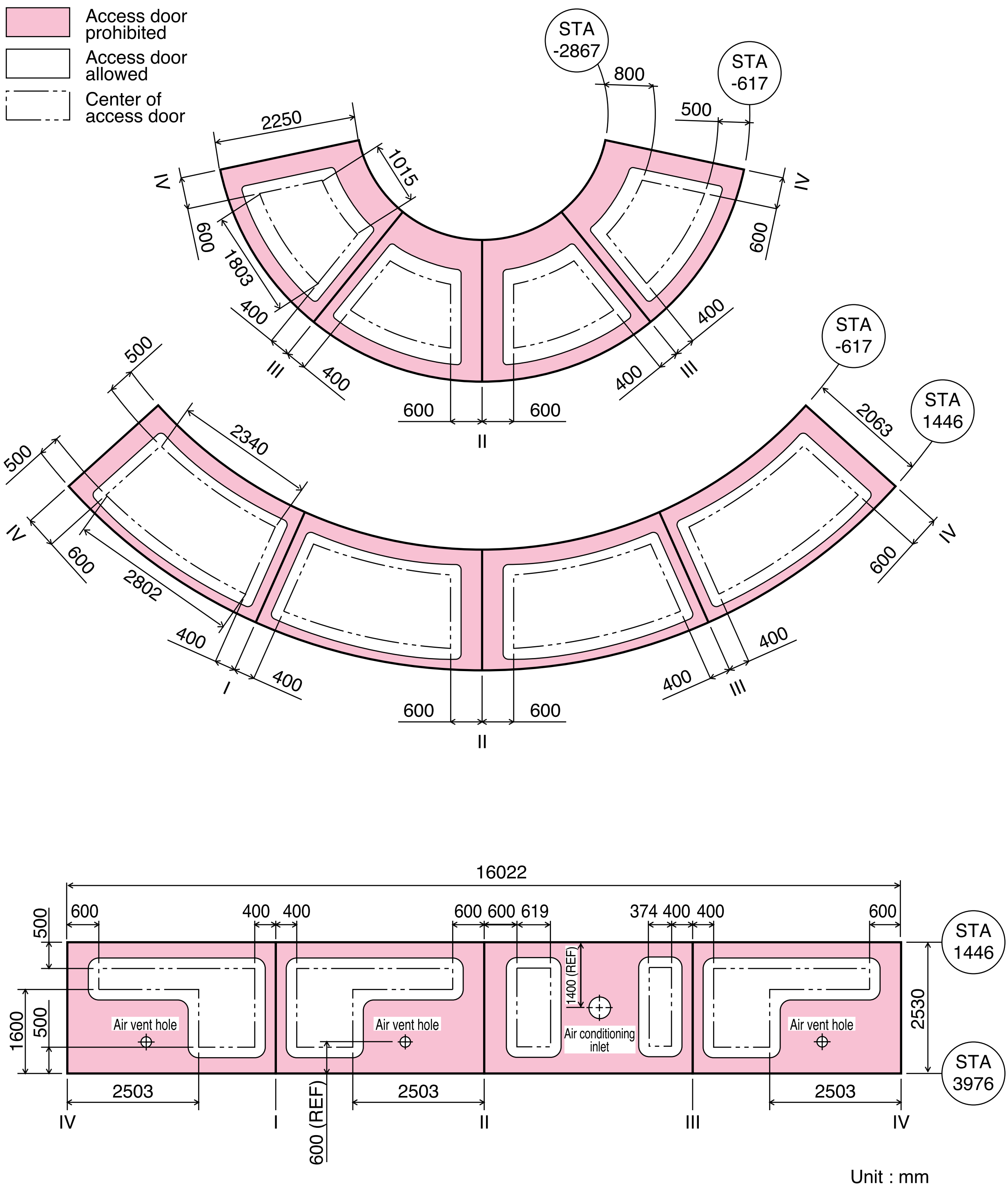
Figure 4.4.22 Allowable areas of ø 450 access door on model 4/4D short lower fairing | C



<Note>

*1 : This door is fixed and for launch vehicle operations.

Figure 4.4.23 Allowable areas of ø 450 access door on model 4/4D clamshell lower fairing | C



A

A

A

<Note>

*1 : Refer to the figure 4.4.11 for the payload fairing angle of conical section.

C

Figure 4.4.24 Allowable areas of ø 450 access door on model 5/4D upper fairing

C

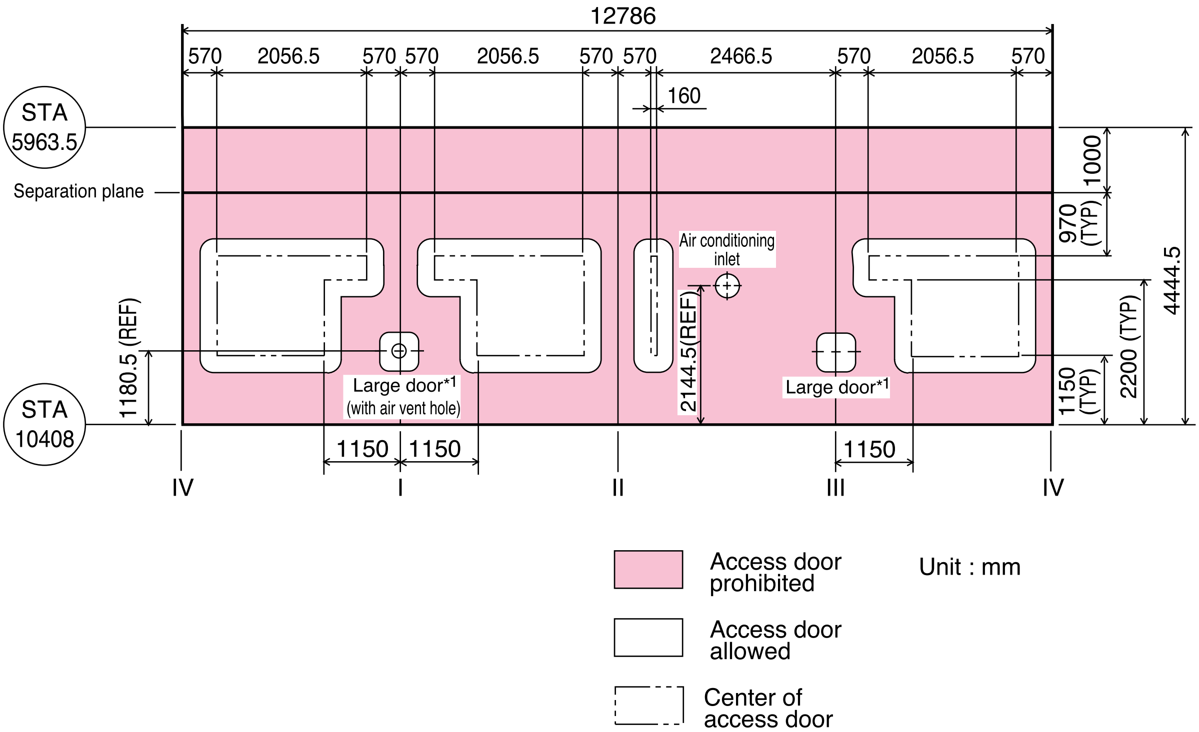
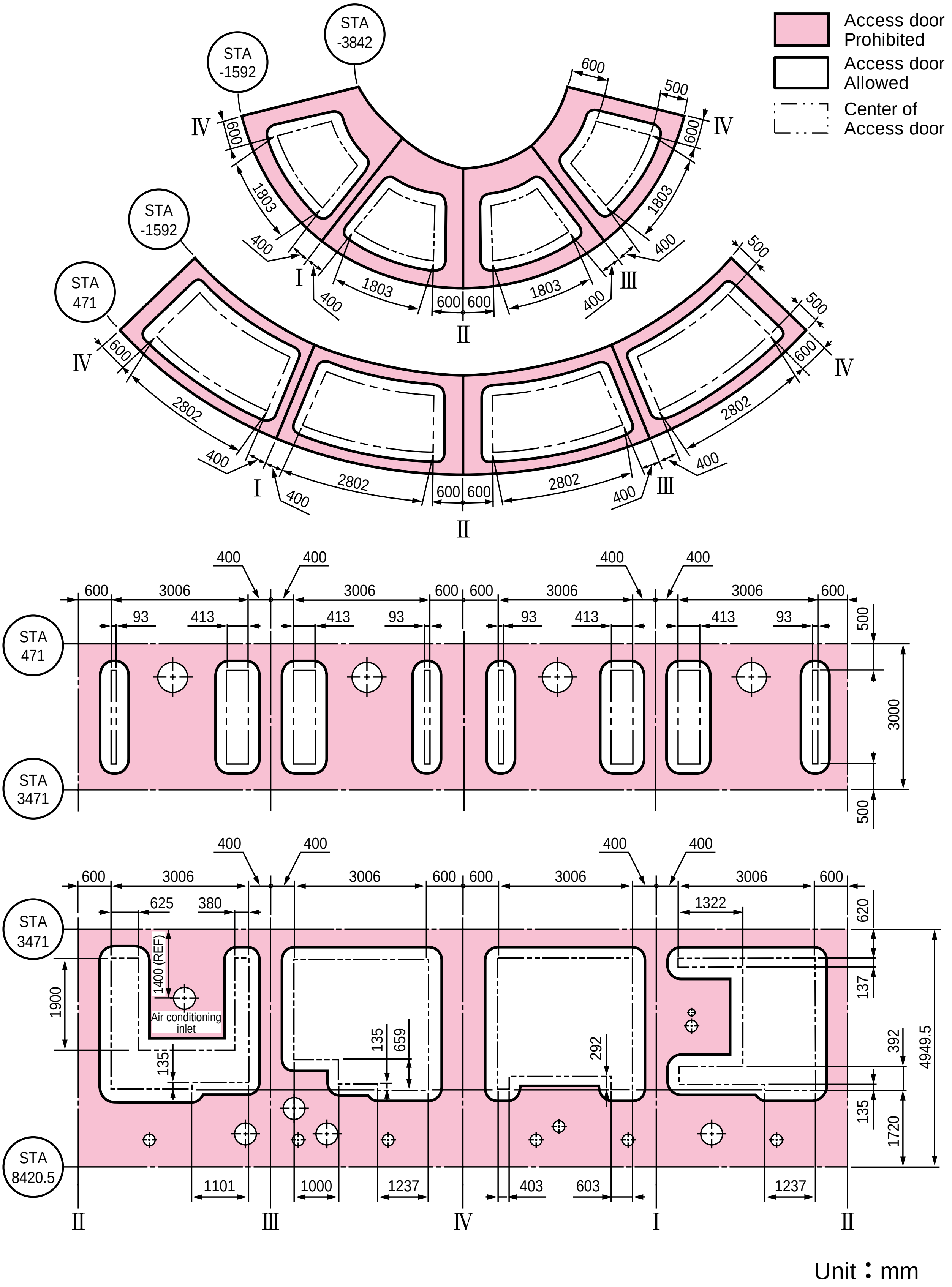


Figure 4.4.25 Allowable areas of ø 450 access door on model 5/4D lower fairing

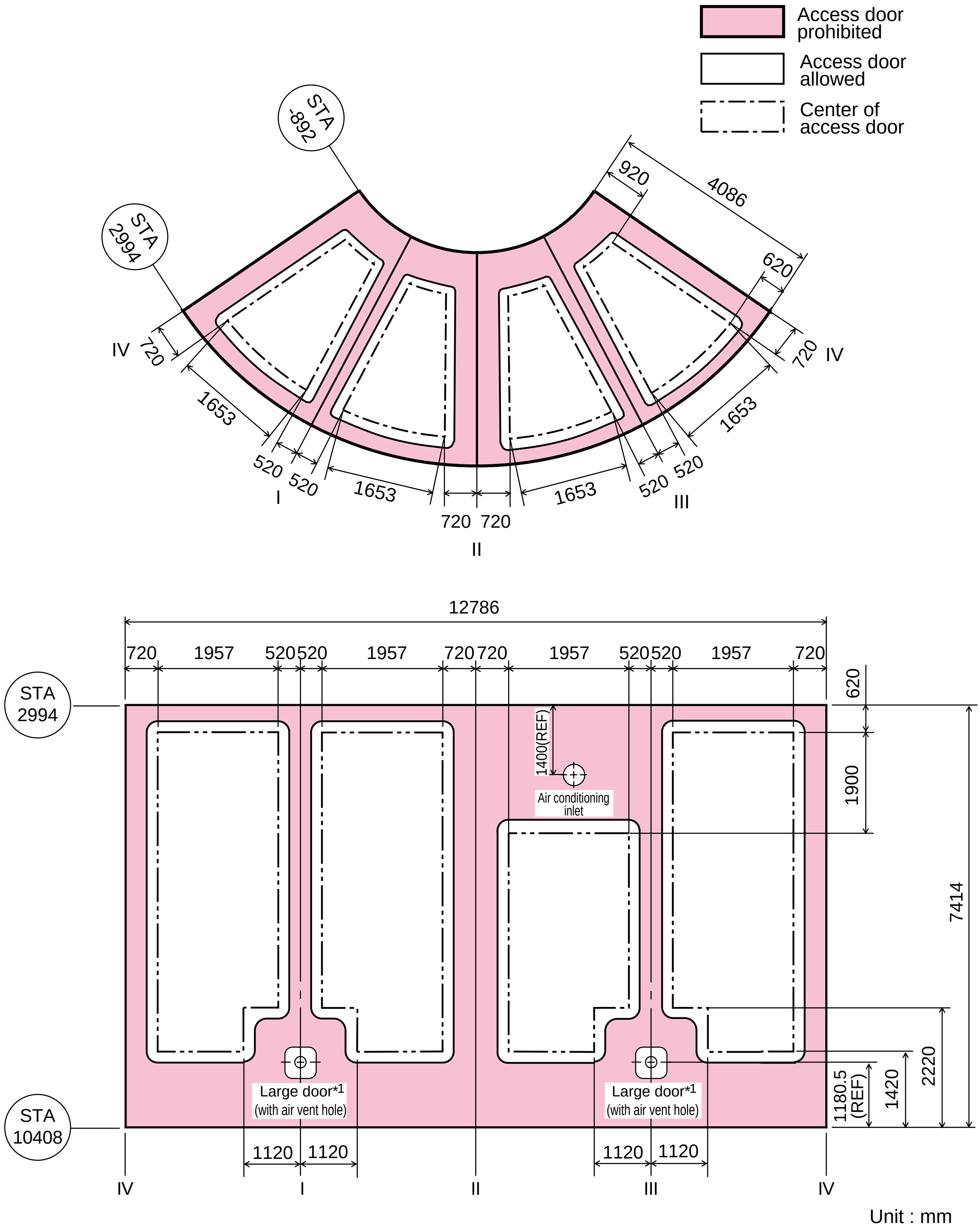


<Note>

*1 : Refer to the figure 4.4.14 for the payload fairing angle of conical section.

Figure 4.4.26 Allowable areas of ø 450 access door on model 5S-H fairing

C



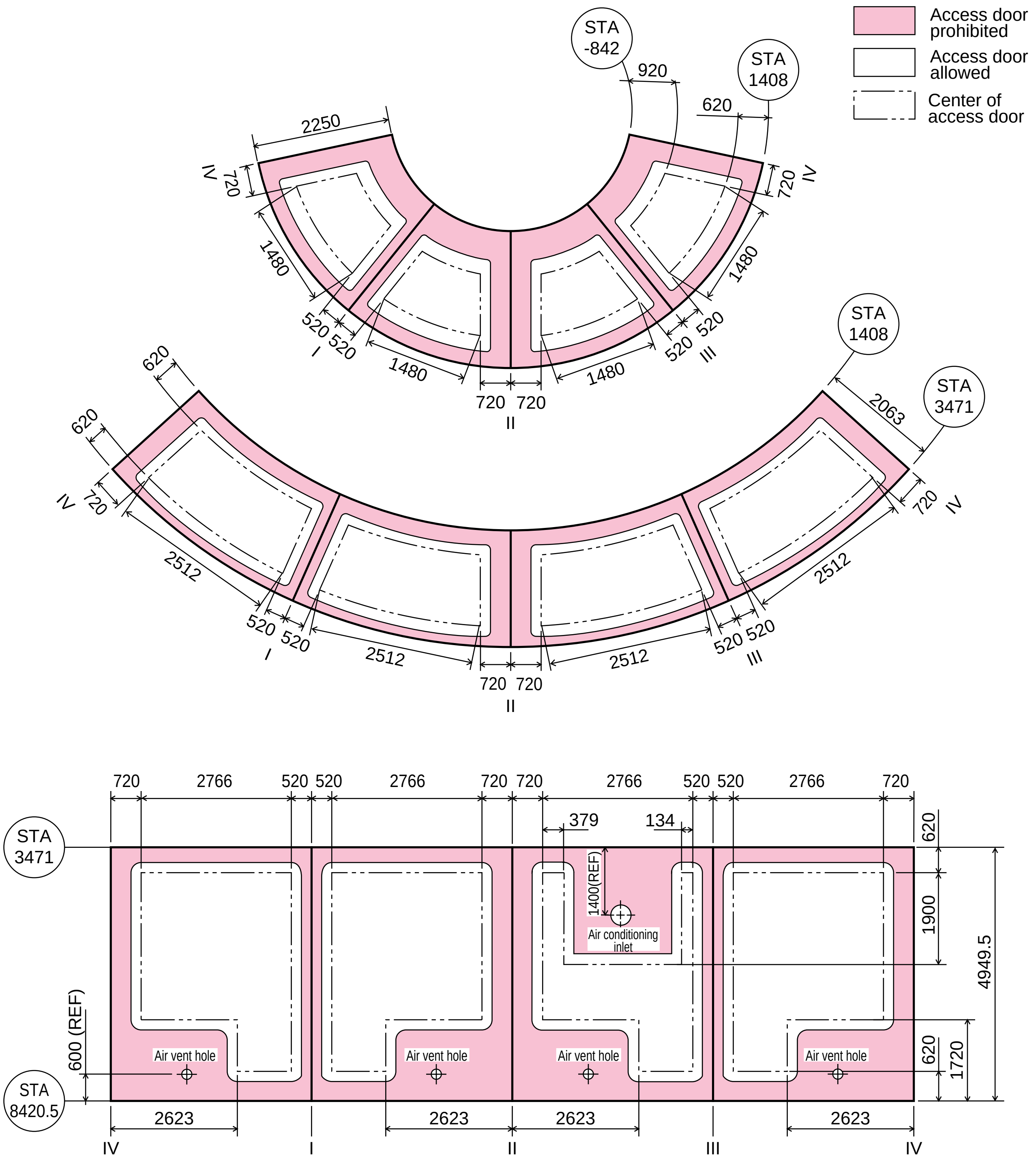
<Note>

*1 : This door is fixed and for launch vehicle operations.

*2 : Refer to the figure 4.4.2 for the payload fairing angle of conical section.

Figure 4.4.27 Allowable areas of ø 600 access door on model 4S fairing

C



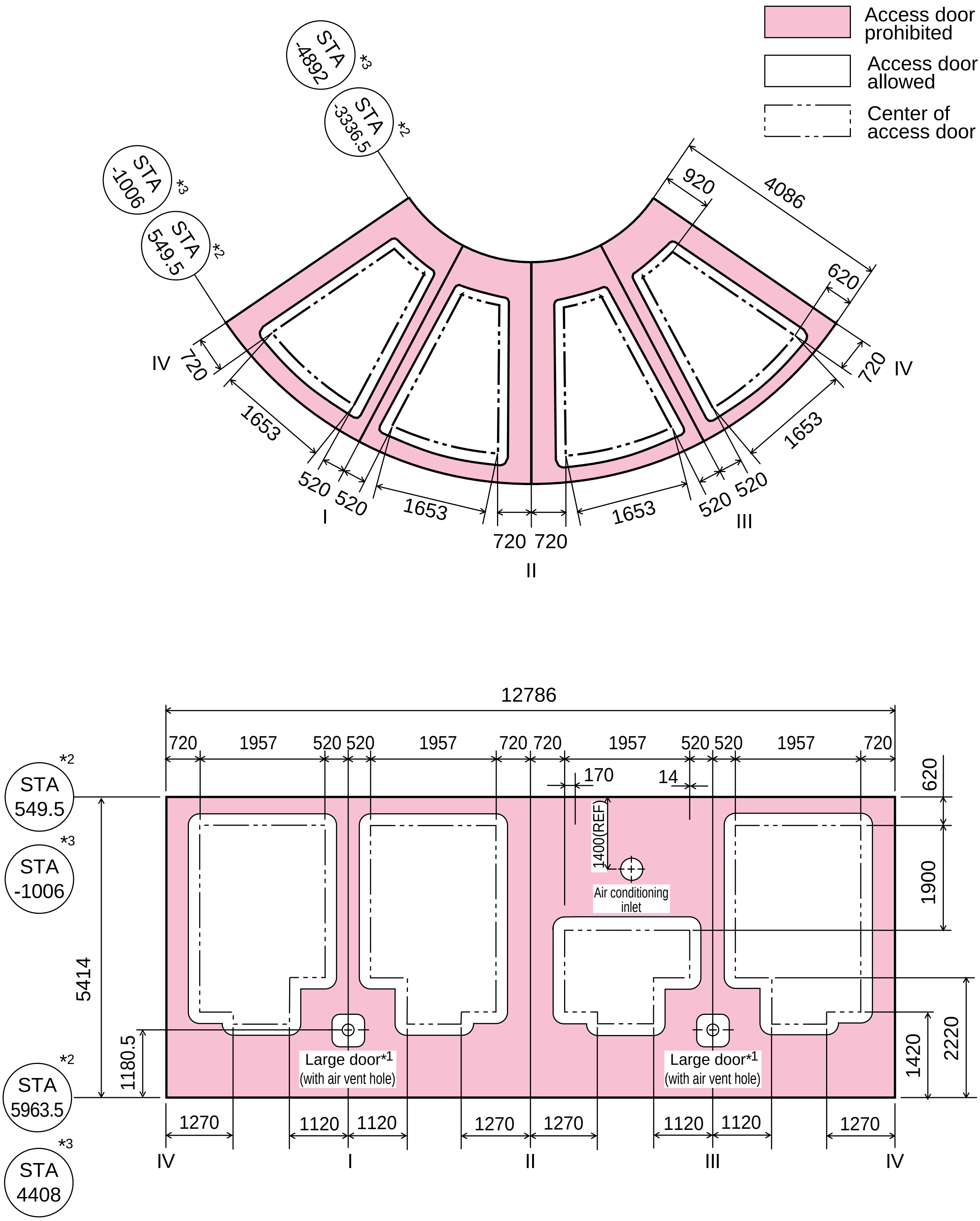
Unit : mm

<Note>

*1 : Refer to the figure 4.4.4 for the payload fairing angle of conical section.

Figure 4.4.28 Allowable areas of ø 600 access door on model 5S fairing

C

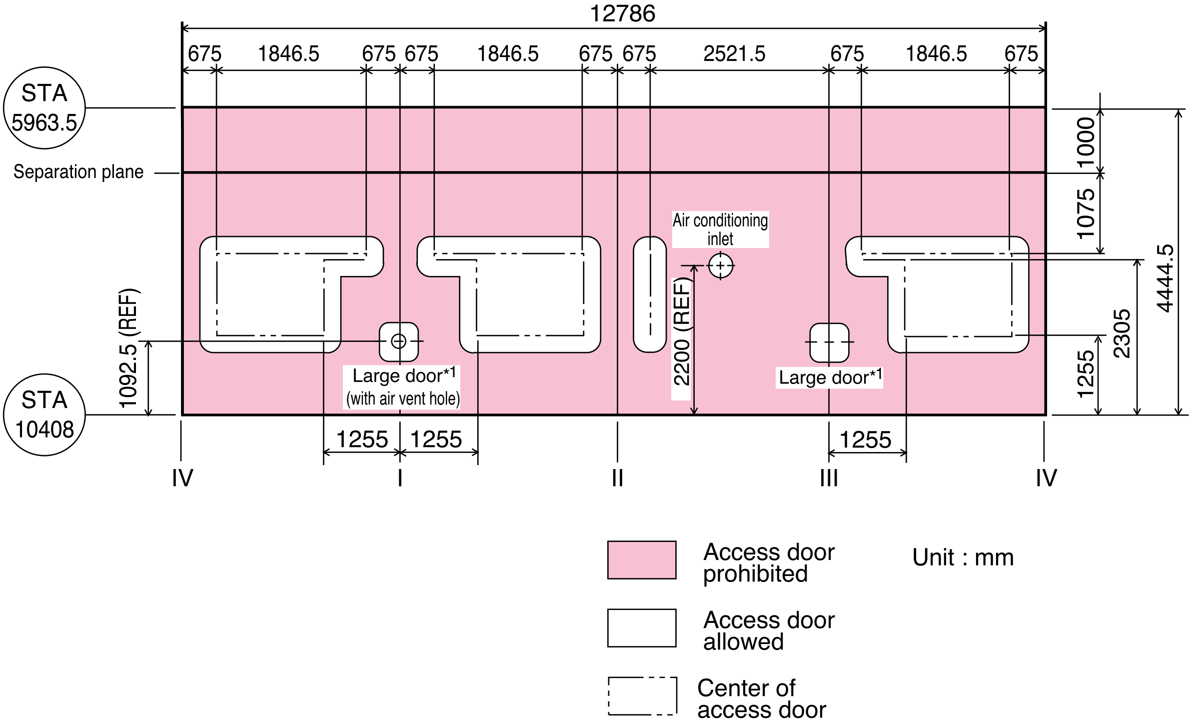


Unit : mm

<Note>

*1 : This door is fixed and for launch vehicle operations.
*2 : This station for the 4/4D-LS
*3 : This station for the 4/4D-LC
*4 : Refer to the figure 4.4.6 for the payload fairing angle of conical section.

Figure 4.4.29 Allowable areas of ø 600 access door on model 4/4D long upper fairing

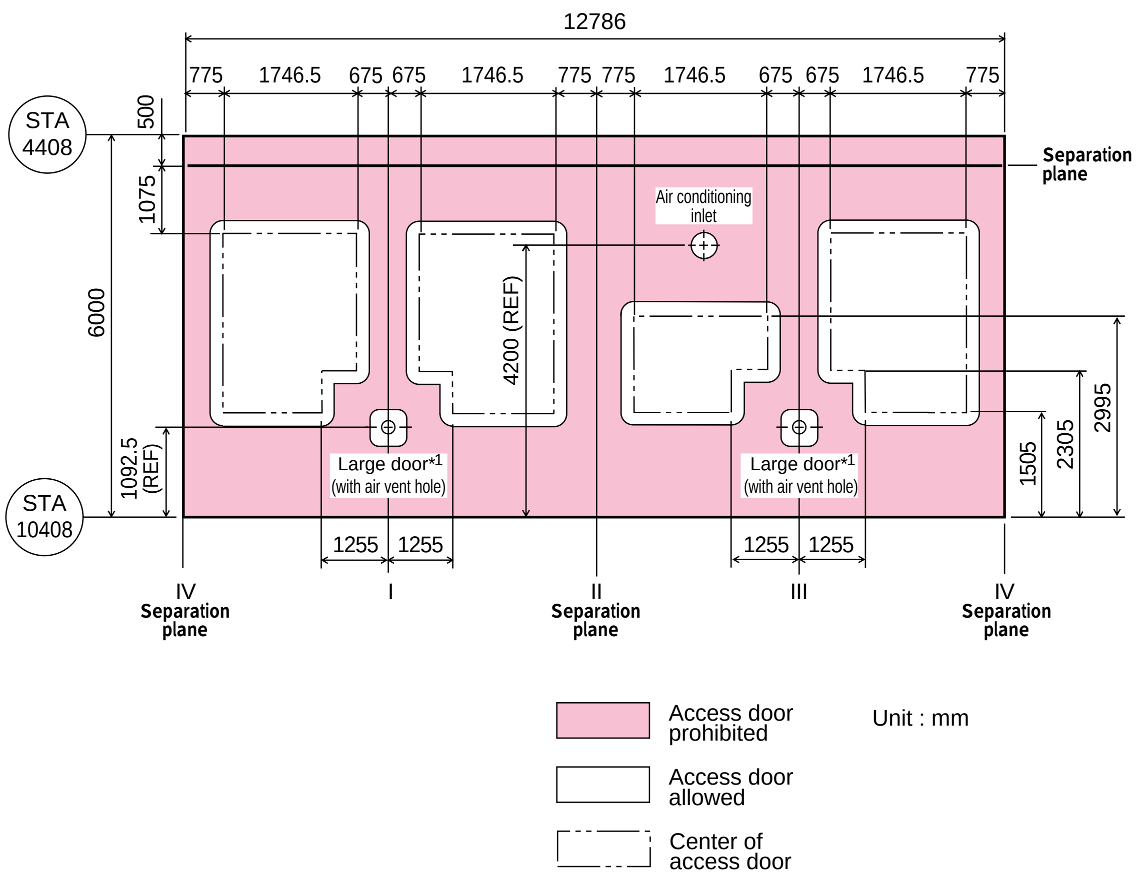


<Note>

*1 : This door is fixed and for launch vehicle operations.

Figure 4.4.30 Allowable areas of ø 600 access door on model 4/4D short lower fairing

C



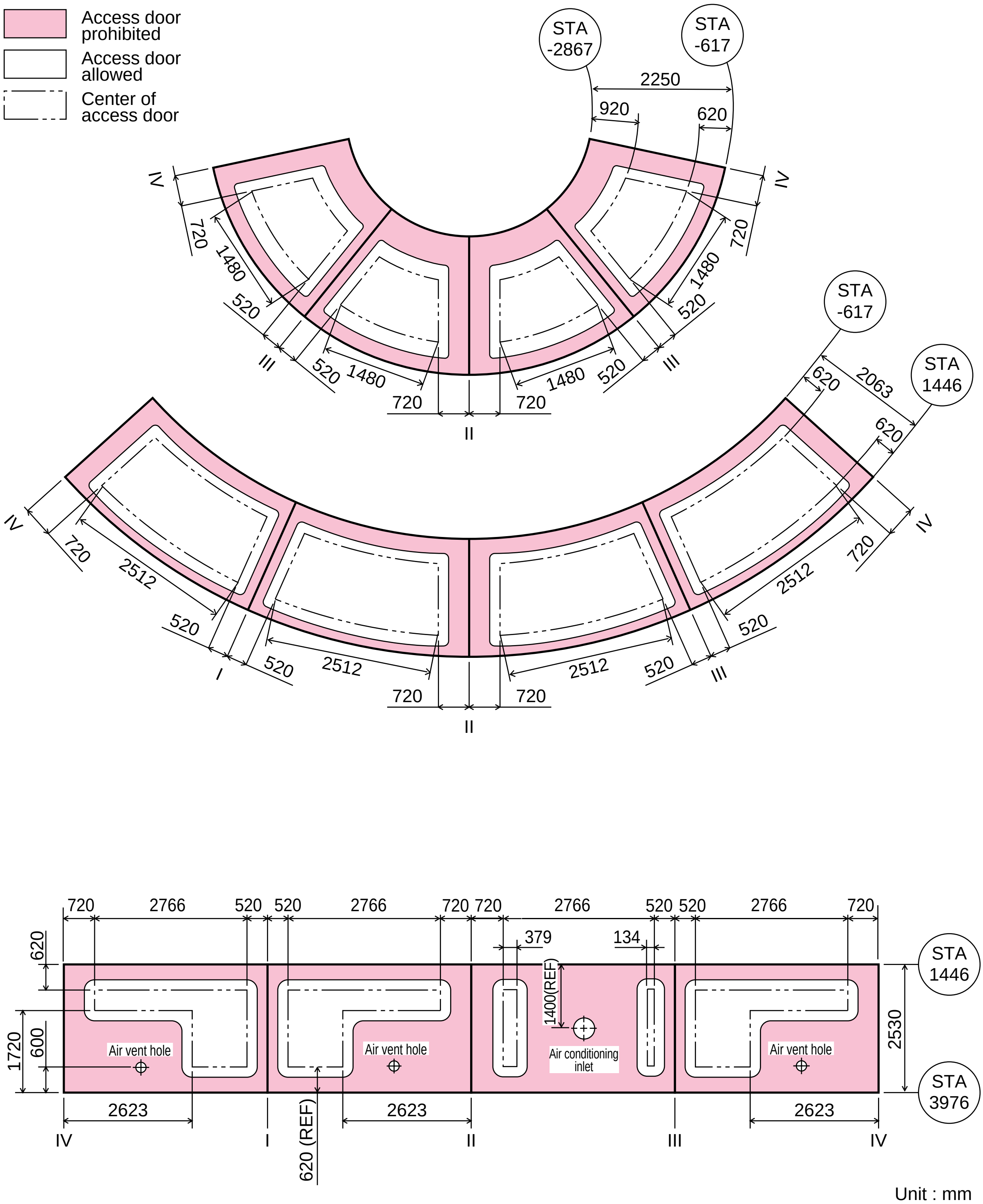
<Note>

***1 : This door is fixed and for launch vehicle operations.**

Figure 4.4.31 Allowable areas of ø 600 access door on model 4/4D clamshell lower fairing

C

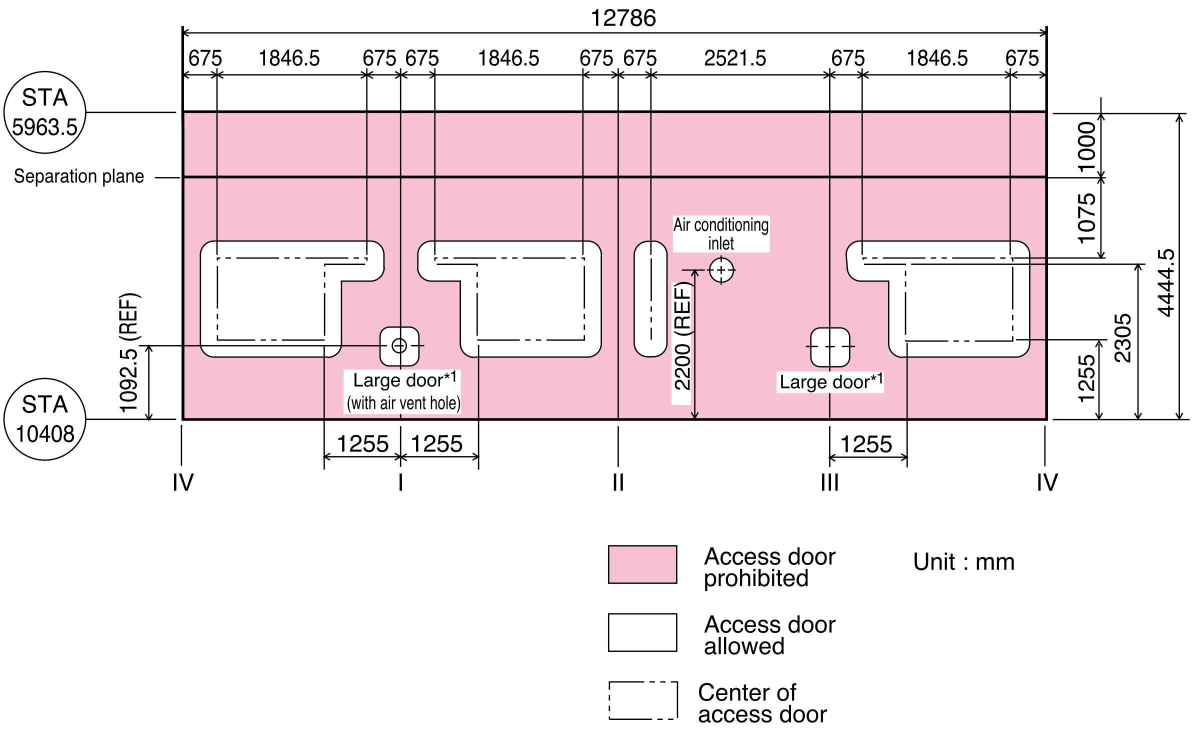
C



<Note>

*1 : Refer to the figure 4.4.11 for the payload fairing angle of conical section.

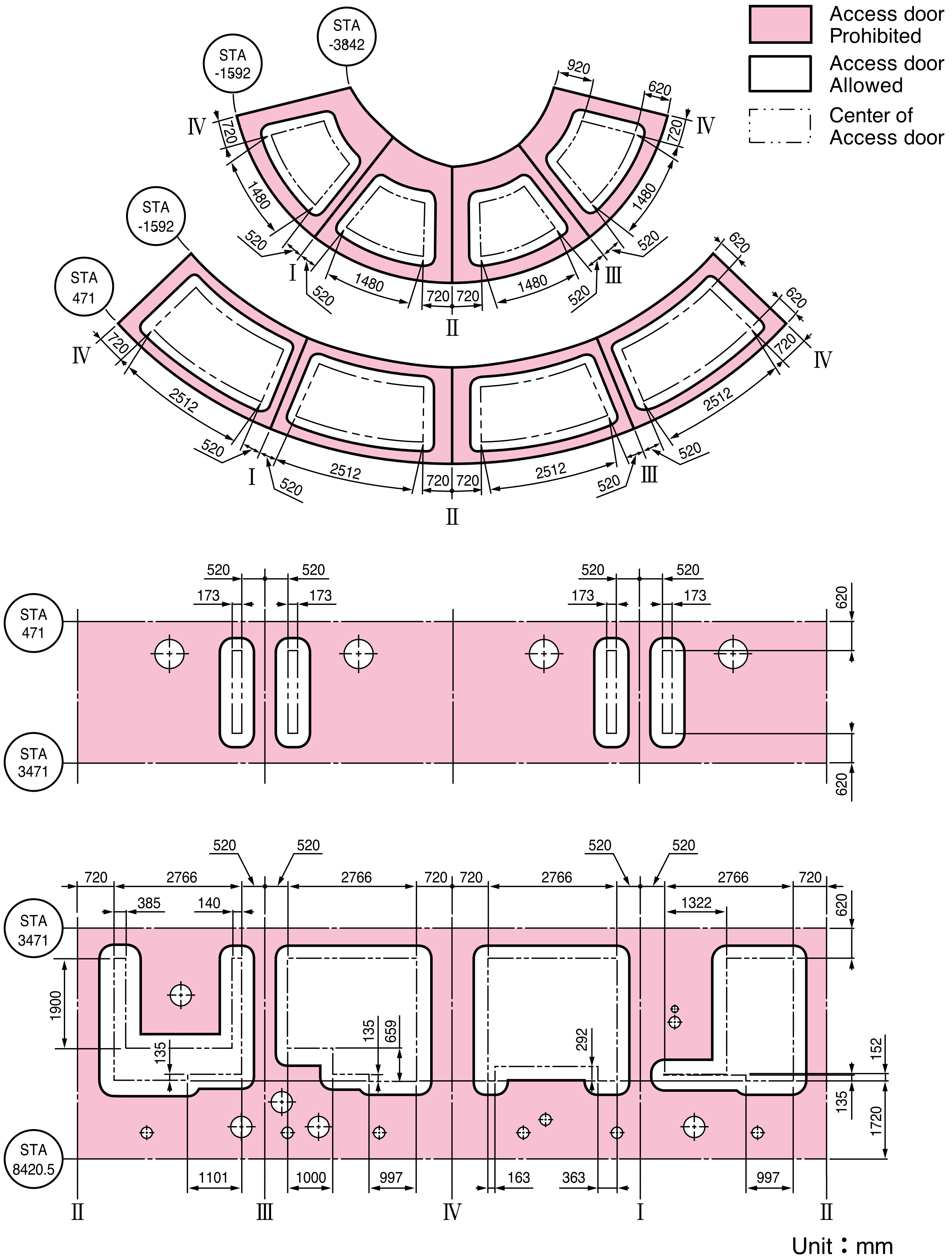
Figure 4.4.32 Allowable areas of ø 600 access door on model 5/4D upper fairing



<Note>

*1 : This door is fixed and for launch vehicle operations.

Figure 4.4.33 Allowable areas of ø 600 access door on model 5/4D lower fairing

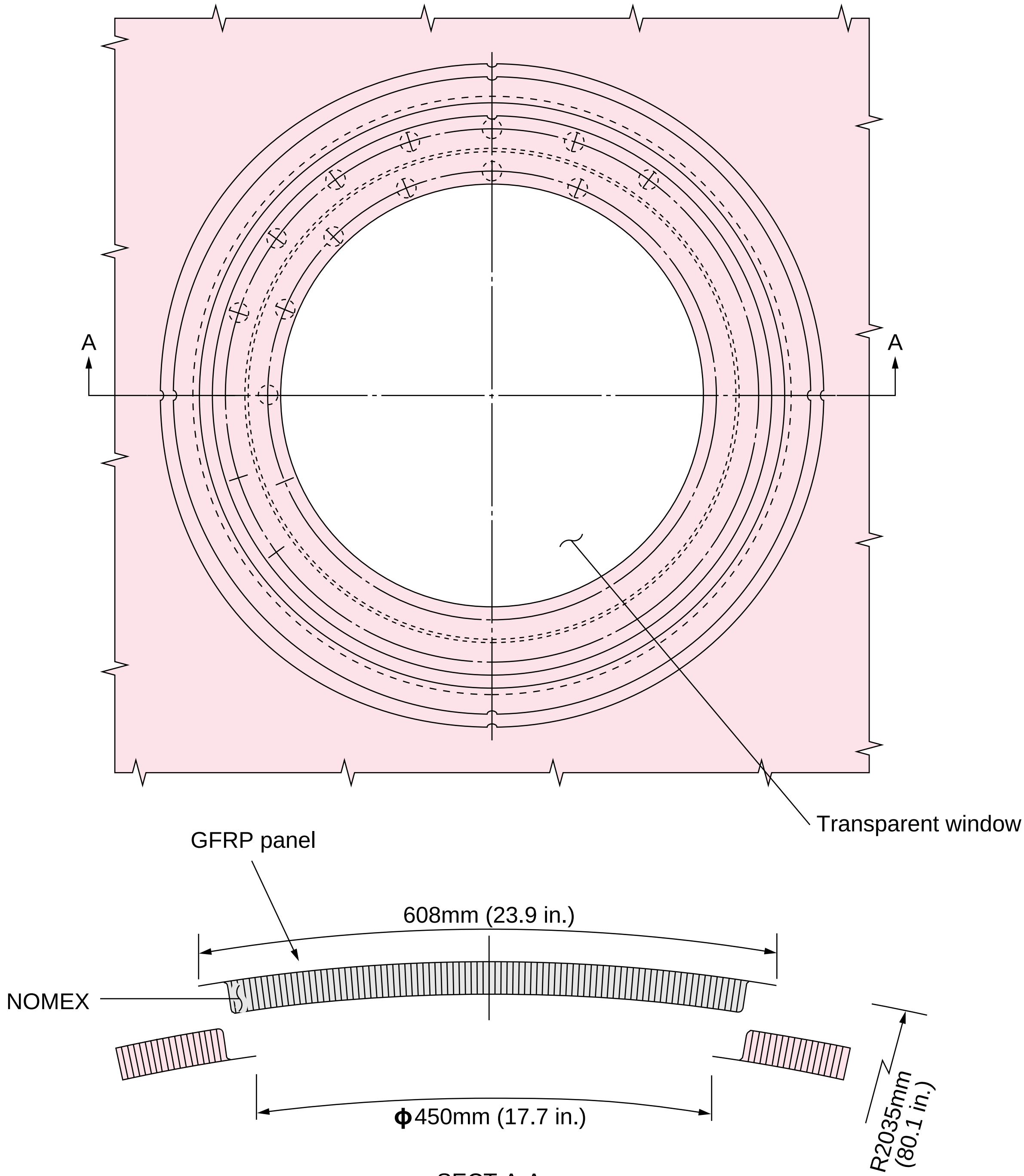


C

<Note>

*1 : Refer to the figure 4.4.14 for the payload fairing angle of conical section.

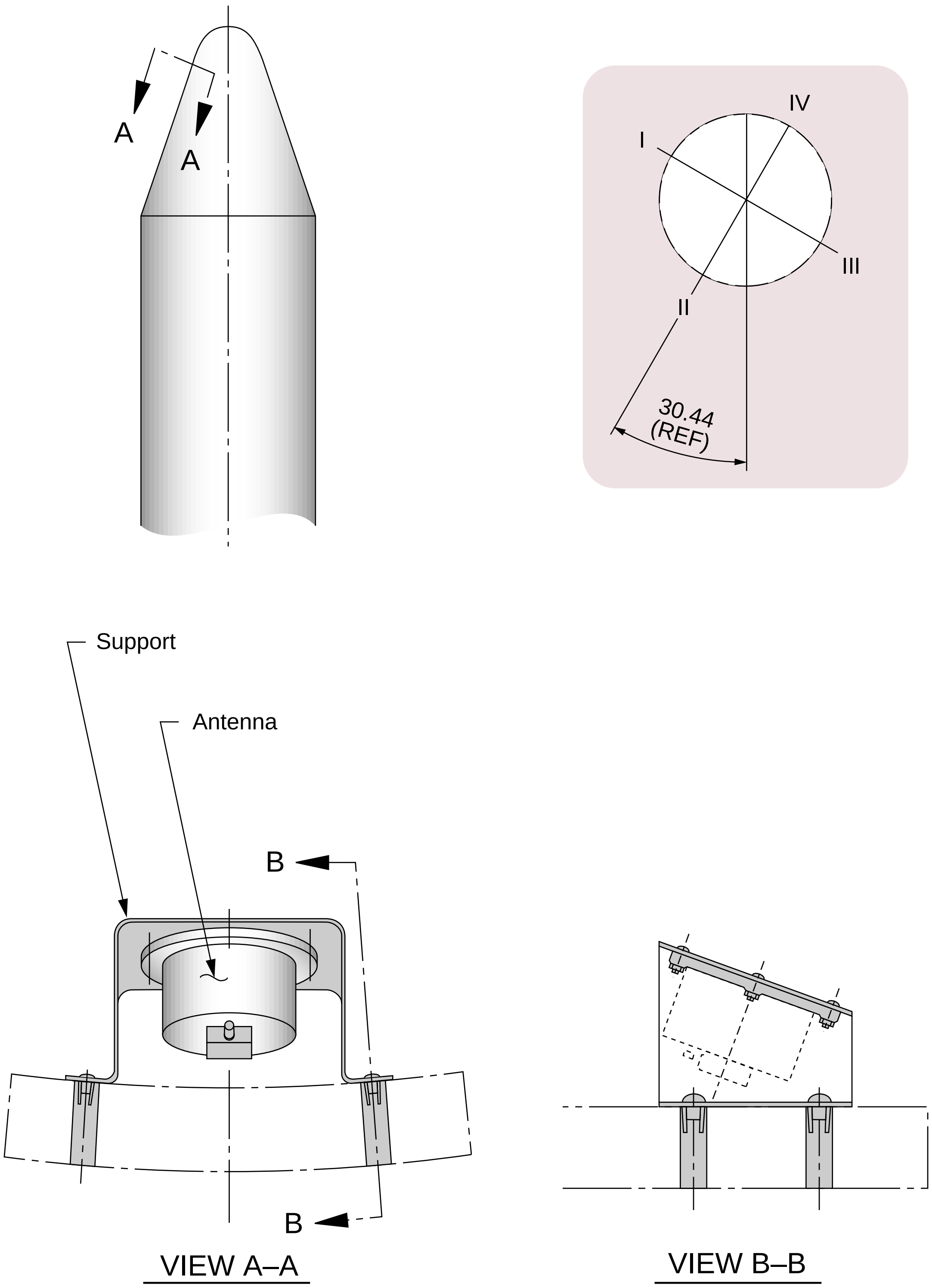
Figure 4.4.34 Allowable areas of ø 600 access door on model 5S-H fairing



Note : **This figure shows the configuration of a transparent window installed in the cylinder section of the 4S fairing.**
The size of the window should be equivalent when it is installed in the cone section.

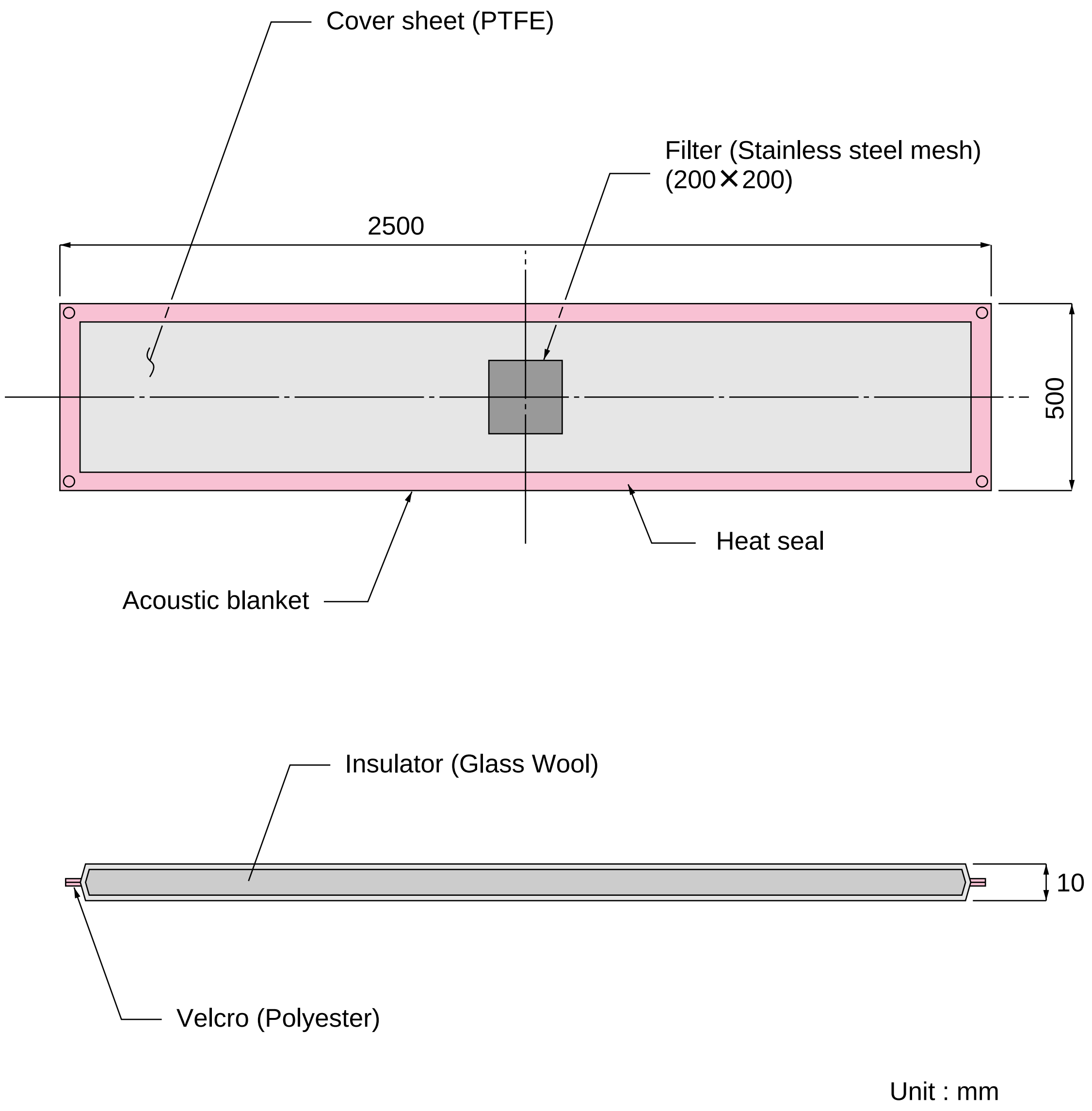
Figure 4.4.35 Transparent window

| C



Unit : mm

Figure 4.4.36 Typical Installation of internal antenna | C



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

Figure 4.4.37 Typical Configuration of acoustic blanket

4.5 Payload Adapter

4.5.1 Configuration of payload adapter

The interface parameters of the payload adapter are defined below.

a) Figure and dimensions of separation plane
b) Detailed dimensions of mating point of adapter and spacecraft
c) Installation dimensions of separation switches

4.5.1.1 Separation mechanism

The H-IIA launch vehicle provides clamp bands and separation nuts as the separation mechanism on the adapter.

(1) Clamp band mechanism

The V flange of the rear frame of the spacecraft is connected to that on 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.

The clamp band tension is designed to ensure no gaps exist between the spacecraft and the adapter frames on the ground and in flight environment. C

(2) Separation nut mechanism

The spacecraft is connected to the top of the adapter at 4 or 8 points by separation nuts. To separate the spacecraft, the separation nuts are released.

4.5.1.2 Ejection mechanism

Separation springs can be installed in the adapter to separate the spacecraft from the launch vehicle. If the spacecraft requires the launch vehicle to back away using retro thrusters, separation springs are not installed.

When the separation springs are installed, static loads shall act on each mating structure as a limiting load, in addition to the acceleration load described in § 3.2.1.

4.5.1.3 Spacecraft separation monitoring switches

The standard launch vehicle has two microswiches on the top of the adapter to monitor spacecraft separation. It is possible to locate the actuator pad of the spacecraft microswitch on the top of the adapter.

4.5.1.4 Nomenclature of payload adapter

(1) "1194" means the interface diameter at the top of the adapter is 1194 mm.

(2) "M" means the adapter has a clamp band separation mechanism.

(3) "S" means the adapter has a separation nut separation mechanism.

4.5.2 Adapter types

Table 4.5.1 shows characteristics of the payload adapters for the H-IIA launch vehicle. C

Detailed information related to the payload adapter is presented in APPENDIX 3. C

4.5.3 Mission modifications

4.5.3.1 Separation springs

Separation springs can be installed inside an adapter to force the spacecraft away from the launch vehicle. These springs are not applied to adapters 2360S and 3470S.

a) Number of separation springs : 4 to 8

4.5.3.2 Umbilical connectors

Umbilical connectors can be installed on an adapter to meet the user's requirements.

a) Number of umbilical connectors : 2
b) Location of umbilical connectors : Decided by discussion with user
c) Specification of umbilical connectors : Described in § 4.6.6.4

Table 4.5.1 Characteristics of payload adapters

Items / Model	Spacecraft Interface diameter (mm)	Height (mm)	Mass (kg)	Connecting device	Connecting points	Applications	
937M	945	900	90	clamp band	–	None	A
937MH	958	900	80	clamp band	–	None	B
937M-Spin	959	605	130	clamp band	–	NASDA : GMS-5	A
937M-Spin-A	958	1000	150	clamp band	–	None	B
1194M	1215	480	100	clamp band	–	NASA : TRMM NASDA : ETS-VII MOT : MTSAT ESA : ARTEMIS	B
1666M	1666	1000	100 (T.B.D.)	clamp band	–	None	A
1666S	1664 (PCD)	1000	100 (T.B.D.)	separation nuts	4	None	B
2360S	2360 (PCD)	300	40	separation nuts	8	NASDA : ETS-VI, COMETS	B
3470S	3472 (PCD)	1089	350	separation nuts	4	NASDA : ADEOS	B
1666MA	1666	480	100	clamp band	–	None	C
239M	239	100	5	clamp band	–	None	

Launch vehicle interface diameter : 2190 PCD
Spin : With spin table

4.6 Electrical and RF Interface

4.6.1 General

This section describes electrical interfaces such as electrical grounding between the launch vehicle and the spacecraft; service interfaces from the launch vehicle; specifications of interface connector (umbilical connector); and RF interface issues such as transparency of payload fairing, constraints of spacecraft RF transmission, etc.

4.6.2 Electrical grounding

The launch vehicle and spacecraft must maintain the same electrical potential in flight, for which grounding is necessary. The grounding reference point should be located on the separation plane of the spacecraft and launch vehicle, where the grounding should be provided by mechanical contact of both sides at connection.

MIL-B-5087B class S (less than 1Ω) resistance requirement is applied to the H-IIA launch vehicle grounding. Therefore, surface finish of the spacecraft separation structure should satisfy the above requirement. The spacecraft provider should contact NASDA if MIL-B-5087B class R (less than 2.5 mΩ) is required.

4.6.3 Umbilical interface

4.6.3.1 Umbilical lines for single launch mission

(1) The umbilical interface block diagram between the spacecraft and the ground facilities for a single launch is shown in Figure 4.6.1.

(2) Two coaxial cable lines can be provided for the spacecraft RF link.

(3) The specifications for interface connectors with the spacecraft (the spacecraft umbilical connector) are defined in the Interface Control Specifications (ICS) for each spacecraft. The maximum number of connector pins is 120.

(4) The interface to the ground facilities is conducted using the umbilical connector located on the payload fairing.

4.6.3.2 Umbilical lines for dual launch mission

(1) The umbilical interface block diagram between the spacecraft and the

ground facilities for a dual launch is shown in Figure 4.6.2.

(2) The specifications of interface connectors for the spacecraft in the upper fairing and lower fairing are the same as for a single launch mission, § 4.6.3.1 (3). | A

| A

4.6.4 Command and power interface

Table 4.6.1 shows the electrical interfaces which the H-IIA launch vehicle provides to the spacecraft.

4.6.4.1 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 interface meetings. | A

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 one additional command as an option. However, for dual launch, the launch vehicle hardware spare channels are limited, so special arrangements must be made at the interface meetings. The launch vehicle system can provide ten command signals in total including other pyrotechnic commands, electrical commands and dry loop commands for a dual launch.

Main electrical characteristics of the pyrotechnic command are the following and the wiring diagram for the pyrotechnic command is shown in Figure 4.6.3.

a) Battery	Engine battery (26 AH)	C
b) Voltage	28^{+6}_{-4} VDC	A
c) Ignition timing	To be determined at the interface meetings	
d) Pulse width of igniting signal	To be determined at the interface meetings	
e) Minimum igniting current	To be determined at the interface meetings	
f) Recommended igniting current	To be determined at the interface meetings	
g) Number of power cartridges	To be determined at the interface meetings	
h) Non-igniting current	To be determined at the interface meetings	
i) Bridge wire resistance	To be determined at the interface meetings	
j) Insulation resistance	To be determined at the interface meetings	
k) Insulation resistance after ignition	To be determined at the interface meetings	

4.6.4.2 Electrical command (discrete signal)

(1) The spacecraft organization may request electrical commands (discrete signals) from the launch vehicle organization, if necessary. Total number of command signals is as described in § 4.6.4.1. If in a dual launch, both spacecraft require these signals, special arrangements must be made at the interface meetings. These commands are an optional service to the spacecraft. C

(2) The launch vehicle wiring diagram for the electrical commands is shown in Figure 4.6.4. B

(3) Main electrical characteristics of the electrical command

a) Voltage	28^{+6}_{-4} VDC	A
b) Load resistance	To be determined at the interface meetings	
c) Supply current	To be determined at the interface meetings	B
d) Insulation resistance	To be determined at the interface meetings	
e) Supply timing	To be determined at the interface meetings	
f) Supply time (duration)	To be determined at the interface meetings	
g) Spacecraft circuit condition	Insulated from ground and structure	

Design details will be established at the interface meetings.

4.6.4.3 Dry loop command

(1) The spacecraft organization may request dry loop commands from the launch vehicle organization, if necessary. Total number of command signals is as described in § 4.6.4.1. If in a dual launch, both spacecraft require these signals, special adjustment is required at the interface meetings. These commands are an optional service to the spacecraft.

(2) The launch vehicle wiring diagram

The wiring diagram of the electrical commands is shown in Figure 4.6.5.

(3) Main electrical characteristics of the electrical command | A

a) Supply timing	To be determined at the interface meetings
b) Supply time (duration)	To be determined at the interface meetings
c) Supply current	To be determined at the interface meetings
d) Circuit resistance in launch vehicle	To be determined at the interface meetings
e) Insulation resistance	To be determined at the interface meetings

Design details will be arranged at the interface meetings.

4.6.4.4 Power supply | A

If the spacecraft organization requires power supply, the launch vehicle can provide it to the extent which depends on the mission. Detailed specifications of power will be arranged at the interface meeting and so on. | C

4.6.5 In-flight telemetry

4.6.5.1 Separation status transmission

The second stage telemetry system transmits spacecraft separation status signals for monitoring from the ground.

Spacecraft separation can be ascertained from the signals initiated by microswitches installed on the top of the adapter at the separation plane.

This is a standard service.

In case of the separation after spin up by the launch vehicle (in longitudinal), it may be impracticable to monitor the spacecraft separation in real time.

4.6.5.2 Dynamic environments data transmission

The dynamic environments data measured at the adapter are transmitted by the second stage telemetry system.

The in-flight dynamic environments data for each spacecraft measured as standard service are listed below. The frequency is less than 100 (T.B.D.) Hz.

a) Spacecraft separation status 2 ch. | B

b) Temperature of adapter structure 1 ch.

c) Acceleration at adapter structure 3 ch. B

4.6.6 Interface connectors between spacecraft and launch vehicle

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

4.6.6.1 Interface connectors procurement responsibility A

Interface connector receptacles (the spacecraft umbilical connectors) which is installed on the spacecraft, and plugs shall be provided by the H-IIA launch vehicle organization.

The spacecraft organization can procure both plugs and receptacles and perform the fit checks, and the plugs may be furnished to the H-IIA launch vehicle. A

4.6.6.2 Interface connectors for single launch A

Interface connectors for transferring electrical signals for a single launch shall be installed on the payload fairing. In this case, electrical interfaces are maintained until fairing jettison. For this reason, lanyard style push-pull connectors are used for the H-IIA vehicle plugs. C

If interface connectors are installed on the adapter, electrical interfaces will be maintained until spacecraft separation.

Figure 4.6.6 shows interface connectors for a single launch.

4.6.6.3 Interface connectors for dual launch A

Interface connectors for transferring electrical signals for a dual launch shall normally be the same as in a single launch case for the upper fairing, and on the adapter for the lower fairing. C

If interface connectors are installed on the adapter, electrical interfaces will be maintained until spacecraft separation.

Figure 4.6.7 shows interface connectors for a dual launch.

4.6.6.4 Standard interface connector specifications

The standard interface connector specifications shall conform to the requirements of NASDA-ESPC-915 or ESA/SCC SPEC. No. 3401/008, contacts SPEC. No. 3401/009. B

Connectors with the specifications shown in Table 4.6.2 shall generally be used. These connectors are manufactured by CIE DEUTSCH.

Table 4.6.2 shows the standard interface connector specifications.

4.6.6.5 Other interface connector characteristics

(1) Plug disconnecting characteristics at fairing jettison

a) Disconnecting angle and force of the plugs:

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

The disconnecting force of the plugs shall be as shown below.

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

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

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

e) Key position : Key position of interface connectors shall be set so that connector keys are oriented in the forward direction of the H-IIA launch vehicle.

B

Shell size	Min. disconnecting force (N)	Max. disconnecting force (N)
3	5.3	88.3
7	6.6	88.3
12	8.8	150.0
19	13.2	167.7
27	17.7	176.5
37	26.5	194.2
61	30.9	196.6

(2) When interface connectors are located on the adapter

The key position of interface connectors shall be set so that connector keys are oriented in the external radial direction of the H-IIA launch vehicle. Interface connectors are located as each adapter type defines; 2 connectors should be located 180° opposite each other. The receptacle surface should face the separation plane.

4.6.7 RF constraints

4.6.7.1 Fairing transparency for spacecraft RF communications

The fairing RF transparency will be specified after the location of the spacecraft antenna is fixed.

4.6.7.2 Operating constraints

The spacecraft shall not radiate narrow-band electrical fields exceeding the "acceptable H-IIA radiation susceptibility level" as the worst case of the sum spurious level shown in Figure 4.6.8. The radiation emission level is defined at the spacecraft separation plane (at the lower spacecraft separation plane in a dual launch). A C

4.6.8 Electrical and RF requirements for launch phase

4.6.8.1 Electrical requirements

The spacecraft organization shall satisfy the following constraints in the final preparation phase leading up to lift-off.

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

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

4.6.8.2 RF requirements

Launch vehicle on-board equipment has frequencies as follows: C

VHF telemeter transmitters	: 295 to 297 MHz (standard), 294 to 296 MHz (option)
UHF telemeter transmitters	: 2289 to 2291 MHz (standard), 2200 to 2290 MHz (option: changeable according to RF frequency of spacecraft)
SHF telemeter transmitters	: 14.855 to 14.865 GHz (option)
Radar transponder	: 5.23 to 5.786 GHz (transmission & reception)
Command destruct receiver (CDR)	: 400 to 500 MHz
Global positioning system receiver (GPSR)	: 1.425 to 1.675 GHz (option)

For "Hot launch" spacecraft, the spacecraft organization shall satisfy the following constraints.

a) Spurious radiation interference levels from the launch vehicle and TNSC will not exceed those given in Figure 4.6.8. Spacecraft's acceptable spurious radiation levels shall satisfy this constraints. C

b) The spacecraft telemetry frequency band above must not overlap the launch vehicle bands.

c) The spacecraft shall not radiate a narrow-band electrical field at the spacecraft separation plane exceeding the limit set (as the worst case of the sum spurious level) in Figure 4.6.8. C

4.6.9 RF Link Interface

4.6.9.1 General

RF link for S-band telemetry/command between the spacecraft on the H-IIA launch vehicle and the user GSE can be provided. RF link between the spacecraft on-board and Masuda Tracking and Communication Station (MTCS) can be also provided (option). A

The interface conditions are defined as follows:

(1) Link path

a) From the spacecraft on-board to STA2

b) From the spacecraft on-board to ML GSE room

c) From the spacecraft on-board to MTCS

(2) Operation phase

a) When ML is in VAB after the spacecraft VOS

b) During the transfer of ML from VAB to PAD

c) Before the launching after ML transfer to PAD

4.6.9.2 RF link with ML/STA2

RF link for S-band telemetry/command between the spacecraft and STA2 can be provided after the spacecraft VOS. The following routes can be prepared to make RF coupling with the spacecraft. A

(a) Fairing internal antenna and RF coaxial umbilical cable

(b) Air link with ML umbilical mast horn antenna via fairing RF transparent window

The route above can be switched by remote-control from STA2 or ML GSE room. Available numbers and locations of fairing RF transparent windows need the discussion. A

The spacecraft is connected to ML GSE room with RF through a coaxial cable on the umbilical mast and bi-directional amplifier. The users can communicate with the spacecraft by hooking up their RF GSE to the IDF in ML GSE room. When users need to communicate with the spacecraft from STA2 checkout room, fiber optics can be used between STA2 and ML (See 4.6.9.2(1),(2)).

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

Before ML is transferred from VAB or after being fixed on the LP, RF link between ML and the checkout room in STA2 is provided through fiber optics and RF/Optical Signal Converter (modulator/demodulator). The similar RF/Optical Signal Converter is installed in the checkout room in STA2. The user GSE can be connected with RF/Optical Signal Converter in STA2. RF/Optical Signal Converter can be remote-controlled from STA2. The RF link schematic in VAB is shown in Figure 4.6.9. The RF link schematic on LP is shown in Figure 4.6.11. For dual launch, RF link can be provided for each user. The communication with the spacecraft can be performed by installing RF GSE in ML GSE room instead of using a route to STA2, however, personnel are not allowed to stay at ML GSE room after X-5Hr (the time all personnel should leave).

(2) RF link during ML transfer

During transfer of ML from VAB to LP, fiber optic network is not available. RF link during transfer is established by using air propagation between ML and VAB. ML during transfer is connected with VAB via RF Air link. RF / Optical Signal Converter is installed in VAB. STA2 and VAB are connected via fiber optics (see Figure 4.6.10). The route from STA2 should be switched on VAB side after ML transfer begins. When the communication with the spacecraft is performed by installing RF GSE in ML GSE room without using a route to STA2, personnel can stay at ML GSE room.

4.6.9.3 RF Link with MTCS

A

RF link in S-band between the spacecraft on-board and Masuda Tracking and Communication Station (MTCS) can be provided. The installation of user's equipment in MTCS or the use of NASDA tracking network would be optional.

(1) When ML is in VAB

The spacecraft in the VAB cannot be connected directly from the MTCS via Air link. RF link between MTCS and the spacecraft shall be performed via STA2. As described in 4.6.9.2, RF link between the spacecraft and the checkout room in STA2 is provided. From the checkout room in STA2 to MTCS, the link can be provided through a STA2 inhouse coaxial cable and STA2 outside antenna for MTCS.

(2) During ML transfer

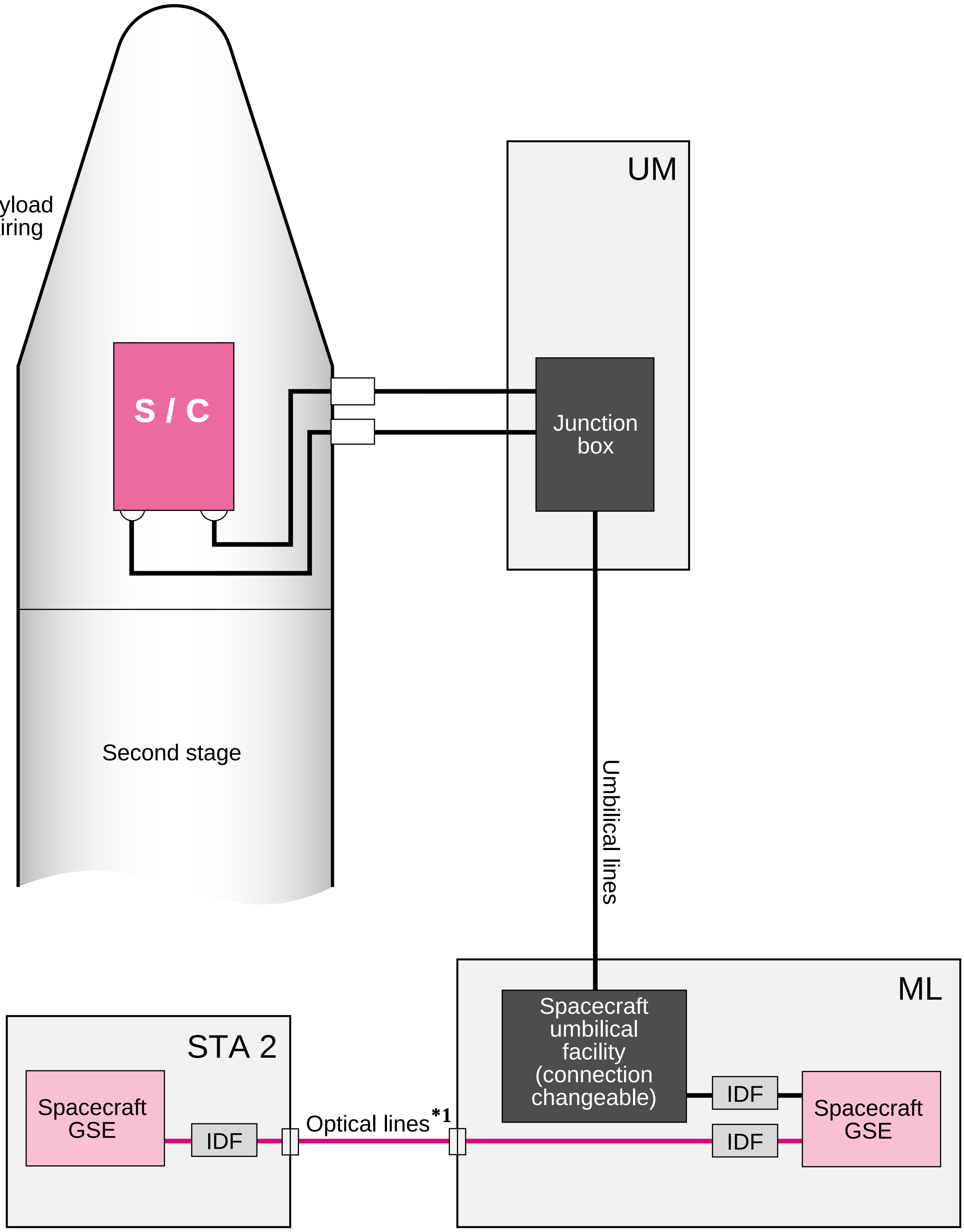
The spacecraft can be linked directly from MTCS via Air link. In this case, RF transparent windows need to be installed. The numbers and locations of RF transparent windows need the discussion. A

(3) After ML is transferred on LP

The spacecraft can be linked directly from MTCS via Air link (as same as (2) above). Furthermore, via fiber optic network, the spacecraft on the LP can be connected with MTCS through STA2.

One faring RF transparent window is provided for each user as standard. When Air link is needed for the link between umbilical mast horn antenna and the spacecraft, the transparent window shall be located on the line connecting TT/C antenna with horn antenna. When Air link between MTCS and the spacecraft is used, the RF window shall be located on the line connecting TT/C antenna with MTCS. As umbilical mast horn antenna is seen in the completely different direction from MTCS from the view of the spacecraft, two fairing RF transparent windows (or more) may be installed. In this case, the installation of the second window or further would be optional. See § 4.4.4. for the constrains regarding the installation of RF transparent windows.

The relative clocking between the spacecraft and the fairing may affect the locations of the transparent windows. The locations of other access doors or the large doors may also be affected.



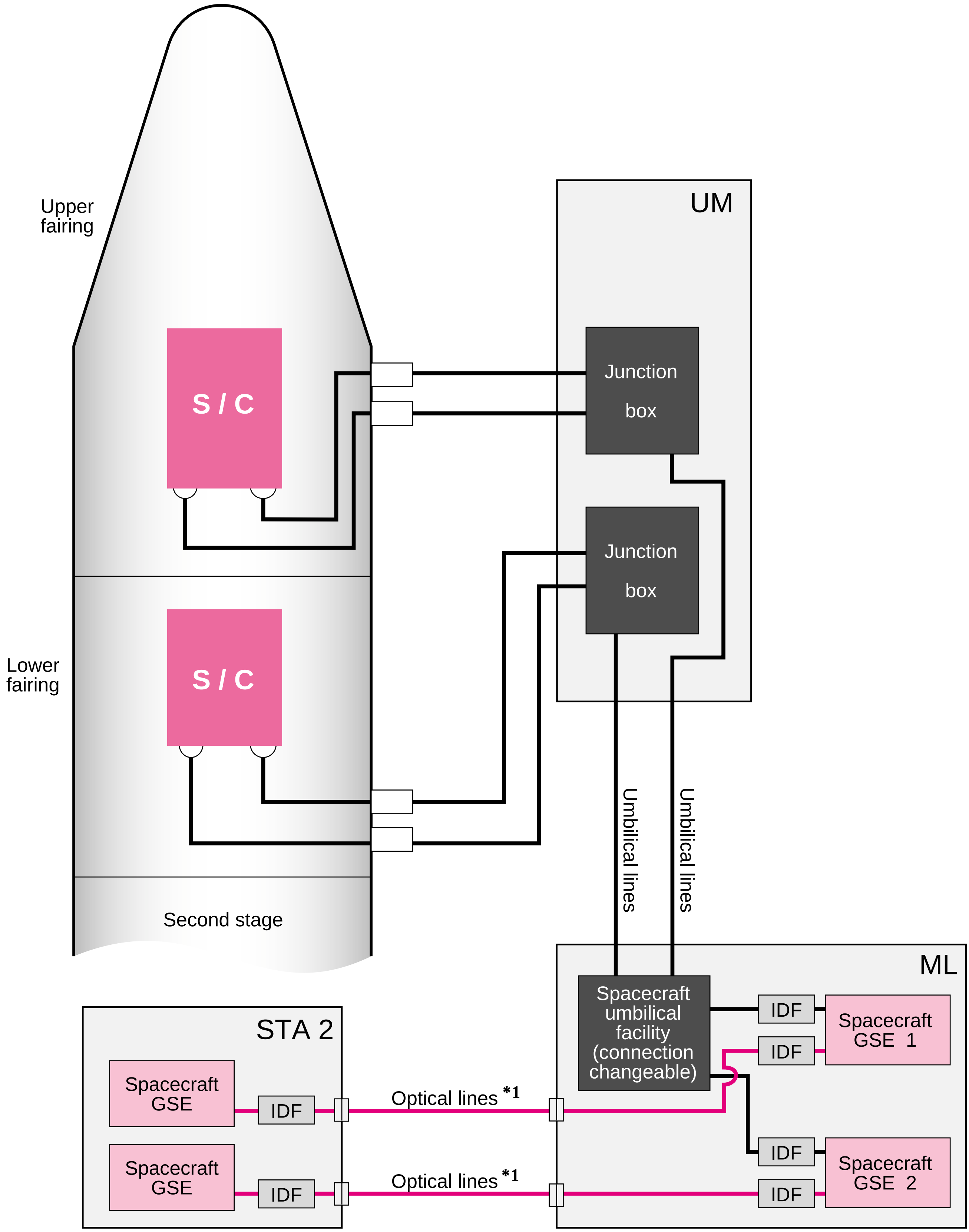
A

A

<NOTE>
IDF : Interface Distribution Facility
*1 : They have no modem between STA2 and ML.

A

Figure 4.6.1 Umbilical interfaces for single launch



<NOTE>
IDF : Interface Distribution Facility
*1 : They have no modem between STA2 and ML.

Figure 4.6.2 Umbilical interfaces for dual launch

Table 4.6.1 Electrical interfaces

Item	Interface access	Description	Related paragraph(s)
Umbilical interface	Electrical interface DBAS type connector	1) Single launch (a) Wire quantity : 120 wires 2) Dual launch *Upper spacecraft* (a) Wire quantity : 120 wires *Lower spacecraft* (a) Wire quantity : 120 wires	4.6.3.1 4.6.3.2
Command Interface	Same as umbilical interface	1) Spacecraft separation command --- standard (When user provides payload adapter) 2) Pyrotechnic command --- option 3) Electrical command --- option 4) Dry loop command --- option	4.6.4.1 4.6.4.1 4.6.4.2 4.6.4.3
Telemetry interface	--------	1) Separation status --- standard 2) In-flight environmental data --- standard	4.6.5.1 4.6.5.2

A

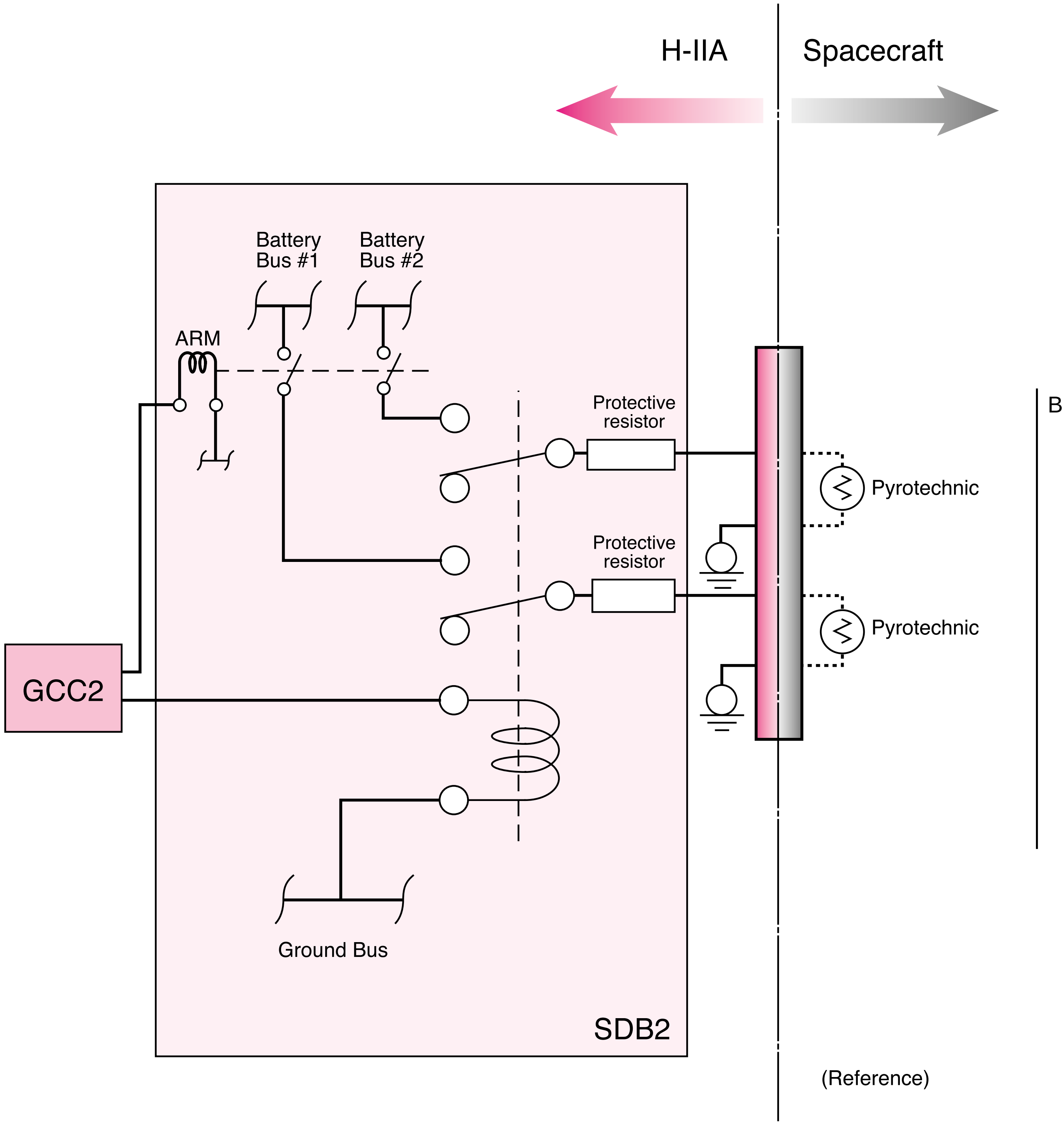


Figure 4.6.3 Pyrotechnic command wiring diagram

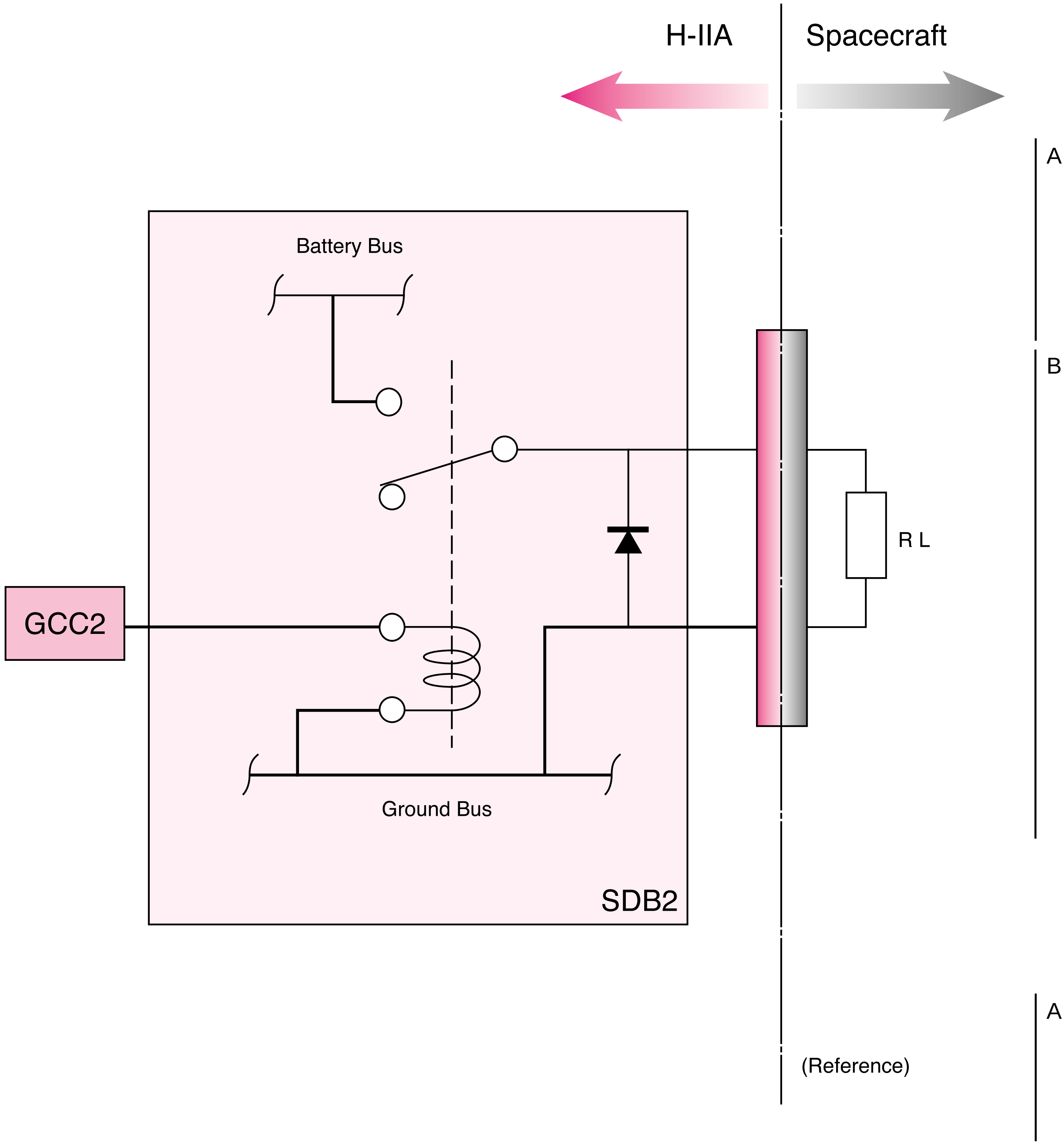


Figure 4.6.4 Electrical command wiring diagram

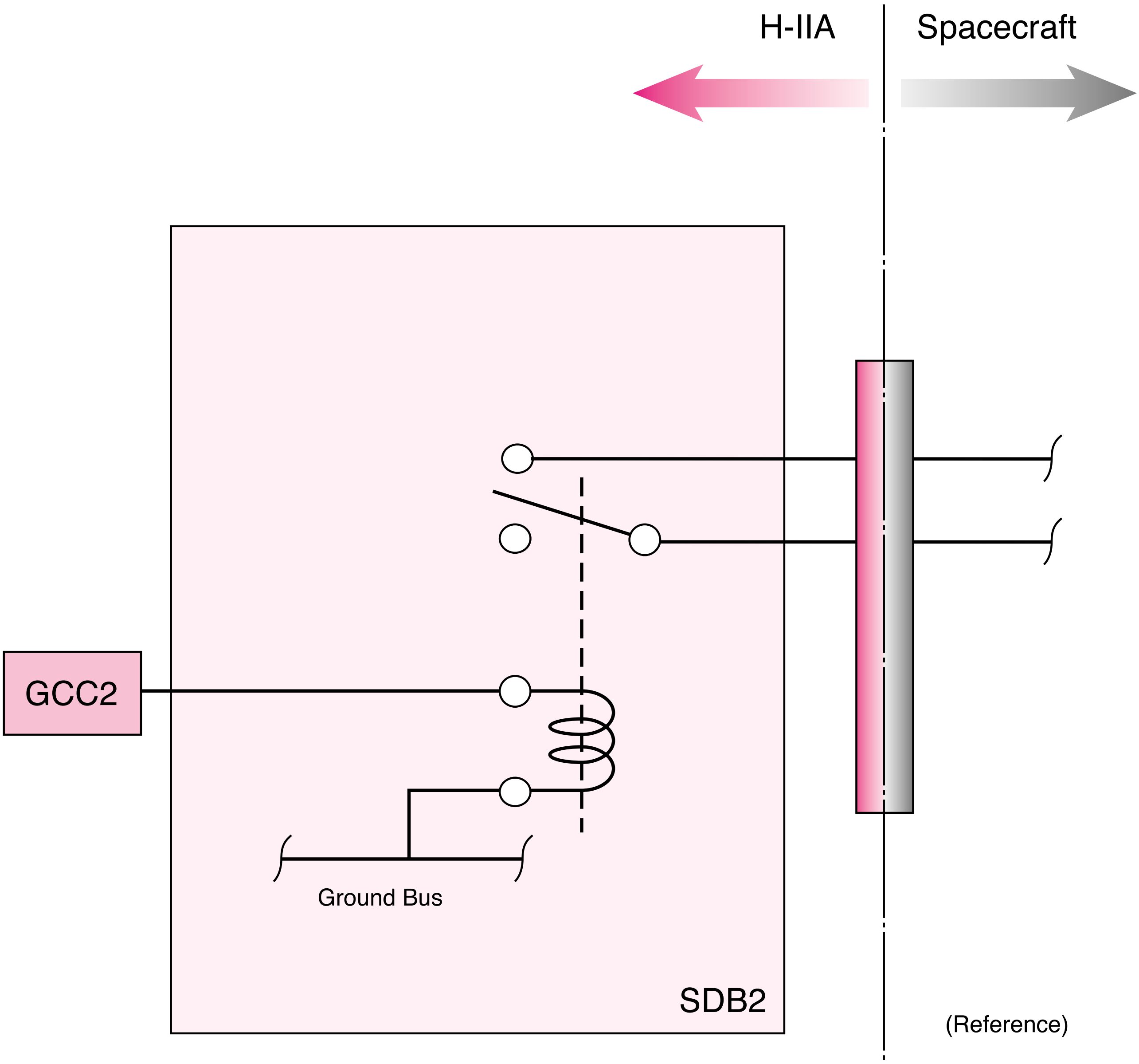


Figure 4.6.5 Dry loop command wiring diagram

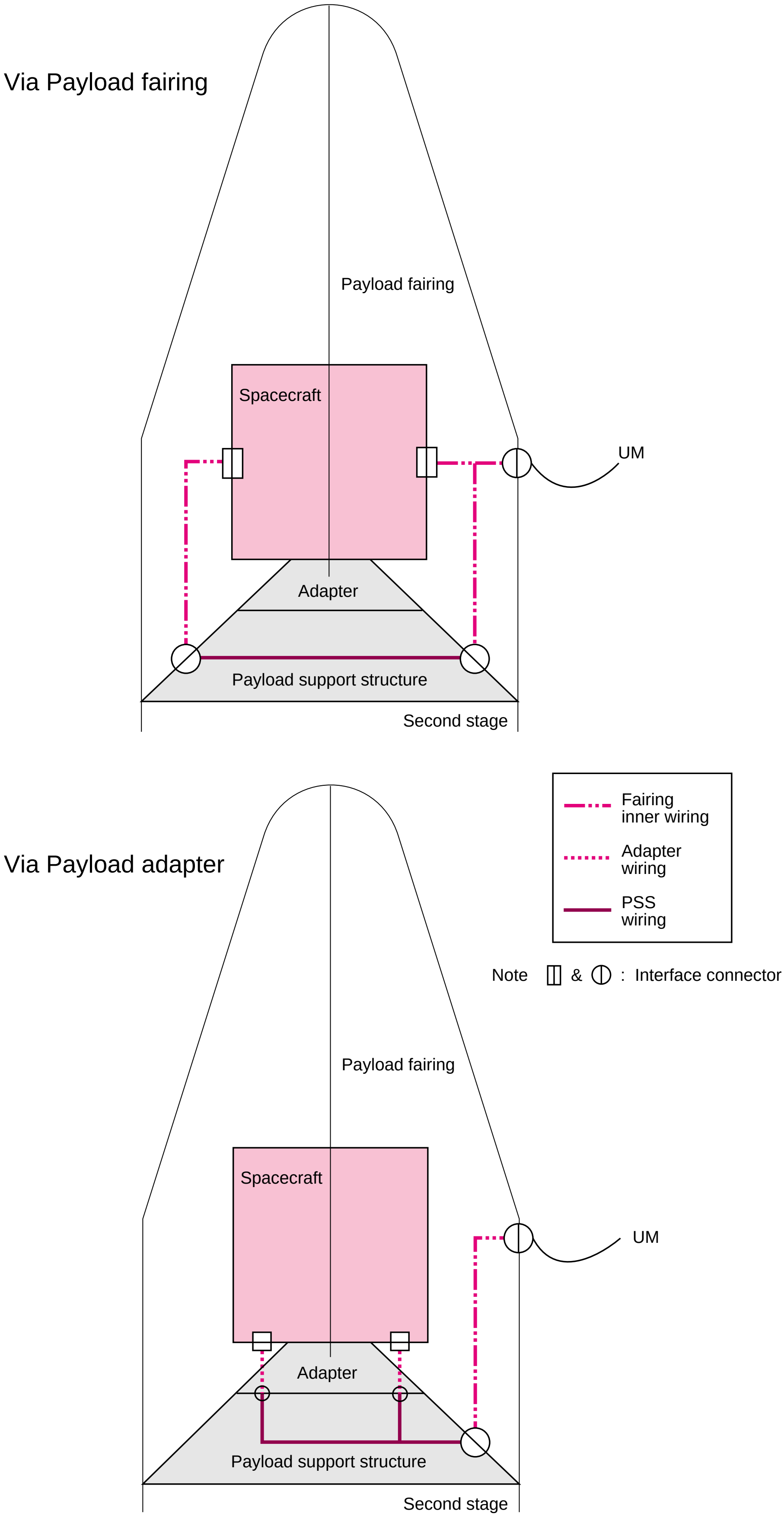
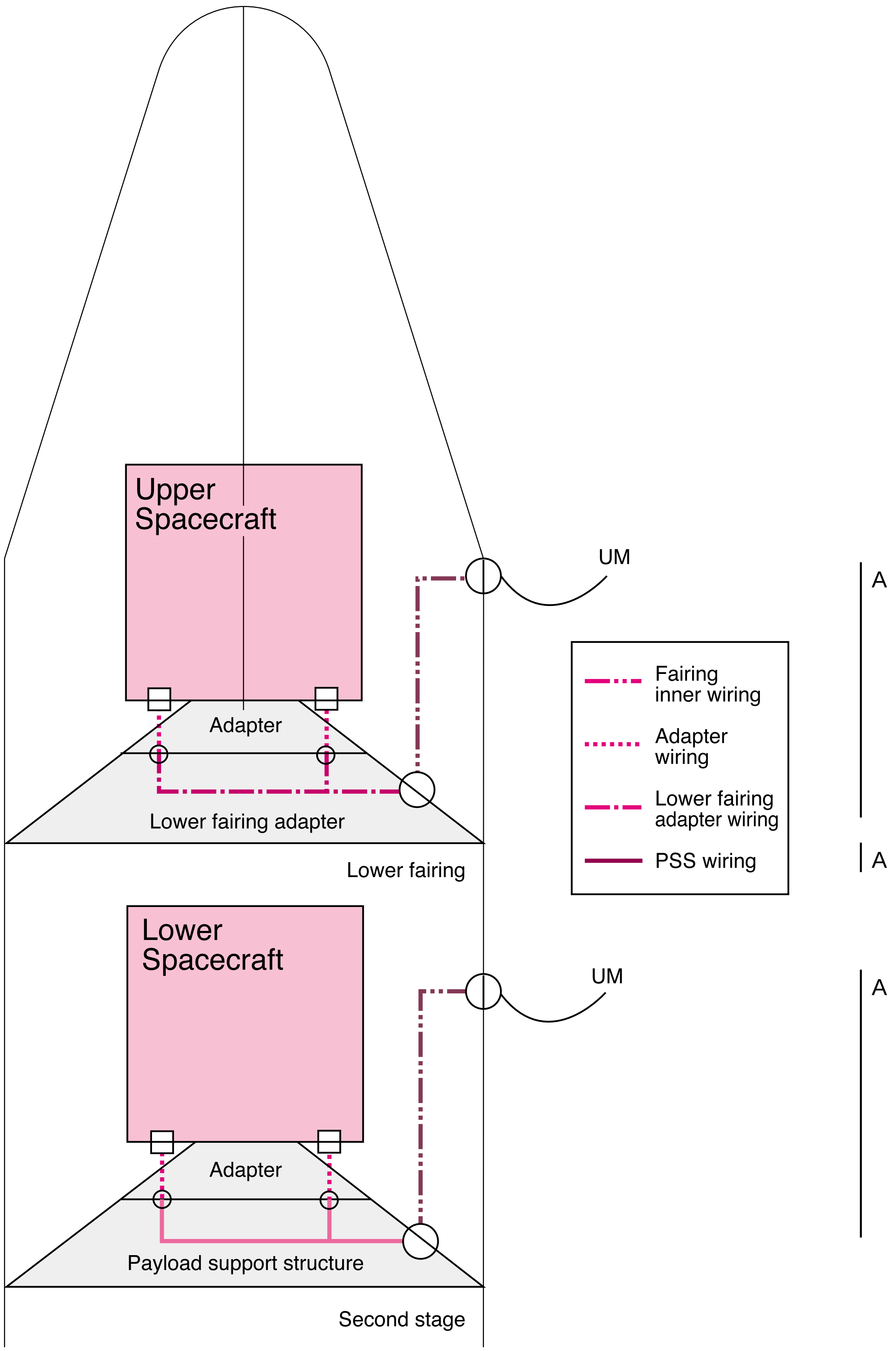


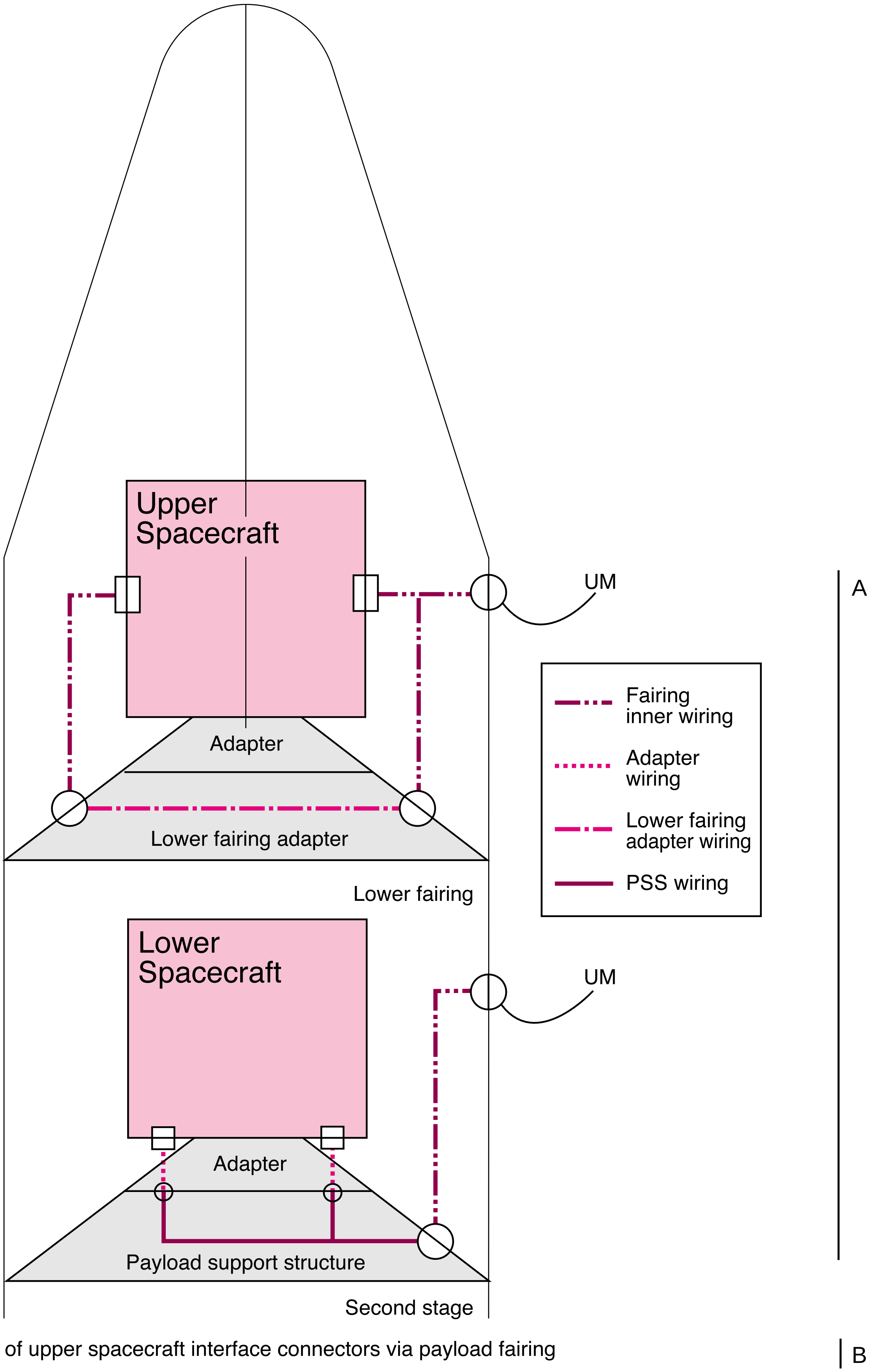
Figure 4.6.6 Interface connectors for single launch



Example of upper spacecraft interface connectors via payload adapter B

Note ▯ & ⦶ : Interface connector A

Figure 4.6.7(1/2) Interface connectors for dual launch (via payload adapter)



Example of upper spacecraft interface connectors via payload fairing

Note ◫ & ⦶ : Interface connector

Figure 4.6.7(2/2) Interface connectors for dual launch (via payload fairing)

Table 4.6.2 Standard interface connector specification

DBAS■■ ■ ■■-O ■N-■■■■

Lanyard cable length (to be applied only to plugs)

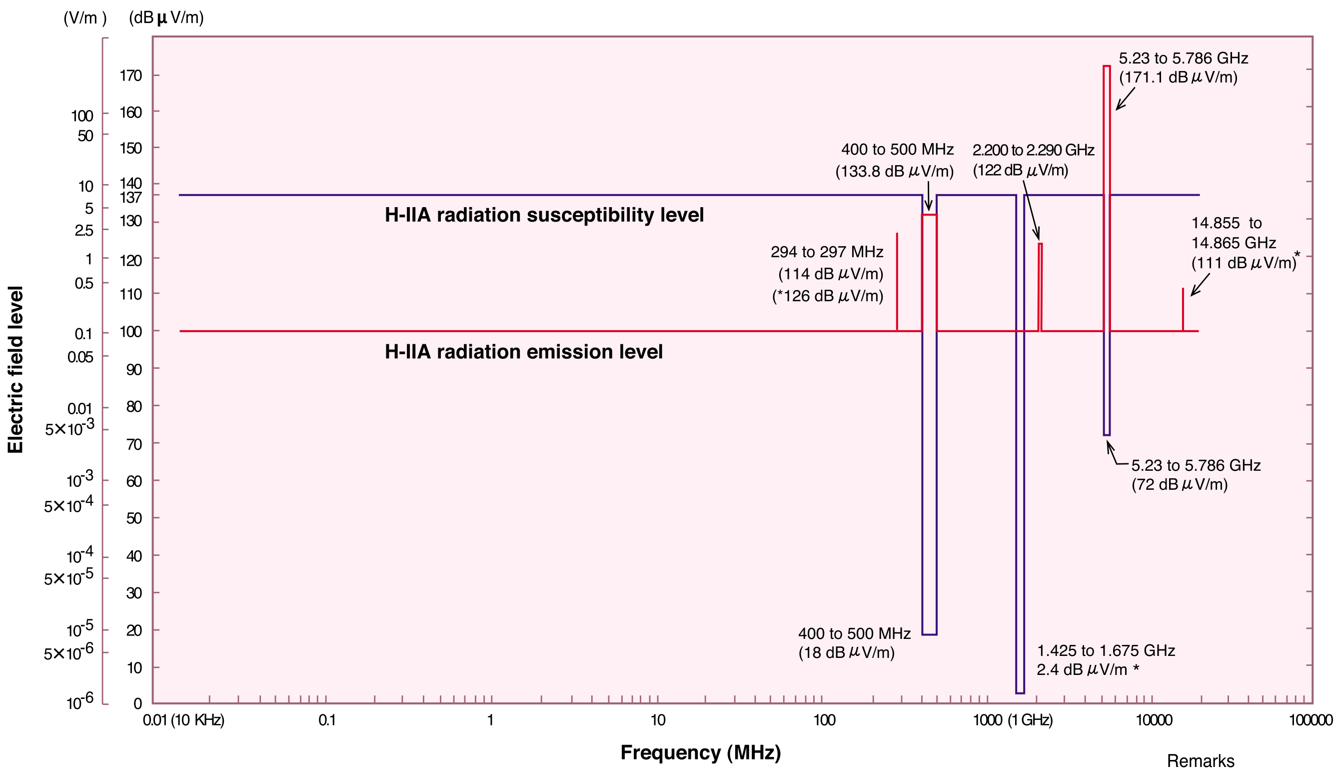
— A614 = 134.3 ± 2.1mm A
— B614 = 164.3 ± 2.1mm
— B864 = 188.0 ± 2.5mm
— C614 = 194.3 ± 2.1mm
— E614 = 217.0 ± 2.1mm
— L614 = 244.4 ± 2.1mm

P : Pin
S : Socket

Shell size
3,7,12,19,27,37,61

Grounding
- : No
G : Yes

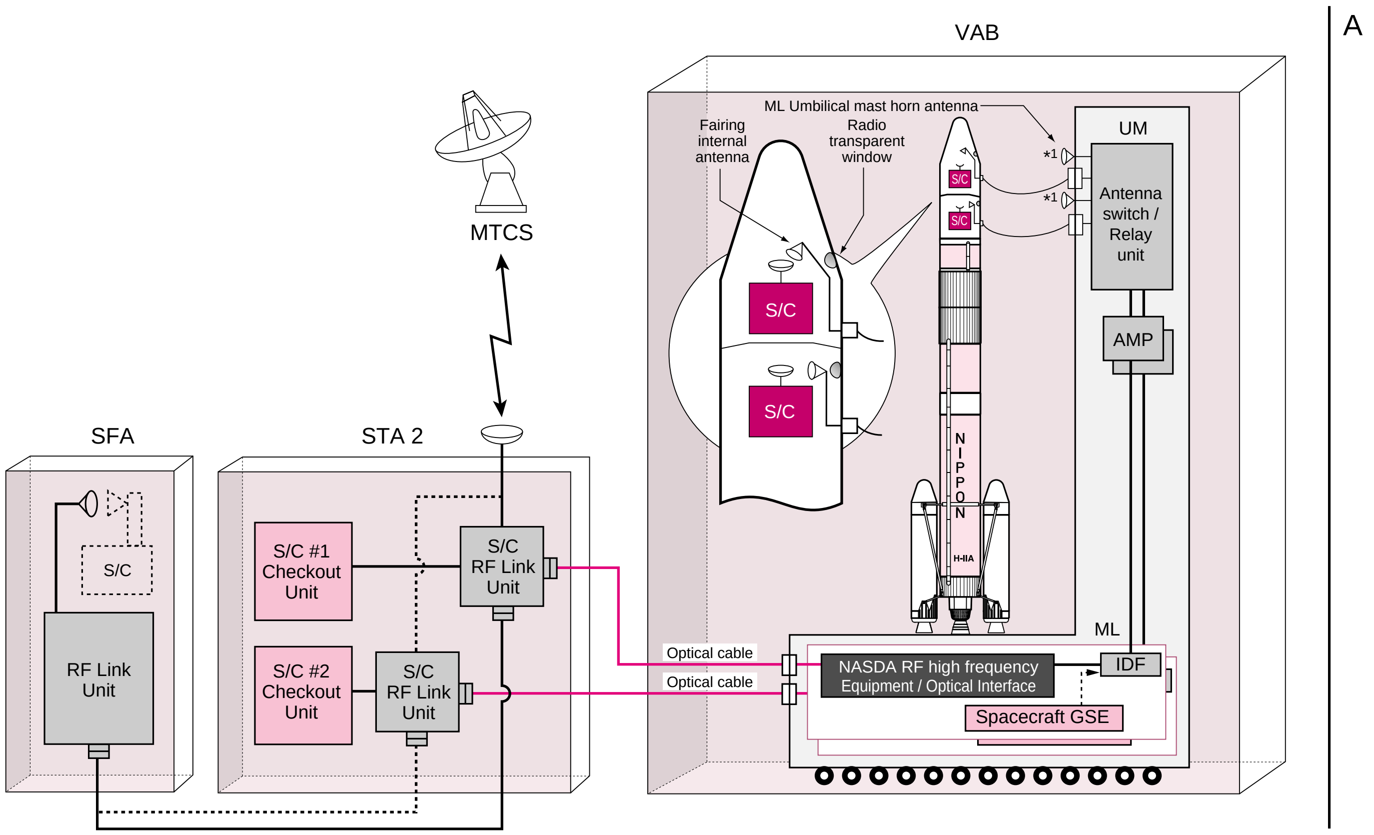
70 ..... Square Flange Receptacle B
74 ..... Single Hole Mounting Receptacle
78 ..... Plug with Lanyard
79 ..... Rack and Panel Plug



C

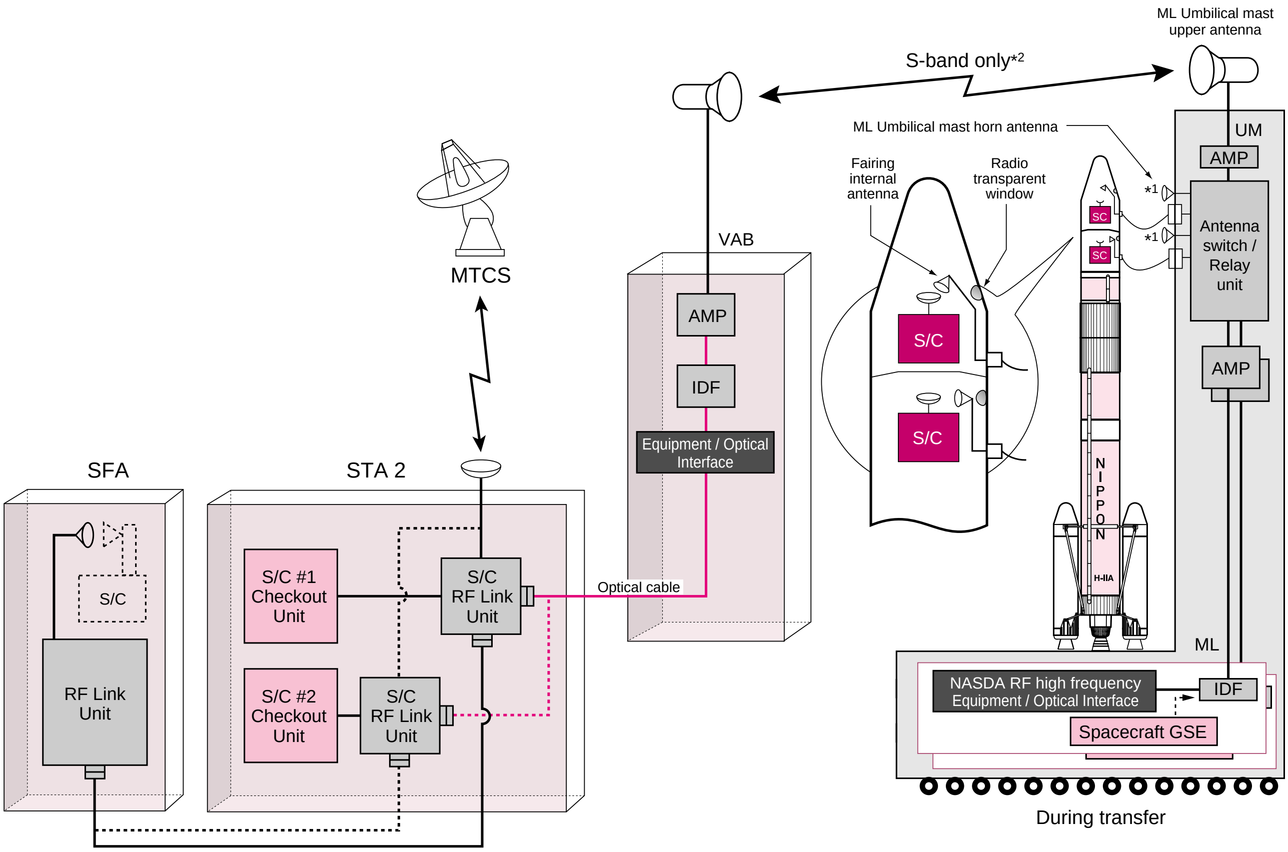
Remarks
* : Option

Figure 4.6.8 Acceptable spurious radiation levels



<Note>
*1 : This antenna is used if necessary.

Figure 4.6.9 VAB RF link schematic (for dual launch)



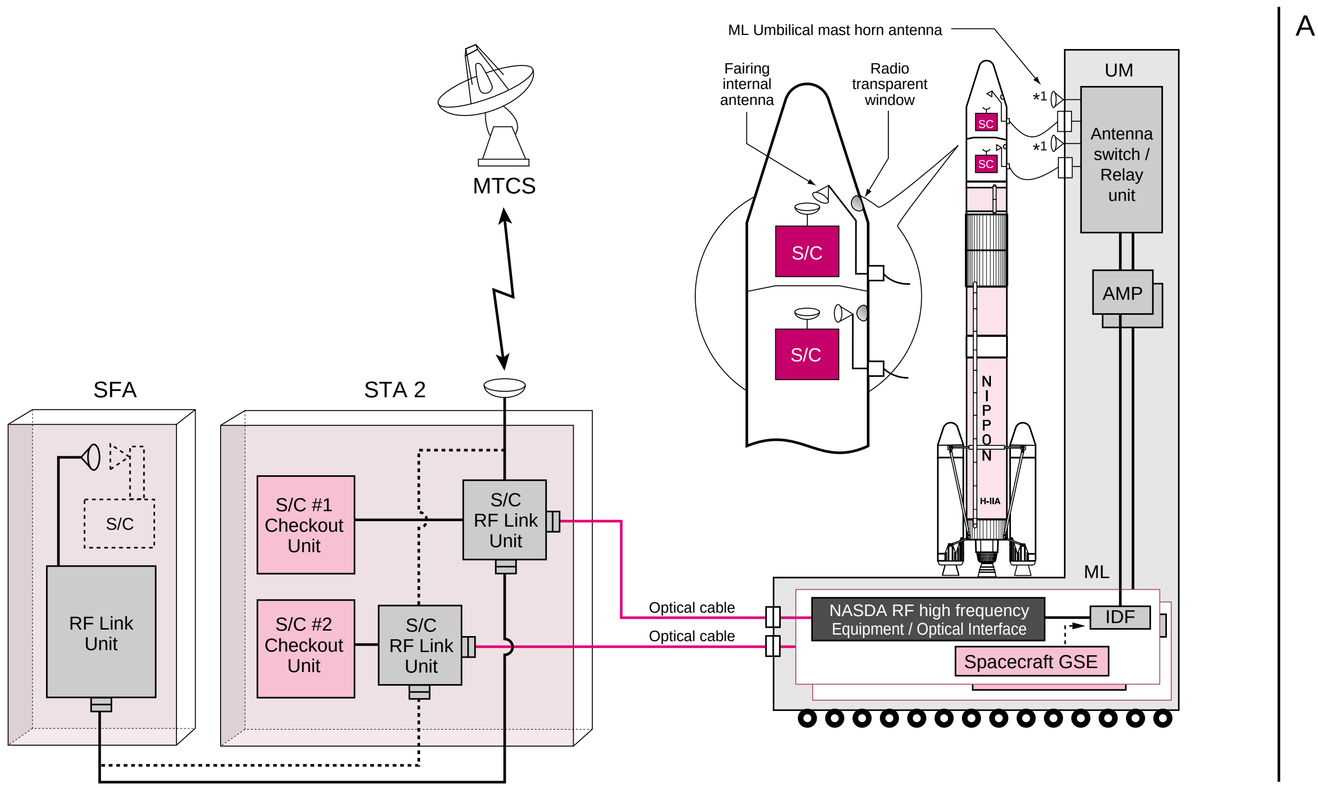
A

<Note>

*1 : This antenna is used if necessary.

*2 : RF link is only for either one of the spacecrafts.

Figure 4.6.10 RF link schematic during ML transfer (for dual launch)



<Note>

*1 : This antenna is used if necessary.

Figure 4.6.11 Launch pad RF link schematic (for dual launch)

4.7 Other Ground Equipment Interfaces

4.7.1 Power

Several types of electrical power are available at the launch complex for spacecraft use.

Commercial AC power is used for basic facility operations. Usable electrical power in the STA2 and the SFA, and after mating to the launch vehicle (on the ML) is described in APPENDIX 2 (A2.3, A2.4 and A2.6).

4.7.2 Liquids and gases

All chemicals to be used will be in compliance with the requirements restricting ozone-depleting chemicals. Gaseous helium (GHe) and gaseous nitrogen (GN_2) are available at STA2, 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.

4.7.3 Propellant / gas sampling and analyzing

| A

Liquids and gases provided for spacecraft use will be sampled and analyzed. Gases, such as hypergolic fuels and oxidizers, water, solvents, and hypergolic decontamination fluids can be analyzed, if necessary.

4.7.4 Filling equipment room

For filling the spacecraft with propellant and pressurized gas, a filling equipment room is available either at SFA and TSA. The filling equipment room is designed for the safety of the operators when hazardous work is performed and the electrical equipment in this room is proof against hydrazine explosion to prevent secondary accidents due to leakage of the propellant from the spacecraft propulsion system.

Air for a protective suit (scape-suit) is usable.

CHAPTER 5.

LAUNCH OPERATIONS

CHAPTER 5. LAUNCH OPERATIONS

5.1 General

5.1.1 Scope

This chapter provides users with information on typical launch operations at the launch site.

The users are required to meet the requirements specified in this chapter and specified separately in the “Launch vehicle payload safety requirements (NASDA-STD-14B)” for spacecraft and in the “Ground Support Equipment (GSE) manual”, with respect to the safety management, the safety design and the launch site operations at Tanegashima Space Center (TNSC).

5.2 Overview of the launch-related organizations

To coordinate the launch services, a member of the “Office of Space Transportation Systems” shall be appointed to a Program Manager for each spacecraft. The Program Manager shall be responsible for contracts and interface coordination (including technical items) of the launch. Figure 5.2.1 shows the NASDA headquarters’ launch operations organization and individual responsibilities.

The user shall appoint a spacecraft interface manager to act as a single contact point with NASDA for the launch service coordination.

The spacecraft interface manager shall be responsible for coordination required after the launch contract is signed.

NASDA shall organize a launch operations team for launch operations to be performed at TNSC.

The Program Manager appointed as a single contact point and the spacecraft interface manager shall be responsible for interface coordination on technical matters before implementation of the launch operations at TNSC before the NASDA launch operation team is established.

The NASDA launch operations team, with the spacecraft interface manager, shall be responsible for interface coordination of technical matters in the launch operations after the NASDA launch operations team is established, via the NASDA Mission Director (NMD) acting as a contact point.

Although the NMD shall be responsible for the technical interface coordination of the launch operations carried out at TNSC, the Program Manager finally commands this coordination.

Figure 5.2.2 shows the relationship between the user and NASDA after establishment of the NASDA launch operations team.

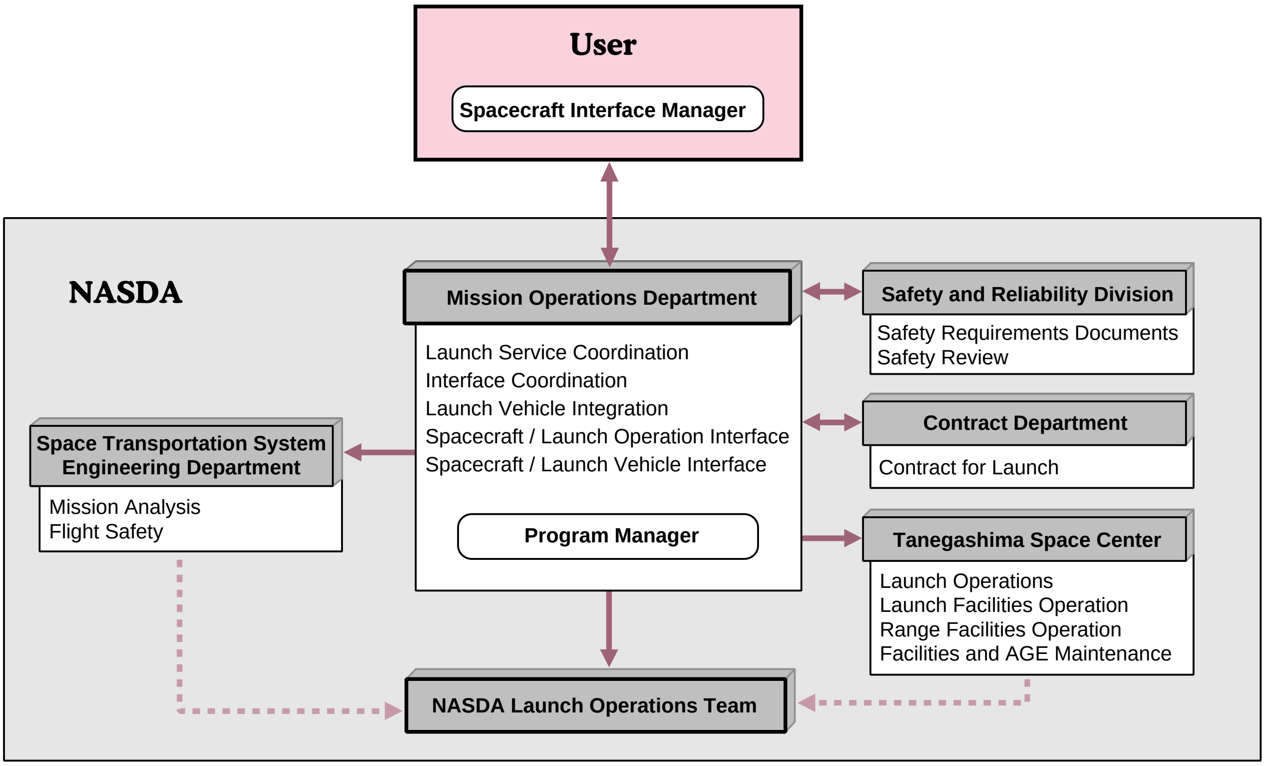


Figure 5.2.1 Launch operations organization of NASDA

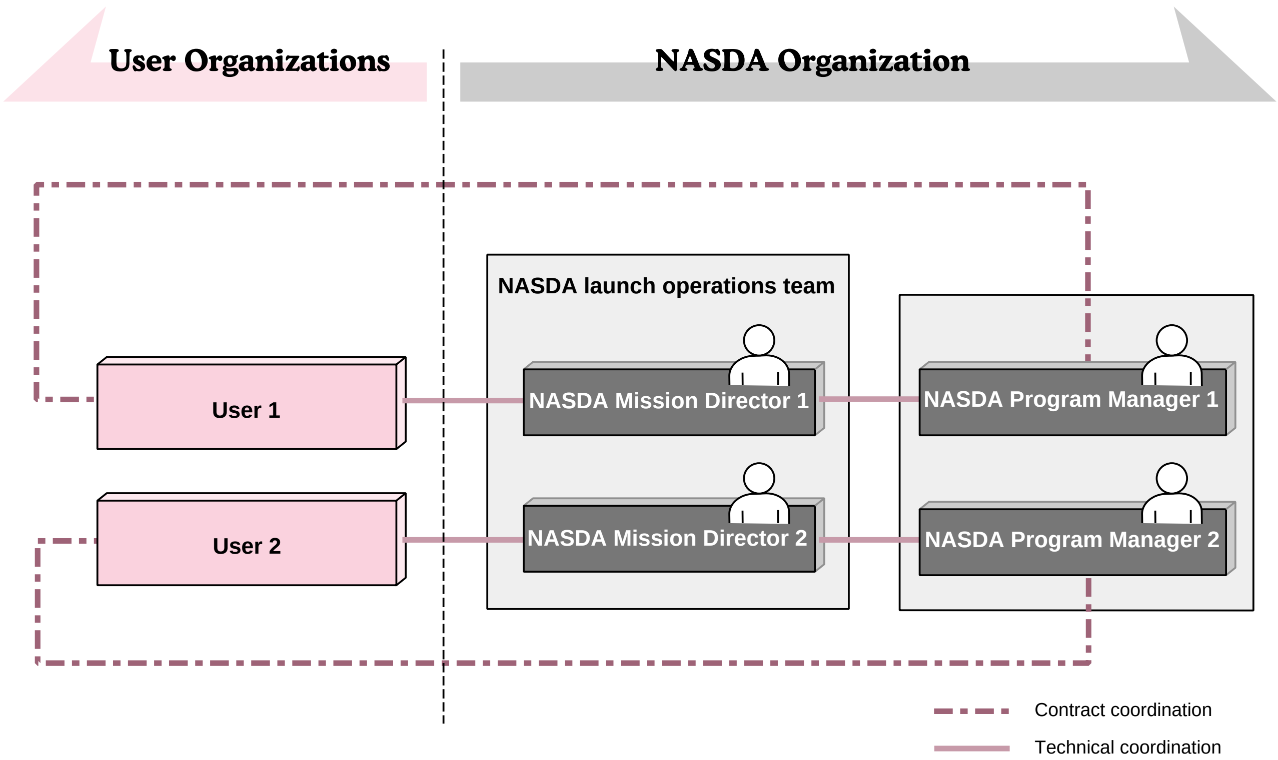


Figure 5.2.2 Relationship between the user and NASDA after establishment of NASDA launch operations team A

5.3 Launch Operations Requirements

The launch operations requirements which specify the safety requirements, operation interface requirements and associated requirements should apply to the design and fabrication of spacecraft, and the launch operations. It should be noted, however, that the information related to the hardware and software of the launch vehicle is not included, for which the Program Manager should be responsible. | C

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

5.3.1 Safety requirements

The spacecraft organization shall meet the requirements specified in the "Launch vehicle payload safety requirements (NASDA-STD-14B)" (safety requirements) separately specified by NASDA with regard to the safety management, the safety design and the launch site operation at TNSC of the spacecraft. (Refer to § 6.5.) | A

5.3.2 Launch operations interface requirements

The launch operations interface requirements are specified in the spacecraft / H-IIA Interface Control Specifications (ICS) related to launch operations to be established separately upon agreement. It should be noted that the interface control specifications for tracking control and associated matters shall be established separately, if they are necessary. (Refer to § 6.3.)

5.4 Responsibility and Organization

The NASDA launch operations team launches the H-IIA vehicle from the Yoshinobu launch complex of Osaki Range in TNSC. The spacecraft preparation shall be under the responsibility of the user. The buildings and the related facilities and GSE to be actually utilized shall be determined when the launch contract is signed.

5.4.1 Launch operations organization

During the launch operations at TNSC, the user is required to appoint an Operations Manager for the spacecraft. The spacecraft organization's Operations Manager (Spacecraft Interface Manager acting as a contact point) shall coordinate the actual operations with the NASDA Mission Director. All coordination prior to the launch operations and coordination of matters (other than technical matters) related to the launch operations after the NASDA launch operations team is established shall be conducted by the spacecraft organization's Operations Manager and the Program Manager.

Figure 5.4.1 shows the organization chart for launch operations except for Y-2 ~ Y-0. A

The countdown (preparation for lift-off) shall be supervised by the NASDA launch conductor (LCDR). The spacecraft organization must appoint the following officers and assign them to the above operations.

(1) Operations Manager for spacecraft

The operations manager for spacecraft conducts all countdown operations of spacecraft, and informs the NASDA General Director of completion of prelaunch operations for the spacecraft.

(2) Spacecraft Conductor (SC)

The Spacecraft Conductor shall get correct information on the progress of the spacecraft operations and issue appropriate instructions for respective operations. The Spacecraft Conductor shall notify LCDR of the spacecraft preparation progress.

The operations prior to the countdown are performed according to the spacecraft organization's network of command. The organization chart should be submitted to NASDA before starting operations at TNSC. (When these operations are carried out by NASDA, NASDA will set up such an organization.)

Figure 5.4.2 shows the organization chart for Y-2 ~ Y-0. A

5.4.2 Responsibility

The operations manager for spacecraft shall be responsible for all the spacecraft operations to be performed at TNSC.

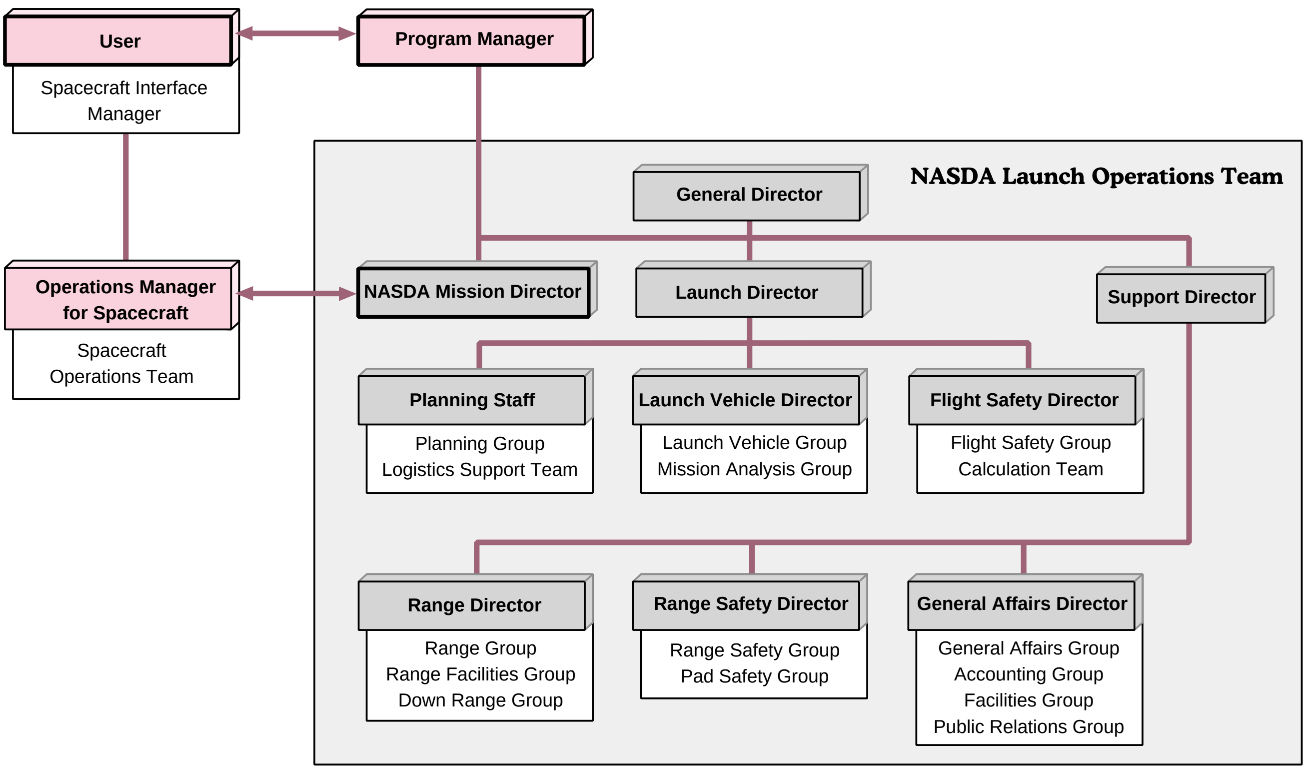
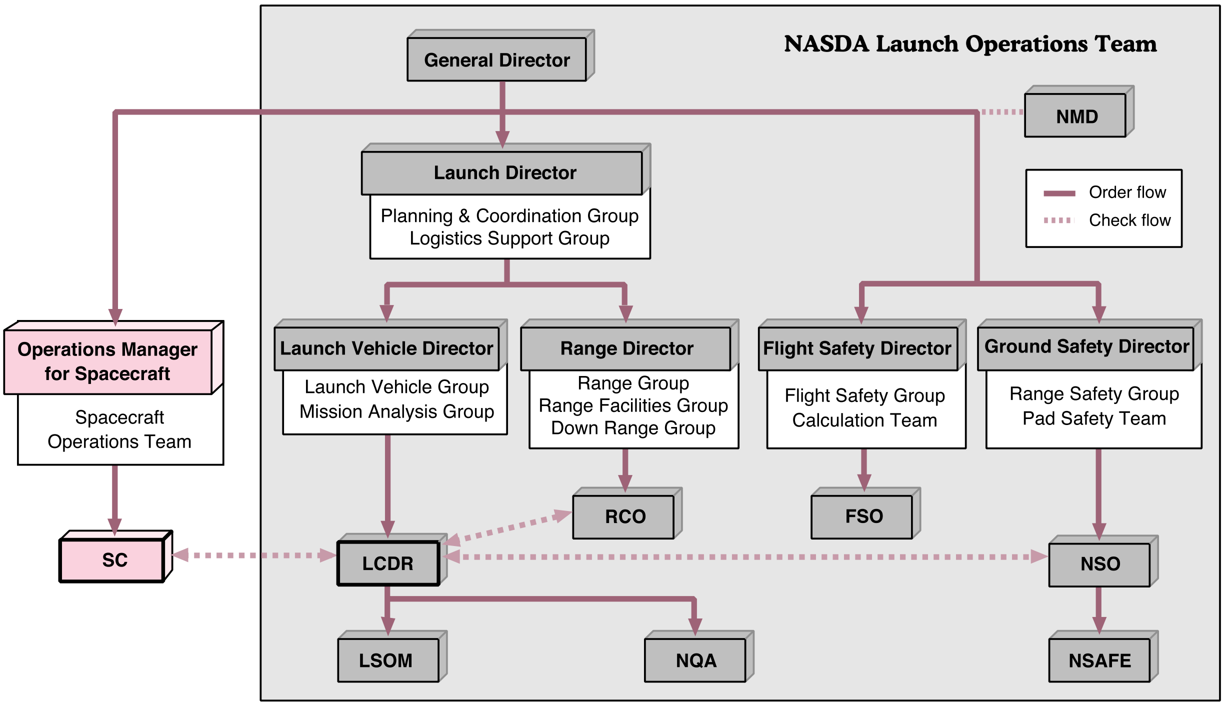


Figure 5.4.1 Organization chart for launch operations (except for Y-2 ~ Y-0) | A



A

SC Spacecraft Conductor
LCDR Launch Conductor

LSOM Launch Site Operations Manager
NQA NASDA Quality Assurance Monitor

RCO Range Control Officer
FSO Flight Safety Officer

NSO NASDA Safety Officer
NSAFE NASDA Pad Safety Officer
NMD NASDA Mission Director

C
A

Figure 5.4.2 Organization chart for Y-2 ~ Y-0

A

5.5 Restrictions

5.5.1 Restrictions on the ground

In order to ensure safety, some spacecraft operations, such as accessing the spacecraft, RF radiation and switching, power interruption, and so on are prohibited in some cases. The launch date is subject to change due to weather, launch vehicle malfunction, or other reasons.

Restrictions on the launch date and launch window are shown in § 5.5.2.

Detailed restrictions shall be coordinated and confirmed at the interface meetings.

5.5.2 Restrictions on launching

The launch date and launch window must be determined by taking many factors into account. This paragraph describes the major restrictions. Details are determined at the interface meetings.

5.5.2.1 Launch window

(1) Launch period

The H-IIA launch vehicle is launched during two periods, June to September and November to February. Further details are to be decided after negotiation with NASDA.

(2) Launch date

The launch date, including the alternative date, shall be set in one of the two launch periods through coordination among NASDA, the users and other concerned organizations.

(3) Launch window

The launch window shall be established by NASDA within the period determined by spacecraft organization analysis (including tracking control), considering restrictions such as shadow and sun angle and other relevant factors.

The launch window shall be set after the final mission analysis is completed.

To maximize launch opportunity, a period of the launch window should be 45 minutes or more.

5.5.2.2 Launch postponement

When the launch vehicle cannot be launched within the launch window on the scheduled date, launch shall be postponed 24 hours or more. Figure 5.5.1 shows a typical case of the operations to be performed on the launch day and the number of days of postponement. Although one day postponement is generally the case for H-IIA launch operation, it may vary depending on the necessary operations and the causes of the postponement. Depending on the causes of postponement, access to the spacecraft may take much time.

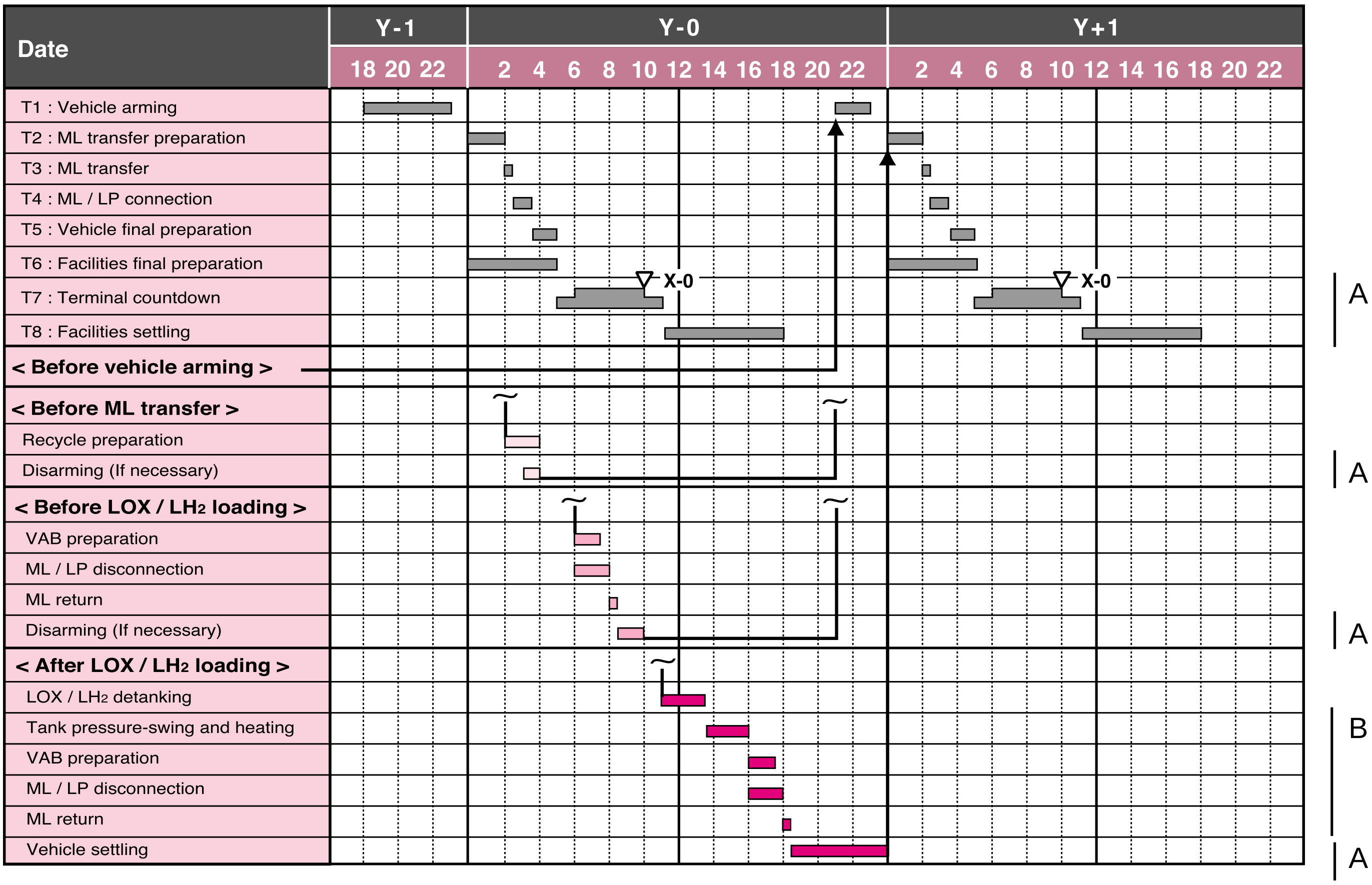


Figure 5.5.1 Typical launch postponement schedule

5.6 TNSC Facilities and GSE Related to Launch Operations of Spacecraft

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

The special facilities and GSE are operated by NASDA persons under the responsibility of the spacecraft organization.

a) Pyrotechnics storage facility and solid propellant storage facility
b) Hazardous material storage (LPSA, LOSA) (propellant storage and loading)
c) STA 1 and STA 2 (spacecraft functional test)
d) Nondestructive Test Facility (NDTF) (solid motor X-ray inspection)
e) Spacecraft and Fairing Assembly Building (SFA) (propellant loading, battery charging, encapsulating into the payload fairing, etc.) A
f) Third stage and Spacecraft Assembly Building (TSA) (propellant loading)
g) Solid Booster Test Building (SBB)
h) Vehicle Assembly Building (VAB) (mating to launch vehicle, battery charging, final checkout, arming, etc.)
i) Movable Launcher (ML) B
j) Takesaki Range Control Center (RCC)
k) Other specifically requested facilities and GSE

Figure 5.6.1 shows the location of spacecraft-related buildings in TNSC's Osaki Launch Range.

A

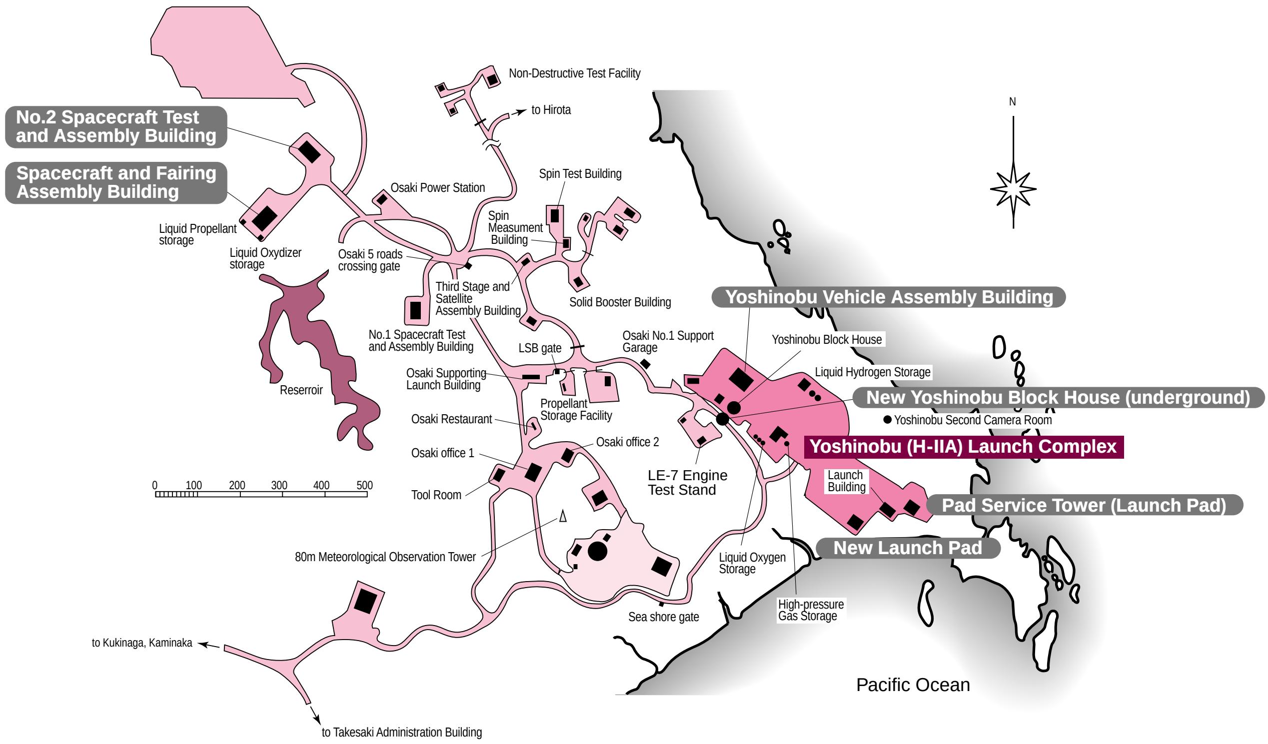


Figure 5.6.1 Location of spacecraft-related buildings in TNSC’s Osaki Launch Range

5.7 Launch Operations

This section describes the typical spacecraft launch operations performed at TNSC.

5.7.1 Spacecraft-related operations and programs

The spacecraft launch operations at TNSC can be broadly classified into three phases:

Phase 1	Spacecraft preparation and functional test
Phase 2	Spacecraft hazardous operations
Phase 3	Joint operations by spacecraft and launch vehicle organizations

Phase 1 operations are independent without any direct interface with the launch vehicle system. They are performed in STA2 and / or STA1.

Phase 2 operations are independent without any direct interface with the launch vehicle system. They consist of hazardous operations and are performed in the SFA, the TSA, the SBB, the NDTF, and others.

Phase 3 operations are performed jointly by the launch vehicle and spacecraft organization, and are performed in the SFA and Yoshinobu Vehicle Assembly Building (VAB).

Generally in the initial planning of spacecraft launch operations, the spacecraft organization is expected to perform all operations from delivery of the GSE (checkout equipment) to TNSC to its removal from TNSC within 45 days (working days) (40 days before the launch, and five days after the launch). The spacecraft / fairing assembly shall generally be mated to the launch vehicle (spacecraft VOS) four days before launch (Y-4).

Figure 5.7.1 shows a typical spacecraft launch operations schedule.

Figure 5.7.2 depicts a 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 or Shimama Seaport, the spacecraft and GSE are transported to TNSC on the public roads. (See Appendix 2.)

The spacecraft delivered by the spacecraft organization to the STA 1 or STA 2 (specified in advance), are unpacked and installed by the spacecraft organization.

NASDA will support these tasks. Hazardous materials, such as solid motors, pyrotechnics, propellant, and explosives, are delivered by the spacecraft organization to the specified place where they are to be stored by NASDA.

Consumable materials required for the operations shall be prepared by the spacecraft organization. Propellant and high-pressure gas may be provided by NASDA with extra charge.

The spacecraft is transported to the test 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 the STA 1 or STA 2. Hazardous operations, such as installing pyrotechnics and loading propellant, must not be performed in these facilities. However, inert high-pressure gas systems can be charged if sufficient safety is ensured (this operation requires permission of the NASDA launch site safety division).

After assembly and functional tests, the spacecraft is loaded by the spacecraft organization into the container for transportation prepared by the spacecraft organization or by NASDA and transported to the SFA or TSA by the spacecraft organization or by NASDA (option). A

Figure 5.7.3 depicts a typical Phase 1 operations flow diagram.

5.7.3 Phase 2 (hazardous operations for spacecraft)

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

Operations in Phases 1 and 2 may be performed in parallel.

Figure 5.7.4 depicts a typical Phase 2 operations flow diagram.

5.7.3.1 Preparing and assembling pyrotechnics and solid motor

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

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

NASDA shall provide support for handling X-ray test equipment of the NDTF, movement of materials, and similar operations requiring use of the NASDA equipment and materials.

After the pyrotechnics have been inspected or installed, the pyrotechnics and the solid motor must be stored in their respective storeroom.

5.7.3.2 Spacecraft operations

(1) Transfer

The spacecraft in the container is transferred by the spacecraft organization or by NASDA (option) from the STA 1 or STA 2 to the SFA or TSA, and is unloaded from the dolly in the air lock entrance room. After the cleanliness in the air lock room has been confirmed, the spacecraft is taken out of the container and is moved into the assembly room (clean room) by the spacecraft organization. A

(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 the SFA or TSA (or in the VAB for special cases) by the spacecraft organization. Also the operations such as depressurization, purging and flushing should be under the responsibility of the spacecraft organization.

Batteries can be charged in the SFA and / or TSA, if hazardous operations are permitted.

(3) Installing the pyrotechnics

The pyrotechnics and the solid motor shall be installed by the spacecraft organization in the SFA and / or TSA, if special permission is obtained from NASDA. However, pyrotechnics wire connection and arming shall be conducted by the spacecraft organization in the VAB as part of the countdown operations.

5.7.3.3 Final spacecraft assembly

(1) Weight measurement

The spacecraft can be weighed in the STA, SFA, or TSA under the responsibility of the spacecraft organization. NASDA has the right to request the spacecraft organization 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 the spacecraft organization.

The spacecraft organization can utilize the weighing equipment of NASDA.

(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 interface meeting; operations by telecommunications signals via the umbilical line or RF signals are allowed.

5.7.4 Phase 3 (joint operations by spacecraft and launch vehicle organizations)

Phase 3 consists of joint operations by spacecraft and launch vehicle organizations.

Operations from mating the spacecraft and the payload support structure to mating the encapsulated spacecraft on the launch vehicle are conducted by the launch vehicle organization; the spacecraft organization provides support and monitoring. Operations after mating are performed by both spacecraft and launch vehicle organizations under various restrictions.

Figure 5.7.5 depicts a typical Phase 3 operations flow diagram for a single launch.

Figure 5.7.6 depicts a typical Phase 3 operations flow diagram for a dual launch.

In case of typical operations, the payload adapter is mated to the spacecraft prior to Phase 3, unless there are any restrictions for the spacecraft.

5.7.4.1 Encapsulation into the payload fairing

The first operations in Phase 3 (joint operations by spacecraft and launch vehicle organizations) are to mate the spacecraft and payload support structure and to encapsulate them into the payload fairing in the SFA. These operations are performed by the launch vehicle organization with the support of the spacecraft organization.

The spacecraft is usually handled together with payload adapter by means of the handling jig prepared by the spacecraft organization. The additional weight of the adapter must be considered in the spacecraft and spacecraft handling jig design.

5.7.4.2 Encapsulation for single launch

After the checkout of the spacecraft in the SFA, the spacecraft with the payload adapter is encapsulated into the payload fairing.

The major operations here are as follows:

(1) The spacecraft with the payload adapter (PLA) is mated to the payload support structure (PSS) and then the ordnance for the spacecraft separation is mounted.
(The PLA can be usually mated to the spacecraft in any phase prior to Phase 3. If the spacecraft does not accept this sequence, however, the PLA is mated to the PSS prior to mating with the spacecraft and then the spacecraft is mated on the top of the PLA in this phase.)

(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-shell is mated in parallel with the spacecraft.) | C
Figure 5.7.7 depicts a typical encapsulation sequence for single launch.

5.7.4.3 Encapsulation for dual launch

After the checkout of the spacecraft in the SFA, two spacecraft are encapsulated into the payload fairing separately.

The major operations here are as follows:

(1) The lower spacecraft with the PLA is mated on a corresponding PSS and then the ordnance for the spacecraft separation is mounted.
(The PLA can be usually mated to the spacecraft in any phase prior to Phase 3. But if the spacecraft does not accept this sequence, the same operations are conducted as a single launch.)

(2) The upper spacecraft with the PLA is mounted with the ordnance for the spacecraft separation in parallel with the lower spacecraft.

(3) 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.

(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 5.7.8 depicts a typical encapsulation sequence for a dual launch.

5.7.4.4 Transportation of encapsulated spacecraft

The encapsulated spacecraft is mounted on the dolly in the SFA air lock room (1) and is transported to the VAB by tractor. It is separated from the dolly on the lower floor of the VAB, and is hoisted. Figure 5.7.9 depicts a typical transportation sequence of the encapsulated spacecraft.

During transportation, NASDA will monitor the conditions to meet the following requirements.

Temperature	:	5 °C to 30 °C
Humidity	:	less than 60 % RH

Vibration : less than 0.6 $G_{O\text{-}P}$ (for each axis)

$G = 9.80665\ (m/s^2)$ | B

If required air conditioning will be prepared by NASDA (option). | A

5.7.4.5 Mating with launch vehicle

After the encapsulated spacecraft is lifted on the upper floor of the VAB, it is directly mated on the top end of the forward skirt 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 the payload fairing opening spring, quick disconnector (QD), and fairing separation pyrotechnics are attached to their positions. | A

Figure 5.7.10 depicts a typical sequence of the mating the encapsulated spacecraft to 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. (Detailed coordination is to be made with the NASDA mission director.) 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 the NASDA launch site safety division. During these operations, turning on the electric system or RF radiation system is prohibited in the launch vehicle system .

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 shall be determined during the joint operation schedule coordination with the NASDA launch site safety division.

5.7.4.7 Y-3 operation | A

The final preparation for the countdown configuration is performed during precountdown operations. Details shall be determined at the interface meetings and the coordination meeting before countdown.

5.7.4.8 Y-2 ~ Y-0 operation | A

The spacecraft organization shall perform the final functional test using the ground line and RF, and charge the battery within the specified time. Details shall be determined at the interface meetings and the coordination meeting before countdown. Figure 5.7.11 shows a typical countdown schedule. The following gives some parts of restricted operations during the countdown.

(1) Hazardous operations

a) Leakage check of the high pressure bottle of the launch vehicle (Y-3)
During this operation, access to the spacecraft is prohibited. (VAB entrance is controlled.) A

b) Loading propellant for the second stage gas jet system of the launch vehicle (Y-2)
During this operation, all of spacecraft operations are prohibited. (VAB entrance is controlled.) C A

c) Pyrotechnics wire connection by the launch vehicle organization (Y-2)
During this operation, spacecraft operations such as switching or turning on the electric and RF radiation system are prohibited.

d) Vehicle body arming by the launch vehicle organization (Y-1)
During this operation, spacecraft operations such as switching or turning on the electric and RF radiation system are prohibited.

e) Pyrotechnics wire connection and removing the SAD safety pin by the spacecraft organization (Y-0)
During these operations, launch vehicle operations such as switching or turning on the electric and RF radiation system are prohibited.

(2) Others

a) ML transfer (Y-0)
During and after this operation, access to the spacecraft is prohibited.

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 interface meetings and the coordination meeting before countdown.

5.7.4.9 Terminal countdown

Figure 5.7.12 shows a typical terminal countdown sequence of the launch vehicle system on the launch date (Y-0). During terminal countdown, the spacecraft organization shall perform the following operations:

(1) Spacecraft RF flight configuration

The final spacecraft RF flight configuration shall be completed before X-10 (T.B.D.) minutes. No change is allowed until 20 seconds after the separation of the spacecraft, unless the launch vehicle organization agrees.

(2) Final spacecraft inspection and Switching the spacecraft power supply

The final spacecraft inspection and switching the spacecraft power supply from an external to infernal source shall be completed before the completion of the spacecraft preparation.

(3) Solid propellant apogee motor arming (when required)

The spacecraft solid propellant apogee motor shall be armed before X-10 minutes.

The arming timing requires agreement with the launch vehicle organization.

(4) Completion of spacecraft preparation

The spacecraft organization shall complete spacecraft preparation incorporated in the launch vehicle sequence before X-270 seconds. B

Completion of the spacecraft preparation is a prerequisite for the automatic countdown sequence.

(5) Automatic countdown sequence

The launch vehicle organization starts the automatic countdown sequence after X-270 seconds. B

(6) Countdown recycle

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

5.7.4.10 Recycle operations (Launch postponement)

If launching 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 the VAB A
c) Disarming spacecraft systems (if required)
d) Disarming launch vehicle systems (if required)
e) Operations required for repetition of Y-0

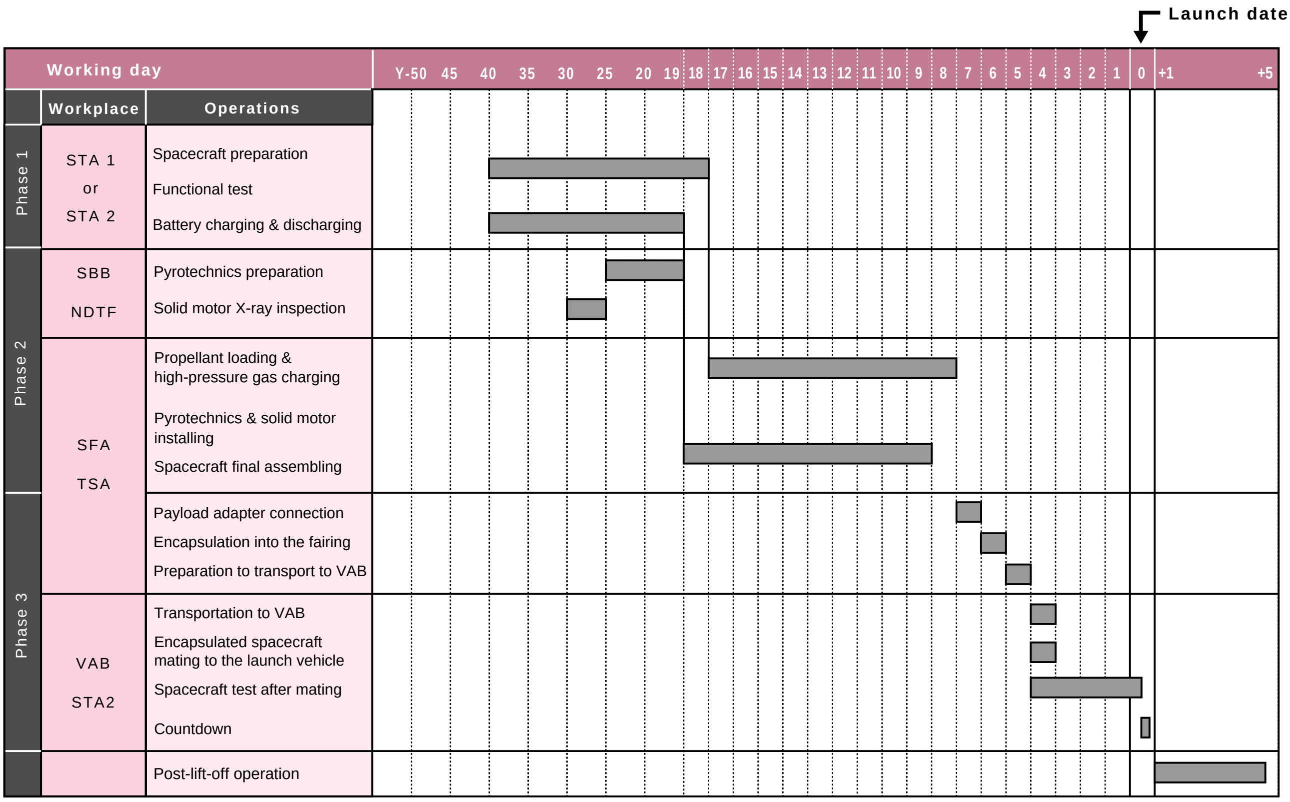


Figure 5.7.1 Typical spacecraft launch operations schedule

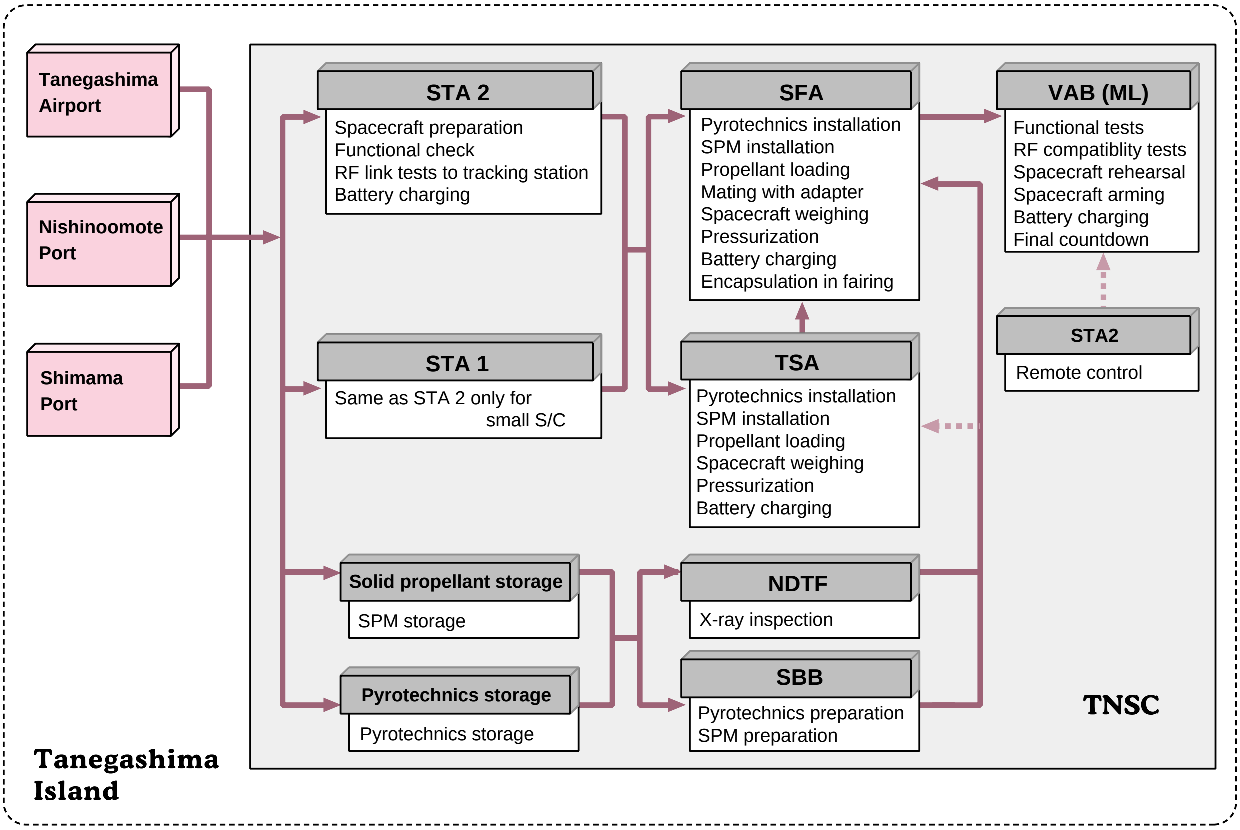
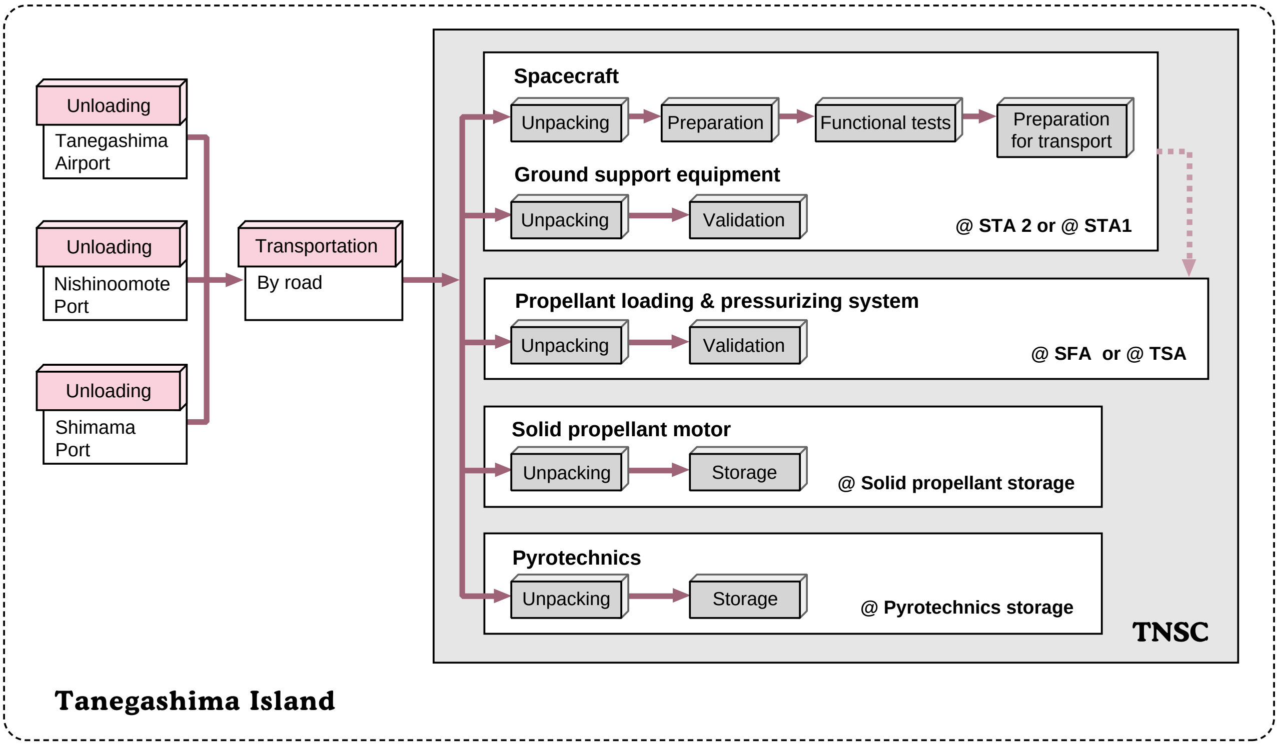


Figure 5.7.2 Typical operations flow diagram for spacecraft preparation at TNSC



| A

Figure 5.7.3 Typical phase 1 operations flow diagram

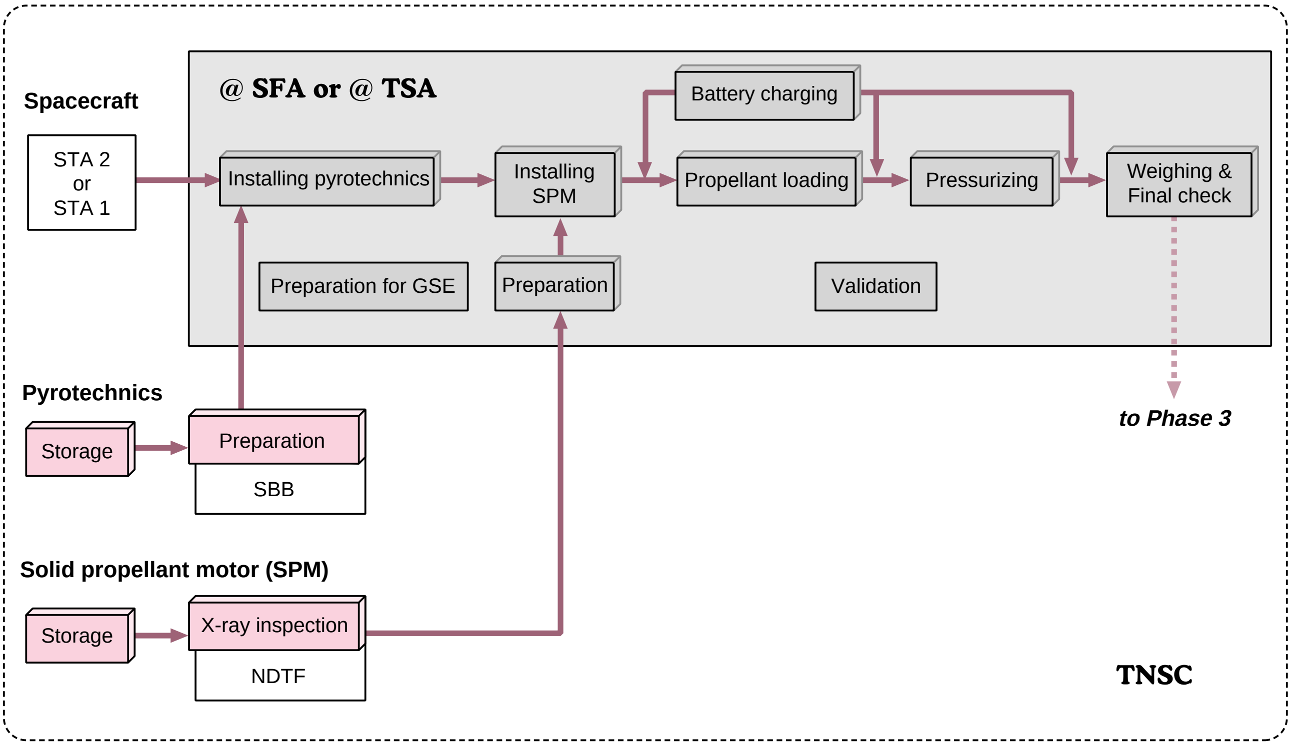


Figure 5.7.4 Typical phase 2 operations flow diagram

Y - 7	Y - 6	Y - 5	Y - 4	Y - 3 ~ 1	Y - 0
Mating with payload support structure Preparation of encapsulation	Encapsulation in fairing	Preparation to transport to VAB	Transportation to VAB Hoisting of encapsulated spacecraft Mating with launch vehicle All system (spacecraft / vehicle) rehearsal (dry run)	Spacecraft test after mating	Countdown ML transfer
@ SFA			**@ VAB**		**VAB → LP**

A

Figure 5.7.5 Typical phase 3 operations flow diagram for single launch (for 4S fairing)

	Y - 10	Y - 9	Y - 8	Y - 7	Y - 6	Y - 5
Upper spacecraft			Preparation of connection to lower fairing	Upper spacecraft connects to lower fairing lid	Encapsulation in Upper fairing	Preparation to transport to VAB
Lower spacecraft	Mating with payload support structure Preparation of encapsulation	Encapsulation in lower fairing		Preparation of encapsulation		
@ SFA						

Y - 4	Y - 3 ~ 1	Y - 0
Transportation to VAB Hoisting of encapsulated spacecraft Mating with launch vehicle All system (spacecraft / vehicle) rehearsal (dry run)	Spacecraft test after mating	Countdown ML transfer
@ VAB		VAB → LP

A

Figure 5.7.6 Typical phase 3 operations flow diagram for dual launch

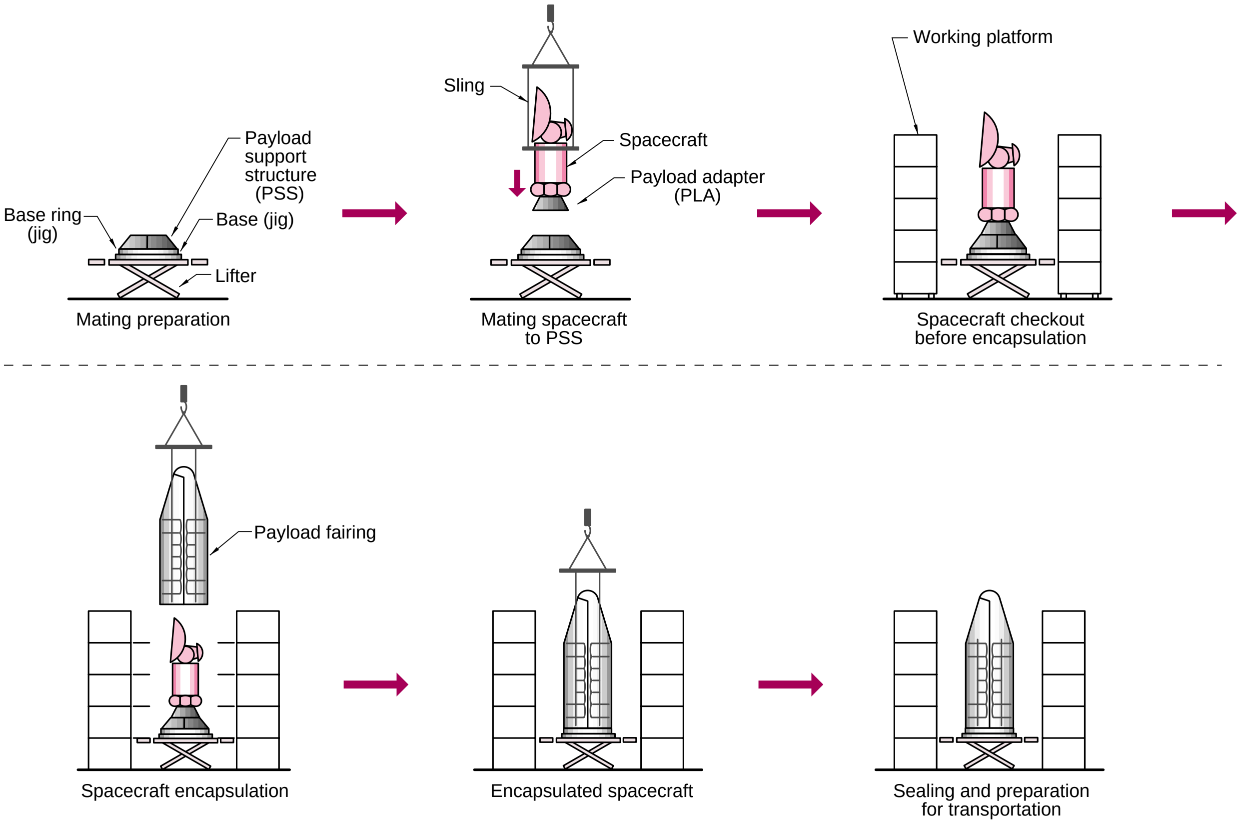
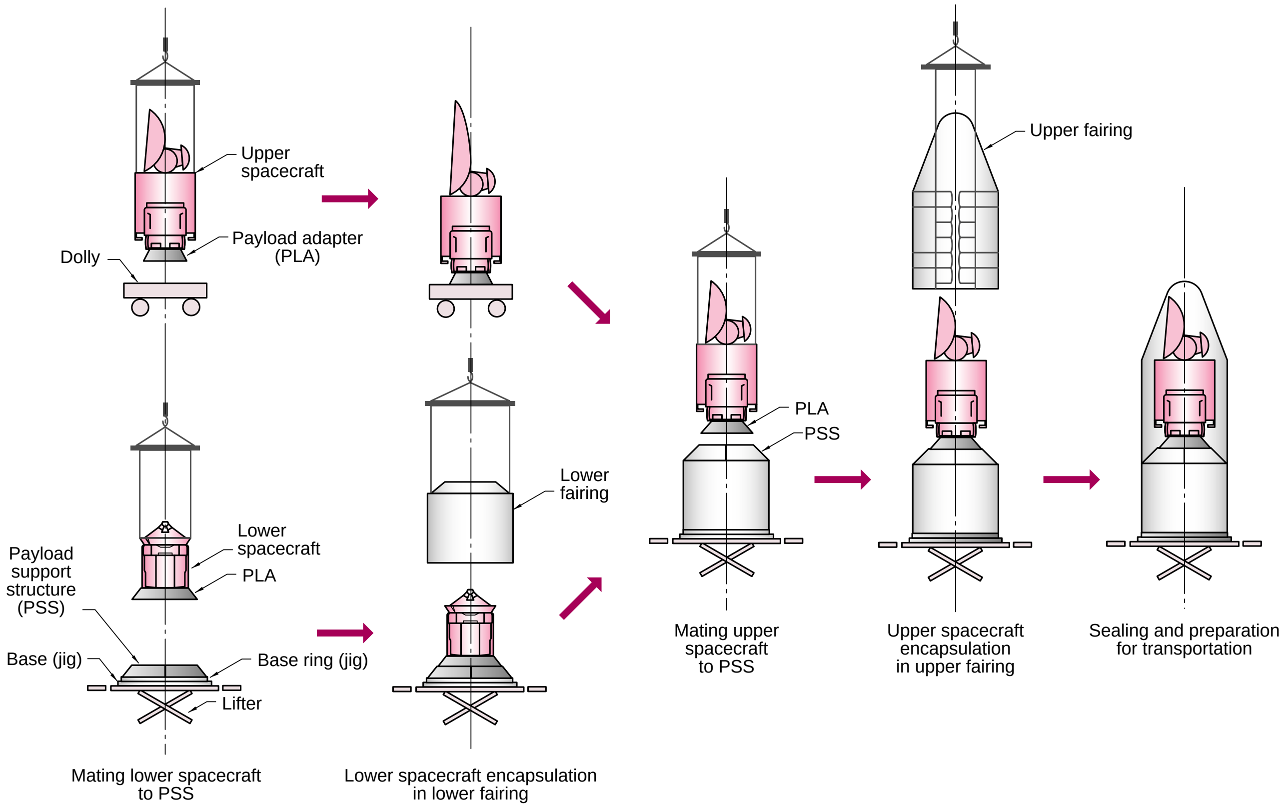
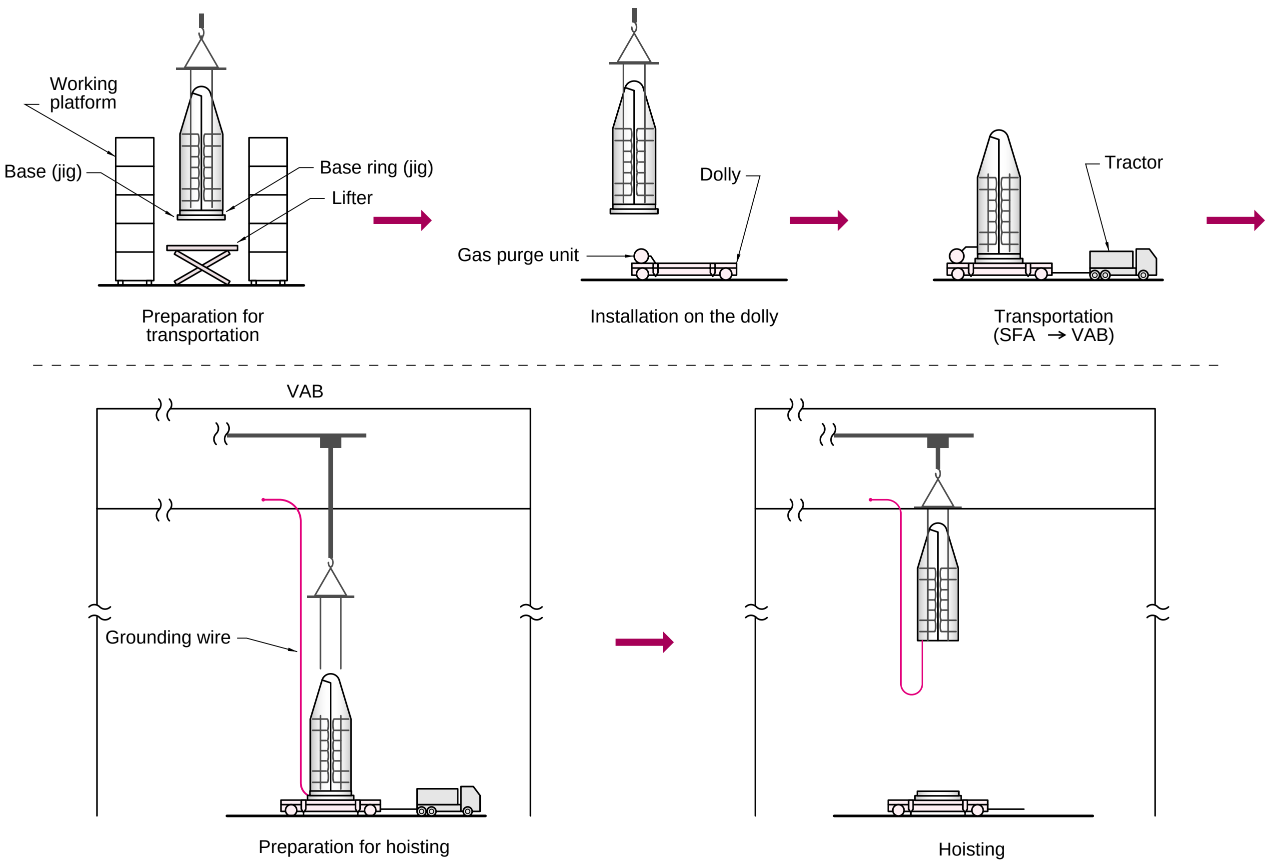


Figure 5.7.7 Typical encapsulation sequence for single launch (for 4S fairing)



| A

Figure 5.7.8 Installation sequence for dual launch



A

Figure 5.7.9 Transportation sequence of the encapsulation spacecraft

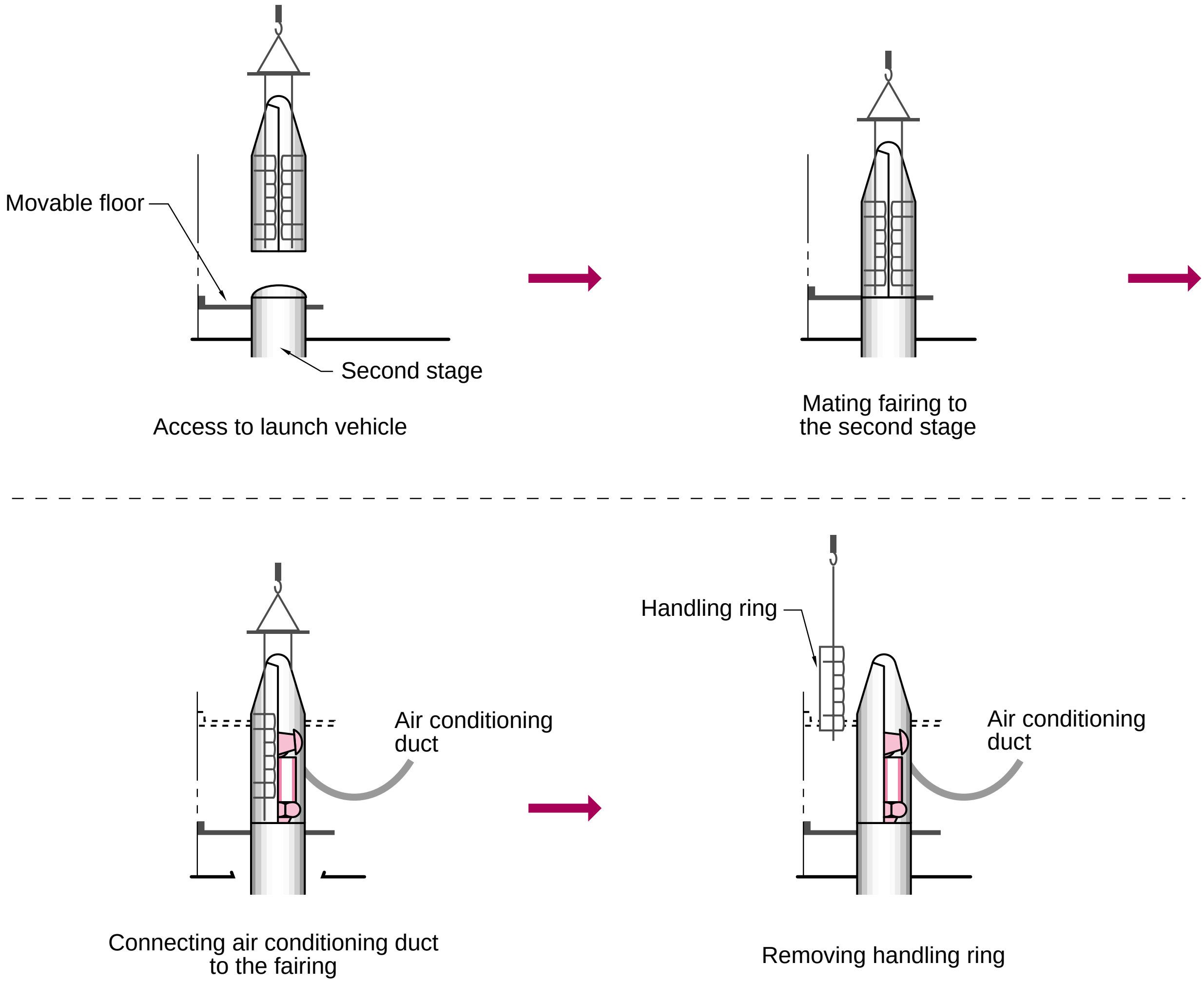
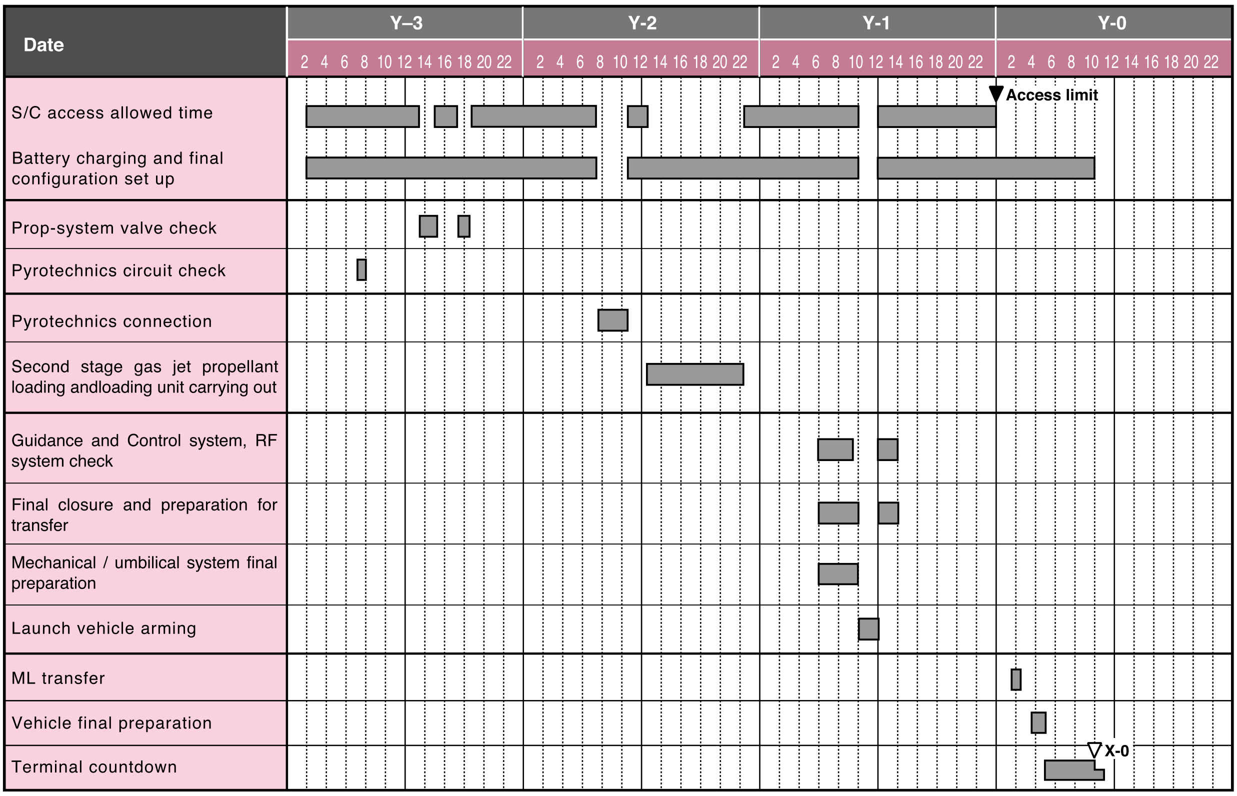
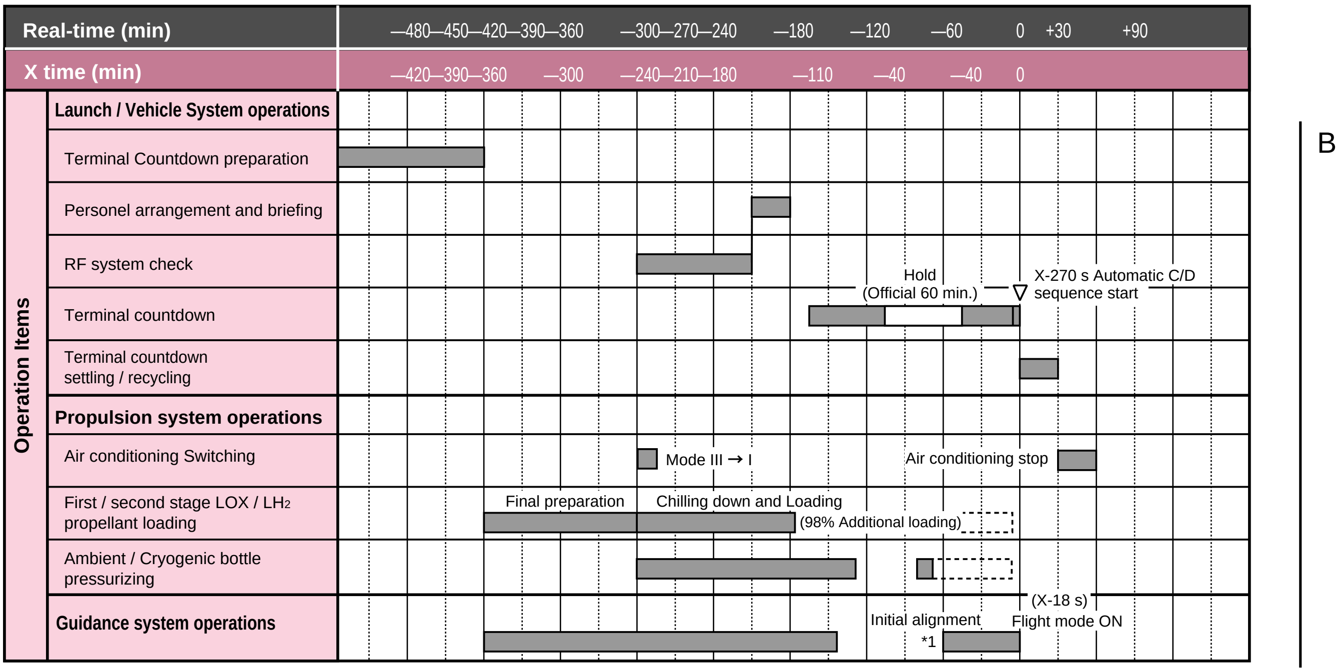


Figure 5.7.10 Encapsulated spacecraft and launch vehicle mating



B

Figure 5.7.11 Typical countdown schedule



(*1 : Perform befor 60 min. Real time)

Figure 5.7.12 Typical launch vehicle system countdown schedule

CHAPTER 6.

INTERFACE MANAGEMENT

CHAPTER 6. INTERFACE MANAGEMENT

6.1 General

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

6.1.1 Launch service organization

The NASDA organization for launch services is described below.

(1) Technical coordination section

Office of Space Transportation Systems

a) Mission Operations Department (MOD) A

The MOD is responsible for general coordination related with launch services as a coordination contact point with the spacecraft organization. In the concrete, the MOD is responsible for the interface between the spacecraft / the H-IIA launch A vehicle and between the spacecraft / launch operations.

Furthermore, the MOD is responsible for procurement of the H-IIA launch A vehicle and also spacecraft safety reviews.

b) Space Transportation System Engineering Department

This section takes charge of mission analyses such as a trajectory analysis, a CLA and so on, and mission modifications of the H-IIA launch vehicle based on each spacecraft's requirements.

(2) Contract section

Contract Department

This section is responsible for a contract related with launch services.

(3) International coordination section

External Relations Department

This section is the contact point in case that an agreement or coordination with the government organization is necessary.

6.1.2 Interface management document

6.1.2.1 Documents to be submitted by the spacecraft organization

The spacecraft organization shall submit to NASDA the following documents which are required for coordinating the interface between the spacecraft and H-IIA launch vehicle system.

Item	When to submit
a. Documents and data required for mission analyses (mission requirements, mass properties, and coupled loads analysis model and integrated thermal analysis model of the spacecraft	Refer to § 6.2.1
b. Schedule of the spacecraft (development, test, launch operation, etc.)	Refer to § 6.5.1 and 6.5.4
c. Safety program plan	Refer to § 6.6
d. Safety data package	Refer to § 6.6
e. Documents required for legal procedures	When NASDA requires

A

6.1.2.2 Document to be submitted by NASDA

After concluding the launch contract, NASDA and the spacecraft organization have an interface meeting, and the results of the agreement between the concerned organizations are incorporated in the "Spacecraft / H-IIA interface control specifications" (hereinafter referred to as "ICS") which specify details of the interface items.

This ICS is maintained and managed by NASDA until launch.

6.2 Interface Work with Spacecraft Organization

Spacecraft / H-IIA launch vehicle interface work is handled by the NASDA MOD. The results of the interface coordination are incorporated in the ICS, whenever applicable. A

6.2.1 Interface schedule / Interface items

The interface schedule is established to define interface tasks including analysis, fit check and review meetings. It specifies the time of the tasks implementation and data exchange.

The details will be coordinated at interface meetings for each mission.

6.2.1.1 Standard mission

Interface schedule and items of the standard mission in which the H-IIA launch vehicle will launch a spacecraft to a standard orbit (such as a geostationary transfer orbit and a polar orbit) are shown in Figure 6.2.1.

6.2.2 Mission analysis

The following analyses are conducted by NASDA for the spacecraft organization.

However, if results of analyses on other similar mission can be adopted, these results are substituted for results of the following analyses.

a) Trajectory analysis
b) Orbit dispersion analysis
c) Sun angle analysis
d) Spacecraft separation analysis
e) Relative trajectory analysis of spacecraft and launch vehicle
f) Spacecraft coupled loads analysis
g) Radio compatibility study
h) Integrated thermal analysis | A

6.2.2.1 Trajectory analysis

Using the mission requirements for the spacecraft organization insertion trajectory (mass properties, sun angle restrictions, requirements at the time of separation, etc.) and launch vehicle data, the trajectory is calculated to meet the mission requirements while satisfying various restrictions, and the major sequence of events and the trajectory parameters at the time of spacecraft separation are supplied to the spacecraft organization.

6.2.2.2 Orbit dispersion analysis

This analysis uses the trajectory worked out in a) to estimate the orbit error at the time of orbit injection caused by error sources including the vehicle body and inertial sensors, and to provide the covariance matrix at the time of spacecraft separation.

6.2.2.3 Sun angle analysis

This analysis uses the trajectory worked out in a) to provide information on the temporal change of the sun angle and eclipse.

6.2.2.4 Spacecraft separation analysis

This analysis provides the attitude angle error and attitude rate at the time of separation by analyzing separation motion of the second stage and spacecraft, based on the most current information on spacecraft mass properties obtained from the spacecraft organization. | C

6.2.2.5 Relative orbit analysis of vehicle and spacecraft

This analysis provides the relative orbit and the relative position between the vehicle and the spacecraft after spacecraft separation. Further this analysis supplies information on pressure and thermal impacts and amount of contamination deposit to which the spacecraft is exposed by the vehicle during the collision avoidance maneuver after the spacecraft separation. A

6.2.2.6 Spacecraft coupled loads analysis

This analysis uses the most current spacecraft dynamic model supplied from the spacecraft organization to obtain the load imposed on the spacecraft and the relative displacement from the payload fairing during launch in combination with the vehicle body model. The analysis result will be supplied to the spacecraft organization. Coordination will be made with the vehicle organization as regards the timing for spacecraft coupled load analysis.

6.2.2.7 Radio frequency compatibility study

RF compatibility is checked based on the H-IIA launch vehicle / spacecraft spurious radiation acceptable levels designated in Figure 4.6.8 and actual data of the spacecraft EMC test.

6.2.2.8 Integrated thermal analysis A

Integrated thermal analysis will be conducted using a thermal model provided by the user if required. This analysis covers the period after mating to the launch vehicle in the VAB up-to the injection into the orbit.

6.2.3 Interface test

6.2.3.1 Fit check of spacecraft and PLA

The fit check of the spacecraft and the payload adapter is conducted by NASDA using the flight model payload adapter or an equivalent model. The payload adapter or test jigs are prepared and transported by NASDA. Time of fit check is given in Figure 6.2.1. Details will be established at the interface meeting. When the equivalent model is used for this check, the fit check using the flight model must be conducted at the launch site prior to launch operations.

6.2.3.2 Umbilical connector disconnection test

A disconnection function between the spacecraft umbilical connector and the payload fairing umbilical connector is checked by the following two tests to verify that normal connection and disconnection are possible. These tests are conducted by NASDA. The test timing is illustrated in Figure 6.2.1.

a) Preliminary disconnection test
b) Launch site disconnection test

(1) Preliminary disconnection test

The preliminary disconnection test is conducted using either of the following two methods:

a) A single connector of the payload fairing (flight model or its equivalent) is transported to the spacecraft, and is subjected to the disconnection test.
b) A single connector of the spacecraft (flight model or its equivalent) is transported to the payload fairing fabrication factory, and is subjected to the disconnection test.

(2) Launch site disconnection test

The umbilical connector disconnection test is conducted after the spacecraft and payload fairing are mated to the second stage of the vehicle at the launch site. This test is applied for only lanyard type connector . A

6.2.3.3 Separation shock test

The separation shock test at the spacecraft separation event is conducted by users using the flight model payload adapter or an equivalent test model.

This test is usually held together with the spacecraft / payload adapter fit check.

Time schedule of the shock test is given in Figure 6.2.1. Details will be established at the interface meeting.

6.2.4 Mission modification

6.2.4.1 Payload fairing mission modification

The following items can be mounted on the payload fairing as requested by the spacecraft organization. The standardized numbers of respective items as described hereinafter are summarized in Table 6.2.1.

Details are described in § 4.4.4.

a) Access doors
b) Large doors
c) Umbilical connectors
d) Radio transparent window
e) Internal antenna
f) External antenna
g) Internal and external antennas
h) Acoustic blankets
i) 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 mounted on the payload adapter as requested by the spacecraft organization.

a) Separation springs
b) Umbilical connectors

6.2.5 Standard services and optional services

Standard services and optional services to launch the spacecraft are provided by the NASDA.

6.2.5.1 Distinction between standard and optional service items

Table 6.2.1 shows a distinction between standard service items and optional service items in the WBS (working breakdown structure) of the mission integration.

6.2.5.2 Notes

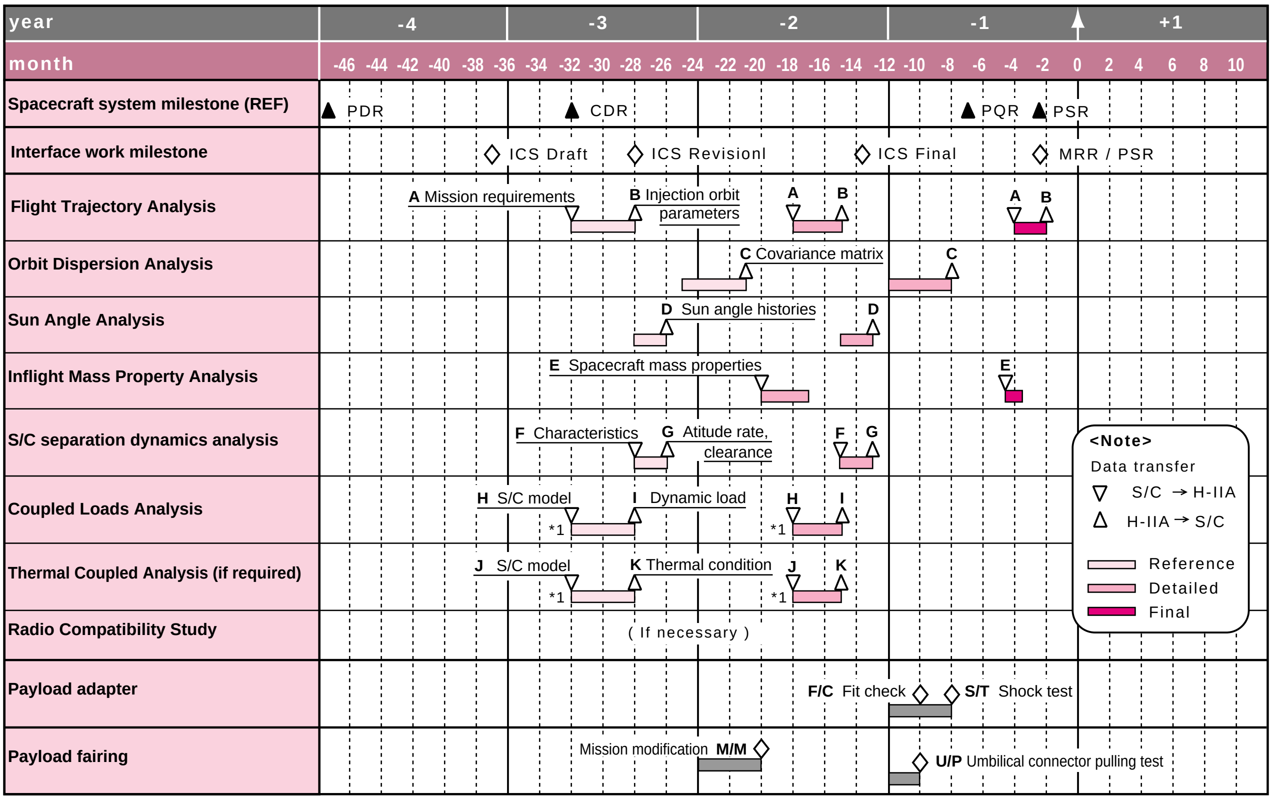
a) The payload adapter selected by NASDA is a standard payload adapter. If the spacecraft organization will use a new payload adapter (PLA), the spacecraft organization is responsible for development and costs of the PLA.

b) The spacecraft organization can utilize the NASDA's PLA for the spacecraft development. But the spacecraft organization is responsible for costs of preparation of the PLA.

c) Standard mission modifications of the mission analysis and the payload fairing are shown in Table 6.2.1. In case that there are additional mission modification items as option, NASDA requires additional costs from the spacecraft organization.

d) Configurations and specifications of spacecraft related facilities and Ground Support Equipment (GSE) of NASDA are shown in the concerned chapter of this manual and H-IIA Payload-Related Facilities and GSE Manual (in preparation).

The spacecraft organization is responsible for costs if modifications of these facilities for the spacecraft are required.



***1 : This item is planned on the spacecraft requirement.**

Figure 6.2.1 Typical spacecraft / H-IIA launch vehicle interface schedule for standard mission

Table 6.2.1(1/6) Mission Integration WBS

WBS No.		Items	Description	Responsibility ※1		Remarks ※2
				Cost	Work	
01		overall				
	-01	interface meeting		SC and LV	SC and LV	standard
02		interface management document				
	-01	ICS		LV (SC)	LV (SC)	standard
	-02	operational plan		SC	SC	SC work
	-03	range safety plan		LV (SC)	LV (SC)	standard
03		flight analysis				
	-01	trajectory analysis				
		SCmass properties	before flight plan,final trajectory	SC	SC	SC work
		analysis	(preliminary trajectory [for new mission]) flight plan,final trajectory	LV	LV	standard
	-02	orbit dispersion analysis	flight plan	LV	LV	standard
	-03	sun angle analysis	(preliminary trajectory [for new mission] and) flight plan	LV	LV	standard
	-04	SC separation analysis	(preliminary trajectory [for new mission] and) flight plan	LV	LV	standard
	-05	relative orbit analysis	(preliminary trajectory [for new mission] and) flight plan	LV	LV	standard
	-06	coupled loads analysis				
		analysis	prior to SC test (normally twice)	LV	LV	standard
	-07	radio compatibility study	prior to SC test	SC and LV	SC and LV	standard
	-08	integrated thermal analysis				
		SC mathematical thermal model		SC	SC	SC work
		analysis	prior to SC test (normally once)	LV	LV	option
	-09	dual launch compatibility analysis		LV	LV	standard

SC : Spacecraft
LV : Launch Vehicle

<Note>
*1. ( ) : support
*2. option : extra cost is required.

Table 6.2.1(2/6) Mission Integration WBS | A

WBS No.		Items	Description	Responsibility ※1		Remarks ※2
				Cost	Work	
04		vehicle manufacture				
	-01	**standard body (H2A202)**	addition for other version	LV	LV	standard
	-02	**fairing (4s)**	addition for other version	LV	LV	standard
	-03	**PLA**	in case of 937M spin, 1194M, 2380S, 3470S	LV	LV	standard
05		PLA				
	-01	**new PLA development**	except for WBS04-03	SC	LV	option
	-02	**mission modification**				
		number of separation springs	standard:4 to 8	LV	LV	standard
		umbilical connector on PLA	to be prepared by LV or SC	LV	SC or LV	standard
		number of separation switches	standard : 2	LV	LV	standard
06		fairing				
	-01	**mission modification**				
		number of access doors	standard : 4 (2 for lower fairing)	LV	LV	standard
		number of large doors	4 for vehicle operation in principle	LV	LV	standard
		number of umbilical connectors	selective PLA / PLF type	LV	LV	standard
		RF link method				
			number of internal antennas (standard : 1)	LV	LV	standard
			number of radio transparent windows (standard : 1)	LV	LV	standard
			number of internal / external antennas (standard : none)	SC	LV	option
			other methods (standard : none)	SC	LV	option
		acoustic insulation blankets	10 mm	LV	LV	standard
		air conditioner		LV	LV	standard

| B

| A

| C

SC : Spacecraft
LV : Launch Vehicle

<Note>
***1. ( ) : support**
***2. option : extra cost is required.**

Table 6.2.1(3/6) Mission Integration WBS

A

WBS No.		Items	Description	Responsibility ※1		Remarks ※2
				Cost	Work	
07		**interface verification test**				
	-01	**SM and PFM s/c test plan**				
		test for launch vehicle		LV	LV	standard
	-02	**mechanical compatibility test / static loads • vibration test**	PLA for test to be prepared by SC in principle			
		PLA preparation and transportation	PLA will be re-fablicated or newly developed	SC	LV	option
			(standard : no separation spring and no connector)			
		AGE and jigs preparation	SC-PLA mating portable stand	SC	SC	SC work
		mating operation		SC	LV (SC)	option
		test		SC	SC	SC work
	-03	**separation shock test**				
		PLA preparation / transportation	PLA will be re-fablicated or newly developed	SC	LV	option
		pyrotechnics preparation		SC	SC or LV	SC work or option
		AGE and jigs preparation	SC-PLA mating portable stand	SC	SC	SC work
		mating operation		SC	LV (SC)	option (SC work)
		test		SC	SC	SC work
	-04	**fit check of flight component**	standard joint work of LV and SC			
		PLA preparation / transportation		LV	LV	standard
		AGE and jigs preparation	SC-PLA mating portable stand	SC	LV	option
		fit check		LV and SC	LV and SC	standard
	-05	**disconnect functional test of umbilical connectors**				
		umbilical connector preparation	to be prepared by LV or SC	SC or LV	SC or LV	SC work or standard
		AGE and jigs preparation	SC-PLA mating portable stand	SC	LV	option
		test		LV and SC	LV and SC	standard
	-06	**test result of STM and PFM**				
		for LV		LV	LV	standard

SC : Spacecraft
LV : Launch Vehicle

<Note>
***1. ( ) : support**
***2. option : extra cost is required.**

Table 6.2.1(4/6) Mission Integration WBS

A

WBS No.		Items	Description	Responsibility ※1		Remarks ※2
				Cost	Work	
08		safety reviews				
	-01	**SAFETY SUBMISSION**	in every phase	SC	SC	SC work
	-02	**reviews**		LV	LV	standard
09		support for legal procedure	if necessary for foreign SC			
	-01		law for high-pressure gas, radio, pyrotechnics, etc.	SC	LV	option
10		modification of facilities	in response to requirements not specified in user's manual	SC	LV	option
11		launch operations	standard working days : within 45 special consideration for over 45 days			
	-01	**interface document**				
		SC launch operation plan		SC	SC	
		joint operation plan		LV (SC)	LV (SC)	
	-02	**phase 0**	preparation for SC operation			
		facilities validation				
			test except GSE (validation cable)	LV	LV	standard
			test including GSE	SC	SC	SC work
	-03	**phase 1**	SC independent work			
		fitting pyrotechnics		SC	SC	SC work
		visual inspection		SC	SC	SC work
		final assembly of SC		SC	SC	SC work
		inspection after final assembly		SC	SC	SC work
	-04	**phase 2**	SC independent hazardous work			
		fitting pyrotechnics		SC	SC	SC work
		preparation of loading equipment	to be prepared by SC side in principle (including NASDA's equipment)	SC	SC	SC work
		inspection of loading equipment		SC	SC	SC work
		propellant loading and high-pressure gas charging		SC	SC	SC work
		post-treatment of loading equipment		SC	SC	SC work
		storage of loading equipment		SC	LV or SC	option or SC work
		treatment of waste		SC	LV or SC	option or SC work

C

SC : Spacecraft
LV : Launch Vehicle

<Note>
***1. ( ) : support**
***2. option : extra cost is required.**

Table 6.2.1(5/6) Mission Integration WBS

A

WBS No.		Items	Description	Responsibility ※1		Remarks ※2
				Cost	Work	
11	-05	**phase 3**	joint work of LV and SC (costs are shared)			standard (SC work)
		SC / PLA mating		LV (SC)	LV (SC)	standard (SC work)
		PLA / PSS mating		LV (SC)	LV (SC)	standard (SC work)
		SC encapsulation into the fairing		LV (SC)	LV (SC)	standard (SC work)
		SC / fairing mating onto the second stage		LV (SC)	LV (SC)	standard (SC work)
		SC system test				
			testing	SC	SC	SC work
			operation of elevator type floor	LV	LV	standard
		Y-0 reheasal		SC and LV	SC and LV	SC work and LV work
		Y-2 ~ Y-0 operation		SC	SC	SC work
	-06	**analysis of liquid and gas**				
		gas analysis		SC	LV	option
		propellant analysis				
			chemical	SC	LV	option
			contamination	SC	LV	option
		IPA circulation and filtering		SC	LV	option
		preparation / operation	post treatment of scrubber and bubbler	SC	LV	option
	-07	**consumable materials**				
		propellant		SC	LV or SC	option or SC work
		high-purity gas and IPA, etc.	for purging, flushing, etc.	SC	LV or SC	option or SC work
		others		SC	LV or SC	option or SC work
	-08	**SC facility operation**				
		supporting operators	one person for standard working day (within 45)	LV	LV	standard
		driver for special motor vehicle	if needed	SC	LV (SC)	option (SC work)

A

SC : Spacecraft
LV : Launch Vehicle

<Note>
***1. ( ) : support**
***2. option : extra cost is required.**

Table 6.2.1(6/6) Mission Integration WBS

A

WBS No.		Items	Description	Responsibility ※1		Remarks ※2
				Cost	Work	
11	-09	**SC transportation in launch site**				
		SC handling		SC	SC (LV)	SC work (option)
		canistar preparation	dia 5m x 10mH	SC	LV	option
		transportation	operation	SC	LV	option
	-10	**other services**				
		security control in SC area		SC	LV	option
		clean tent	class 1000	LV	SC	standard
		equipment calibration	user's equipment	LV	LV	standard
		refrigerator for battery	preparation	LV	LV	standard
			operation	LV	SC	standard
		FALSE floor for SC checkout equipment				
		telephone • facsimile	extension	LV	LV	standard
			outer line	SC	LV	option
		copy	machine preparation	SC	LV	option
			copying paper preparation	SC	LV	option
		SC storage for launch postponement		SC	SC and LV	SC work and option
		modem				
		Blast Shield				
		test connector				
		customer's sticker on the PLF		SC	LV	option
		protective suit and dust-proof suit	special suits to be prepared by SC, if required for SC	SC or LV	SC or LV	SC work or standard
		special cleaning of clean room	if specially required	SC	LV	option
		umbilical line change		LV	LV	standard
12		**orbit parameter of SC separation**	within 30 minutes after separation	LV	LV	standard
13		**report of post-flight analysis**				
		orbit parameter after SC separation (report of SC tracking data)	within one week after launch	SC	SC	SC work

SC : Spacecraft
LV : Launch Vehicle

<Note>
***1. ( ) : support**
***2. option : extra cost is required.**

6.3 Spacecraft / H-IIA and Spacecraft / Launch Operations Interface Control

6.3.1 Interface control document

The “Spacecraft / H-IIA interface control specifications (ICS)” related to launch capability, interface restrictions and launch operations are edited and maintained through the interface meetings to be 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 the table of contents for the ICS.

6.3.2 Coordination items and timing

The master schedule is prepared according to results of coordination with each spacecraft organization by NASDA.

Table 6.3.1 Standard spacecraft / H-IIA interface control specifications

0 SCOPE

1 INTRODUCTION
- 1.1 H-IIA MISSION
- 1.2 SPACECRAFT MISSION OBJECTIVES
- 1.3 SPACECRAFT DESCRIPTION
- 1.4 H-IIA DESCRIPTION
  - 1.4.1 Outline of the H-IIA Launch Vehicle
  - 1.4.2 H-IIA First Stage
  - 1.4.3 H-IIA Second Stage
  - 1.4.4 H-IIA Guidance and Control System
  - 1.4.5 Payload Fairing
  - 1.4.6 Payload Adapter
- 1.5 OSAKI RANGE DESCRIPTION
  - 1.5.1 Yoshinobu Launch Complex
  - 1.5.2 Spacecraft and Fairing Assembly Facilities

2 APPLICABLE DOCUMENTS
- 2.1 SPECIFICATIONS
- 2.2 INTERFACE DRAWINGS
- 2.3 RELEVANT DOCUMENTS
- 2.4 DOCUMENT AMENDMENT PROCEDURE
  - 2.4.1 Introduction
  - 2.4.2 Procedure

3 REQUIREMENTS
- 3.1 GENERAL
- 3.2 MISSION CHARACTERISTICS
  - 3.2.1 Launch Capability
  - 3.2.2 Launch Date
  - 3.2.3 Launch Vehicle Configuration
  - 3.2.4 Launch Window
- 3.3 INJECTION ORBIT PARAMETERS AND CONDITIONS
  - 3.3.1 Flight Plan
  - 3.3.2 Orbit Parameters
  - 3.3.3 Injection Accuracy
  - 3.3.4 Conditions at Separation
- 3.4 SPACECRAFT CONFIGURATION AT LAUNCH
- 3.5 DATA ACQUISITION IN LAUNCH PHASE
- 3.6 SPACECRAFT MASS AND DYNAMIC PROPERTIES
  - 3.6.1 Fundamental Frequency
  - 3.6.2 Primary Structure
  - 3.6.3 Secondary Structure and Flexible Elements
- 3.7 SPACECRAFT DIMENSIONING REQUIREMENTS
  - 3.7.1 Mass Constraints
  - 3.7.2 CoG Constraints
  - 3.7.3 Inertia Constraints (T.B.D.)
  - 3.7.4 Balancing Constraints
- 3.8 MECHANICAL INTERFACES
  - 3.8.1 Axis Definition
  - 3.8.2 Assembly Characteristics
  - 3.8.3 Access and Mounting
- 3.9 ENVIRONMENTAL CONDITIONS UNDER THE FAIRING DURING LAUNCH PREPARATION
  - 3.9.1 Air Conditioning
  - 3.9.2 Thermal Characteristics
- 3.10 IN FLIGHT ENVIRONMENTS
  - 3.10.1 Mechanical Environments
  - 3.10.2 Other Environments
- 3.11 ELECTRICAL INTERFACES
  - 3.11.1 Earth Potential Continuity
  - 3.11.2 Umbilical Link
  - 3.11.3 Electrical Link to the H-IIA Second Stage
- 3.12 RF LINK INTERFACE
  - 3.12.1 Spacecraft TX / RX Characteristics
  - 3.12.2 H-IIA TX / RX Characteristics
  - 3.12.3 Spacecraft Transmission Plan
  - 3.12.4 H-IIA Transmission Plan
  - 3.12.5 Radio EMC Compatibility
  - 3.12.6 Operational Constraints
  - 3.12.7 Radio Link Requirements
- 3.13 FLUID INTERFACE
- 3.14 PYROTECHNIC INTERFACE

4 SPACECRAFT PREPARATION FACILITY INTERFACES
- 4.1 OPERATION FLOW
  - 4.1.1 Spacecraft Operations
  - 4.1.2 Combined Operations
- 4.2 BUILDING STA1 / STA2 (Spacecraft Test and Assembly)
- 4.3 BUILDING SFA (Spacecraft and Fairing Assembly)
- 4.4 BUILDING TSA (Third Stage and Spacecraft Assembly)
- 4.5 BUILDING VAB (Vehicle Assembly)
- 4.6 ML (Mobile Launch table)
- 4.7 RF / VIDEO / DATA LINKS IN EACH BUILDING
- 4.8 ELECTRICAL LINKS IN EACH BUILDING
- 4.9 TELECOMMUNICATION IN EACH BUILDING
- 4.10 SPACECRAFT TRANSPORT CONTAINER
  - 4.10.1 Canister Description and Interface
  - 4.10.2 Requirements during Transfer on Site
- 4.11 OTHERS

5 ENVIRONMENTAL CONDITIONS, INTERFACES DURING TRANSFER
- 5.1 TRANSFER WITHOUT FAIRING
  - 5.1.1 Environmental Conditions during Transportation
- 5.2 TRANSFER WITH FAIRING
  - 5.2.1 Environmental Conditions during Transportation
- 5.3 TRANSFER FROM VAB TO LAUNCH PAD (LP)
  - 5.3.1 General
  - 5.3.2 Ground Environments
  - 5.3.3 Interfaces

6 SPACECRAFT / LAUNCH AREA INTERFACE
- 6.1 SPACECRAFT / LP INTERFACE
  - 6.1.1 Environmental Conditions
  - 6.1.2 Access Facilities / Requirements
  - 6.1.3 RF / Video / Data Links
  - 6.1.4 Umbilical Links
  - 6.1.5 Main Power
  - 6.1.6 Fluids
  - 6.1.7 Telecommunications
- 6.2 SPACECRAFT / BLOCK HOUSE INTERFACE

7 RANGE FACILITIES
- 7.1 TRANSPORT AND HANDLING
- 7.2 FLUID AND PROPELLANTS
- 7.3 TECHNICAL SUPPORT
- 7.4 SAFETY FACILITIES
- 7.5 RANGE COMMUNICATIONS
- 7.6 TELECOMMUNICATIONS
- 7.7 BUILDING FACILITIES
- 7.8 STORAGE AREA
- 7.9 MISCELLANEOUS

8 PLANS
- 8.1 DEVELOPMENT AND TEST PLAN
- 8.2 LAUNCH OPERATION PLAN
- 8.3 PLANNING CONTROL
  - 8.3.1 Planning Control
  - 8.3.2 Launch Site Meeting

9 LAUNCH OPERATION REQUIREMENTS
- 9.1 REQUIREMENT FOR WASH OF TEST EQUIPMENT
- 9.2 REQUIREMENT FOR ANALYSIS OF GAS & LIQUID

10 SAFETY REQUIREMENTS

11 DOCUMENT ITEMS DELIVERY AND REVIEWS

12 RESPONSIBILITY MATRIX

6.4 Mission Analysis

6.4.1 General

Mission analysis is conducted by the vehicle organization to confirm that the spacecraft is injected into the required orbit in the required condition. If a special orbit is specified, analyses for the mission plan and environment will be made according to the following steps, which are to be initiated almost at the same time as the spacecraft development. If the spacecraft mission requirements (mass, orbit, etc.) are similar to those of prescribed missions, the reference planning phase may be omitted.

(1) Reference planning phase

(From 32 months to 20 months before launch)

In the Reference Planning Phase, the major scenario of the mission plan is determined and identified problems are solved.

(2) Detailed planning phase

(From 18 months to 8 months before launch)

The Detailed Planning Phase determines and approves the mission plan; the vehicle is actually launched according to the mission plan completed in this phase. Therefore, the data required for launch operation, tracking control operation, and flight safety operation are generated in this phase.

(3) Final planning phase

(From 6 months to immediately before launch)

In the Final Planning Phase, the measured data for both the vehicle and the spacecraft are used to reconfirm that there is no problem in launching the vehicle according to the mission plan completed in the detailed planning phase.

6.4.2 Reference planning phase

In this phase, the major scenario of the flight plan is determined, and the data below are generated to solve identified problems and to meet the spacecraft development requirements.

Items b) through g) may be normally omitted except when the trajectory must be changed drastically from prescribed missions.

a) Mass property
b) Required trajectory
c) Preparation and estimation of reference trajectory

d) Orbit injection accuracy
e) Loads imposed on the spacecraft during flight
f) Position, velocity, and attitude at orbit injection
g) Sequence of events
h) History of the sun direction during flight
i) Spacecraft separation conditions
j) Thermal conditions imposed on the spacecraft during prelaunch and flight

6.4.3 Detailed planning phase

This phase solves problems identified in the reference planning phase and completes the mission plan. The following data are generated based on the mission plan:

a) Preparation and estimation of the detailed trajectory
b) Orbit injection accuracy
c) Load imposed on the spacecraft during flight
d) Position, velocity and attitude at orbit injection
e) Sequence of events
f) History of the sun direction during flight
g) Spacecraft separation conditions
h) Flight safety
i) Thermal conditions imposed on the spacecraft during prelaunch and flight

6.4.4 Final planning phase

In this phase, the following work is performed for the final confirmation of the mission plan:

a) Review of the mission plan (mission readiness review)
b) Final confirmation of the mission plan based on the measured data for both the spacecraft and the vehicle
c) Influence analysis using measured wind at launch site.

6.4.5 Spacecraft coupled loads analysis (CLA)

Spacecraft / H-IIA CLA is conducted to support the spacecraft design and verification with the spacecraft structural dynamic environments (refer to § 3.2) induced by the H-IIA launch. Two cycles of the CLA, preliminary (reference phase) CLA and final CLA, are conducted in normal procedure. Each CLA cycle is conducted at the spacecraft customer's request.

The preliminary CLA using preliminary spacecraft mathematical model can support spacecraft customer to assess compatibility of the H-IIA launcher.

The final CLA using a final spacecraft mathematical model shall verify the specified values of acceleration, loads, and deflections in the ICS.

In case that the payload design properties exceed the specified values in the H-IIA User's Manual, the extensive coordination based on the CLA can tailor interface requirements between the spacecraft and the H-IIA launch vehicle, and may enable the H-IIA to launch the protruded payload.

6.4.6 Integrated thermal analysis

A

Integrated thermal analysis is conducted to verify the spacecraft thermal environments during the H-IIA launching phase.

At the spacecraft customer's request, the integrated thermal analysis using a spacecraft mathematical model is conducted to verify the specified values of spacecraft thermal environments in the ICS. The integrated thermal analysis covers thermal environments from encapsulation to separation of the spacecraft.

6.5 Safety Reviews

6.5.1 Payload safety requirements

The design and launch operations of the spacecraft must satisfy requirements of "NASDA-STD-14B: Launch Vehicle Payload Safety Requirements". A

The spacecraft organization should enforce the safety program according to requirements of NASDA-STD-14B. A

The requirements on a safety program plan are described in § 6.5.1., § 6.5.2.

6.5.2 Requirements on the safety program plan of the spacecraft organization

The spacecraft organization must establish a safety program plan according to NASDA-STD-14B and submit to NASDA arranging its contents for the document "Payload Safety Program Plan". A

When the spacecraft organization establishes the safety program plan, it is recommended to coordinate with NASDA through an interface meeting.

6.5.3 Safety review data package

The spacecraft organization must submit the safety data package to NASDA according to NASDA-STD-14B in every operation phase. NASDA holds internal safety review of this safety data package as a basis for safety. | A

NASDA will inform the spacecraft organization of review results. Details on these contents are described in the document "CFX-97010: Outline of Payload System Safety Review for external Users".

6.5.4 Outline of NASDA safety review

NASDA system safety program plan is established for each spacecraft. Before establishment of this plan, NASDA will coordinate with the spacecraft organization at the interface meeting. Details are explained in CFX-97010.

According to NASDA system safety program plan, NASDA reviews the safety of the spacecraft based on the safety data package submitted by the spacecraft organization, informs the spacecraft organization of review results and requires any treatments if necessary.

6.5.5 Interface

The safety review meeting is set up based on coordination between the spacecraft organization and NASDA.

6.6 Reviews and Other Meetings

The following describes the reviews and other meetings required to launch the spacecraft and launch vehicle.

6.6.1 Reviews before launch operation

6.6.1.1 Mission readiness review (MRR)

This meeting is held once unless otherwise required. The review is carried out in conformity to the mission analysis report, and the results are confirmed to satisfy the requirements of the ICS. The user has the right to attend this meeting as an observer. This meeting is held about 3 months prior to the launch day.

6.6.1.2 Spacecraft interface confirmation review (SIC)

This meeting confirms that the spacecraft manufactures, tests and working program have conformed to the requirements of the ICS, and launch operations can be proceeded at the launch site.

This meeting is held just before spacecraft transportation to Tanegashima Space Center (TNSC).

6.6.2 Reviews for launch operation

6.6.2.1 Launch vehicle readiness review (LVRR)

This meeting is held in TNSC to verify that the launch vehicle and AGE and mission analysis have satisfied technical proceedings to be ready for the countdown operation. The verified results are to be reported in FRR.

The user has the right to attend this meeting as an observer.

This meeting is held toward Y-5.

6.6.2.2 Spacecraft readiness review (SCRR)

This meeting is held in TNSC to verify that the payload and GSE have satisfied technical proceeding to be ready for the countdown operation. The verified results are to be reported in FRR. This meeting is held just before encapsulation of the spacecraft into the payload fairing.

6.6.2.3 Flight readiness review (FRR)

This review confirms that all the organizations involved in the launch are ready for the countdown operations to be started. Each organization reports the current progress of preparation and conclusion of the review meeting. When reviews are finished, countdown operations are started after approval of the launch director is obtained.

The user must report the current preparation progress of the spacecraft and conclusion of the reviews.

This meeting is held toward Y-4.

6.6.3 Safety review

The safety review is explained in § 6.5.

6.6.4 Meetings

6.6.4.1 Interface meeting

This meeting coordinates matters to be described in the ICS and other interface requirements, and is held whenever required.

6.6.4.2 Daily meeting

This meeting confirms the progress of both the launch vehicle and the spacecraft organizations and facilities coordination. It is held every day (except on holidays). SIM and related members of the spacecraft organization have to attend this meeting.

6.6.4.3 Launch site readiness meeting

The purpose of this meeting, held just before the arrival of the spacecraft and associated equipment at TNSC, is to verify that the facilities are configured according to the requirements contained in the ICS.

6.6.4.4 Precountdown coordination meeting <tentative name>

Detailed schedules for the operations of Phase 3 (joint operations by the spacecraft and launch vehicle organizations), are coordinated between the spacecraft and launch vehicle organizations in this meeting. A

APPENDIX 1.

HISTORY OF NASDA LAUNCH VEHICLES

APPENDIX 1. HISTORY OF NASDA LAUNCH VEHICLES

A1.1 General

Japan has developed four kinds of large launch vehicles, N-I, N-II, H-I and H-II since early seventies when Japan started active space development under NASDA. Figure A1.1.1 shows configuration of these launch vehicles.

Table A1.1.1 summarizes launch results of NASDA launch vehicles.

Abstracts of the H-I and H-II launch vehicles are described in the following sections.

A1.2 Abstract of The H-I Launch Vehicle

NASDA has used the H-I launch vehicle as the main launch vehicle since 1986. It has a Delta-based first stage (the MB-3 engine), which is basically the same as the N-I and N-II rockets with the nine strap-on boosters (SOBs), and a domestically developed LOX / LH2 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. The launch site for the H-I is 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 is capable of sending a 4-ton class payload into geostationary transfer orbit.

The H-II is 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, is developed for the first stage. The LE-7 is a high performance engine adopting a high-pressure staged-combustion cycle.

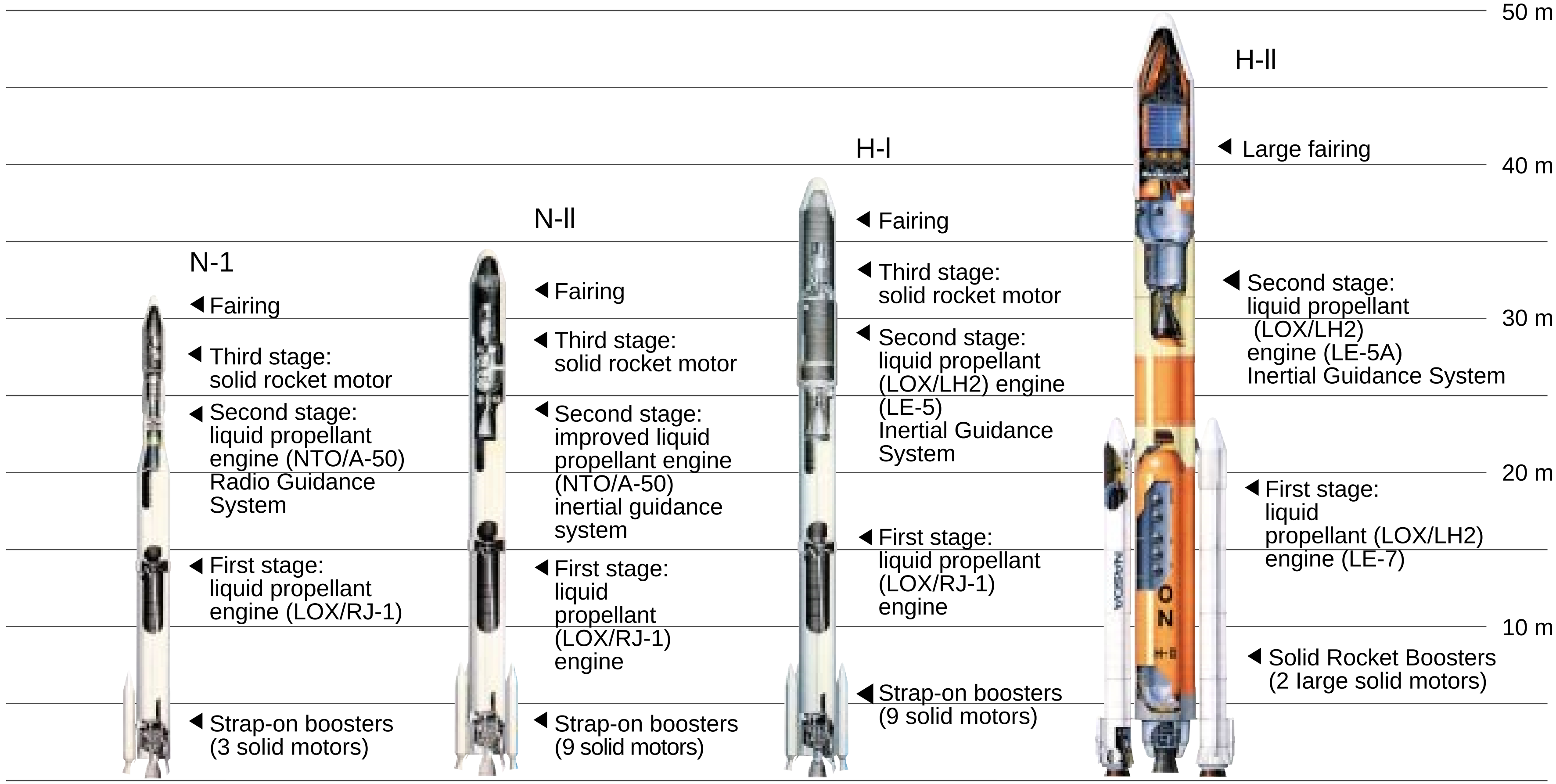
The LE-5A engine, which is the improved version of the LE-5 engine developed for the H-I, is used in the second stage. A strapped-down inertial guidance system utilizing ring laser gyros is employed for the guidance system. The standard payload fairing is 4 meters in diameter and 12 meters in length so that it can 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 is capable of launching planetary probes. The H-II is to be launched from Yoshinobu launch complex which has been newly constructed at the TNSC.

Table A1.1.1 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	6	1[*2]	86	2	1	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[*1]	2[*3]	71	3	2	4	0	1	0
Total	28	3	90	18	3	7	0	4	0

*1 : Include three dual launch missions.
*2 : The third stage collided with the spacecraft after separation.
*3 : In the restart phase, the second stage's engine was cutoff before scheduled plan (F#5).
: The first stage's main engine was cut off before scheduled plan (F#8).



	N-1	N-II	H-I	H-II	
Launched Satellites	7	8	9	5+(3*)	(*: Dual launch)
GTO Mass	260 kg	700 kg	1,100 kg	4,000 kg	
First Flight	1975	1981	1986	1994	

Figure A1.1.1 Configuration summary of NASDA launch vehicles

APPENDIX 2.

OUTLINE OF TANEGASHIMA ISLAND AND TANEGASHIMA SPACE CENTER

APPENDIX 2. OUTLINE OF TANEGASHIMA ISLAND AND TANEGASHIMA SPACE CENTER

A2.1 Tanegashima Island

A2.1.1 Location and topography

Tanegashima Island is located about 40 km southeast of Sata Point, Kagoshima Prefecture, the southernmost part of Kyushu Island, and about 115 km due south of Kagoshima City.

The entire island belongs to Kagoshima Prefecture and is composed of one city and two towns: Nishino-omote City, Nakatane-cho of Kumage District, and Minamitane-cho of Kumage District. The area is 445 km^2 and the population is about 37,000.

Topographically, Tanegashima is a long and narrow island lying from NNE to SSW. It is 58 km long and a maximum of 12 km wide, but only 6 km wide at the narrowest point near the center of the island. It is a flat island, which basically consists of coastal terraces, where the narrowest area is about 8 km long at the center and several stretches of hills of about 200 m above sea level run northeast and southeast of the narrowest area.

Rocky beaches and small dunes are also found along the coast. In the southern part, there are a number of cliffs steeply rising from the sands along the coastline, creating scenic beauty at Kumano, Takesaki, and Kadokura Point.

Yakushima Island has an area of about 505 km^2 and lies to the west of Tanegashima Island. This island is a large round lump of land, most of which is mountainous . The central part of the island consists of a series of mountains including Miyanoura-dake Mountain (1,935 m), the highest in the region of Kyusyu.

Figure A2.1.1 indicates the location of Tanegashima and Tanegashima Space Center.

A2.1.2 Climate

Tanegashima island has a yearly mean temperature of 19.4°C, a maximum temperature of 35.9°C, and a minimum temperature of 0.0°C. The rainfall is heavy, about 2,240 mm yearly mean rainfall. About 1000 mm of this is concentrated from June to September. The yearly mean humidity is 76%.

A2.1.3 Traffic

(1) Air route

Tanegashima Airport, a Class 3 airport (1,500 m x 30 m wireway), is located in Nakatane-cho. Regular flight services are available; five flights to and from Kagoshima Airport and one flight to and from Osaka Airport (by YS-11 or SAAB).

The flight time is 35 minutes between Kagoshima and Tanegashima Airports and 110 minutes between Osaka and Tanegashima Airports. It is about 3 hours from Tokyo to Tanegashima including transit time.

(2) Sea Route

There are three regular cargo/passenger ferry boats and two high-speed jet boats in service between Tanegashima and the Kagoshima Prefecture. These regular liners connect Nishino-omote Port and Kagoshima Port.

Tanegashima has another port at Shimama in Minamitane-cho, where chartered vessels disembark.

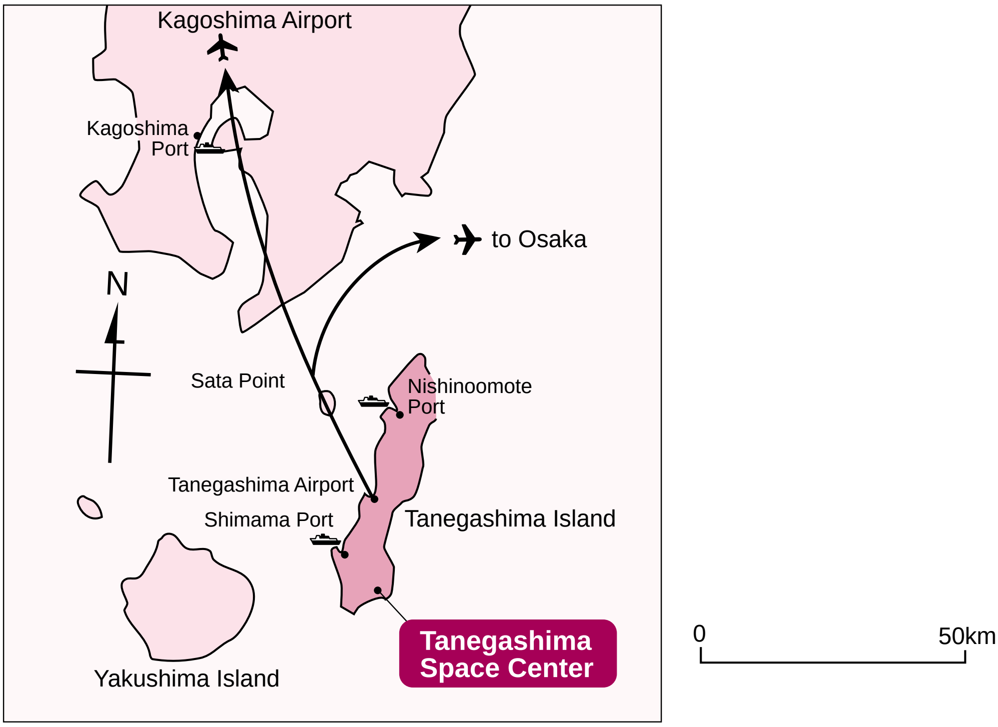
(3) Land transportation

All national roads, prefectural roads and city roads in Tanegashima are well prepared and maintained; all roads used for transporting spacecraft and GSE from Tanegashima Airport, Nishino-omote Port, and Shimama Port to Kaminaka are well paved. Regular bus lines are available from main points of the island including Tanegashima Airport and Nishino-omote Port to Kaminaka.

Taxis and rent-a-cars are also available.

Time required by automobile, such as by taxi, between two main points is as follows:

Nishino-omote Port	⇔	TNSC	: 80 minutes (47 km)
Nishino-omote Port	⇔	Kaminaka	: 70 minutes (43 km)
Tanegashima Airport	⇔	TNSC	: 40 minutes (26 km)
Tanegashima Airport	⇔	Kaminaka	: 30 minutes (17 km)
Kaminaka	⇔	TNSC	: 15 minutes (10 km)



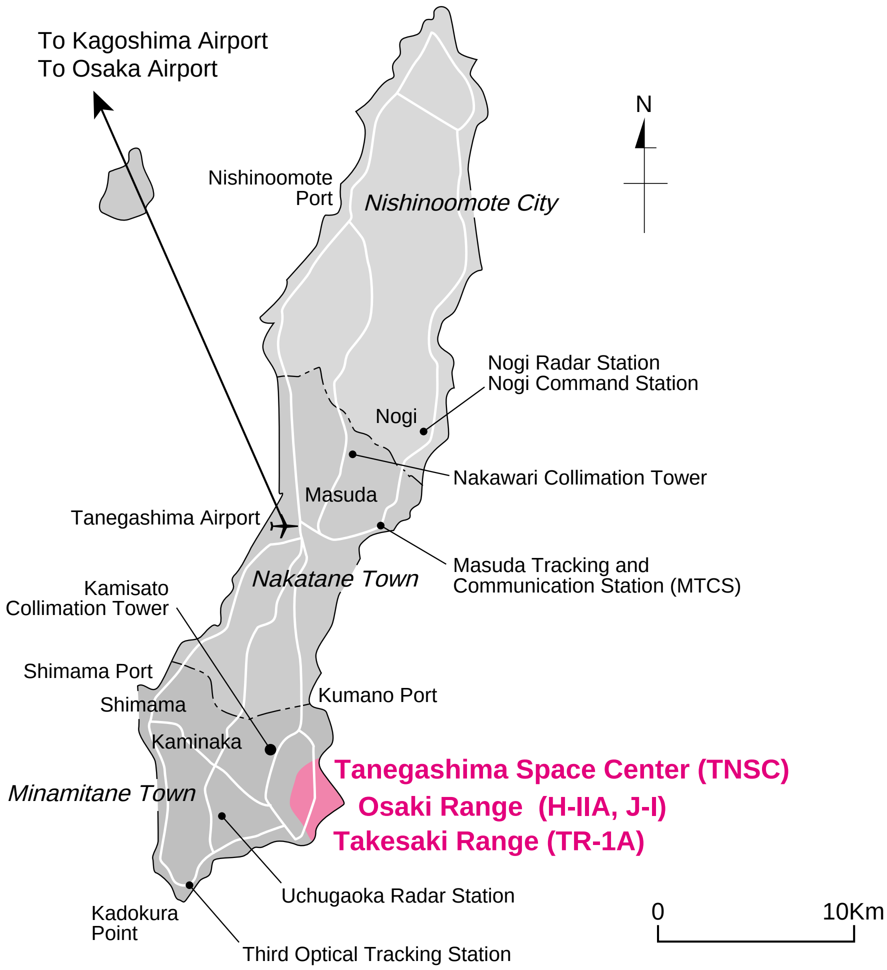


Figure A 2.1.1 Location of Tanegashima space center

A2.2 The Tanegashima Space Center

A2.2.1 General

Tanegashima Space Center (TNSC), located at the southeastern end of Tanegashima island, is the largest rocket range in Japan with a total site area of 8.6 million square meters. The site facilities include the Takesaki Range for small rockets, the Osaki Range for H-IIA and J-I launch vehicles, the Masuda Tracking and Communication Station (MTCS) 18 km north of Osaki, the Nogi Radar Station 6 km north of Masuda, the Uchugaoka Radar Station 6 km west of Osaki, and the optical tracking stations. Ground firing test facilities to verify liquid and solid propellant rockets and other necessary facilities are also included.

The major tasks of TNSC are prelaunch preparation operations including various checks, adjustments, and assembly of satellites and launch vehicles; countdown operation for launch; and tracking and control after launch. Thus, TNSC plays a major role in launching application satellites. It also plays a part in space development by conducting ground firing tests for solid booster motors and liquid rocket engines as well as conducting material tests for space applications using small rockets.

Figure 2.2.1 shows the layout of the major facilities in TNSC.

A2.2.2 Osaki Launch Range

The Osaki Launch Range consists of the Yoshinobu Launch Complex for launching the H-IIA launch vehicle for large spacecraft, spacecraft-related test facilities, and hazardous material related test and storage facilities. The launch pad of the Yoshinobu Launch Complex is located at 30° 23’ 50” north latitude and 130° 58’ 47” east longitude.

Figure A2.2.2 illustrates the layout of the major facilities in the Osaki Launch Range. Figure A2.2.3 shows an overview of the Yoshinobu Launch Complex. And Figure A2.2.4 shows a typical H-IIA launch operations process in the Osaki Launch Range. The following paragraphs describe the major facilities in the Osaki Launch Range.

A2.2.2.1 Yoshinobu Vehicle Assembly Building (VAB)

The VAB is a facility for assembling, inspecting, and adjusting all components of the launch vehicle (first stage, second stage, and SRB-A) transported from factories. Inside this VAB, the Movable Launcher (ML) is housed, and the launch vehicle is assembled on the ML and inspected. | A

Then the spacecraft / fairing composite is transported to the VAB and mated to the second stage. After that, final preparations of the spacecraft as well as the launch vehicle are conducted prior to the launch day maintaining environments for the spacecraft. Details are shown in A 2.6.

A2.2.2.2 Movable Launcher (ML)

The ML, on which the spacecraft/launch vehicle assembly is loaded, moves to the launch pad on the launch day.

The GSE of the spacecraft is set in the pressurized room inside the ML, and moved with the spacecraft. Details are shown in A2.7.

A2.2.2.3 Block House (B/H)

The B/H, which is established under the ground, coordinates and supervises the launching operation for the spacecraft and the launch vehicle by instructing, operating, and monitoring the launch vehicle assembly, adjustment, checkout, propellant loading and other launch activities at the Launch Complex and by remotely conducting the launch control operation including data transmission to the Range Control Center. Start of the countdown and the automatic launch sequence system is instructed from B/H. Launch control operations and data monitoring of the spacecraft are not conducted in B/H.

A2.2.2.4 No. 2 Spacecraft Test and Assembly Building (STA2)

Inside the No. 2 Spacecraft Test and Assembly Building, the transported large spacecraft is unpacked, visually inspected, and assembled.

The spacecraft then undergoes various tests and inspections including radio characteristic tests, functional tests, and compatibility tests. In addition, the overall spacecraft control is conducted from this building, including remote control over the checkout during countdown operations. Details are shown in A 2.3.

A2.2.2.5 Spacecraft and Fairing Assembly Building (SFA)

In this building, the high-pressure leak test of the propellant system is performed, the propellant is loaded and pressurized, and the pyrotechnics and the solid propellant motor are attached. Finally, the spacecraft is installed in the fairing. Details are shown in A2.4.

A2.2.2.6 No. 1 Spacecraft Test and Assembly Building (STA1)

The STA1 is mainly for testing and assembling small or medium size spacecraft in case of a dual launch. The transported spacecraft is unpacked, visually inspected, and assembled here. It then undergoes various tests and inspections including radio characteristic test, functional test, and compatibility test. Checkout of the spacecraft during launching is conducted in the STA2 (not STA1).

A2.2.2.7 Third stage and Spacecraft Assembly Building (TSA)

The spacecraft assembled and tested using the STA1 is transferred to the TSA, where the high-pressure leak test of the propellant system is performed, the propellant is loaded and pressurized, and the pyrotechnics and the solid propellant motor are attached. The spacecraft is installed in the fairing in the SFA. Details are shown in A2.5. | C

A2.2.2.8 Non-Destructive Test Facility (NDTF)

A large X-ray CT system is installed inside the NDTF, for the X-ray inspection of the spacecraft solid propellant motor.

A2.2.2.9 Nakanoyama Telemetry Command Station

The Nakanoyama Telemetry Command Station monitors and records in-flight launch vehicle conditions, receiving telemetry data on acceleration, pressure, temperature and so on, which are transmitted from each stage after lift-off. The Station then relays this data to the Takesaki Range Control Center for flight safety operation. It is also equipped with a flight safety command transmitter. | A

A2.2.2.10 Other facilities

(1) First solid propellant storage, second solid propellant storage and pyrotechnics storage

Pyrotechnics for the launch vehicle, spacecraft system, and solid propellant motor are stored in these facilities.

(2) Third solid propellant storage, fourth solid propellant storage

The solid rocket motor is stored in these facilities.

(3) Solid Booster Test Building (SBB)

Various tests on the solid rocket motor and pyrotechnics are performed here.

(4) Osaki Supporting Launch Building (SLB)

Conducts analysis of gases and propellants, and flushing of cleanliness control components.

(5) Yoshinobu Launch Building (LB)

Operates the air-conditioning system of the launch vehicle including the fairing at the LP.

(6) Osaki 80 m meteorological observation tower

Observes the weather in Osaki Launch Range.

(7) Osaki power station

Generates power mainly for Osaki area in TNSC. This station switches between private and commercial power supply, monitors the power supply system, and monitors the air-conditioning systems in major buildings.

(8) Osaki Office 1

An office for launch vehicle operators.

(9) Osaki Office 2

An office for launch vehicle operators.

(10) Osaki aerospace ground equipment (AGE) storage

Tools, consumables and instruments used in TNSC are stored here and lent.

(11) Yoshinobu combustion test facility

Ground combustion test of the first-stage engine (LE-7A) for H-IIA is performed.

(12) General accident prevention control office

Responsible for accident prevention control in TNSC.

A2.2.3 Takesaki Area

Relevant control facilities are located in the Takesaki area. The following are the main facilities.

A2.2.3.1 Administration Building (AB)

General office building for TNSC workers; contains a reception area staffed by the Administration Department.

A2.2.3.2 Takesaki Range Control Center (RCC)

The Takesaki RCC is responsible for information collection, analysis, instruction, coordination, and monitoring throughout the entire launch operations of the launch vehicles and spacecraft in Tanegashima, including prelaunch operations, ground safety operation, launching, and tracking, to ensure smooth progress in these operations. Details are shown in A 2.8.

A2.2.3.3 Takesaki Observation Stand

This stand is for press and broadcasting personnel and is equipped with an interview room for the reporters; waiting rooms for reportorial staff guests, and launch operation crew; TV-monitoring room; time indicating equipment; dark rooms for developing film; etc.

A2.2.3.4 Takesaki Space Exhibition Hall

To further understanding of space development, interrelations between the space and mankind, the origination of space development, contribution of space development to mankind, the structure and functions of satellites and launch vehicles, exhibits for launching, tracking and controlling are exhibited and open to the public.

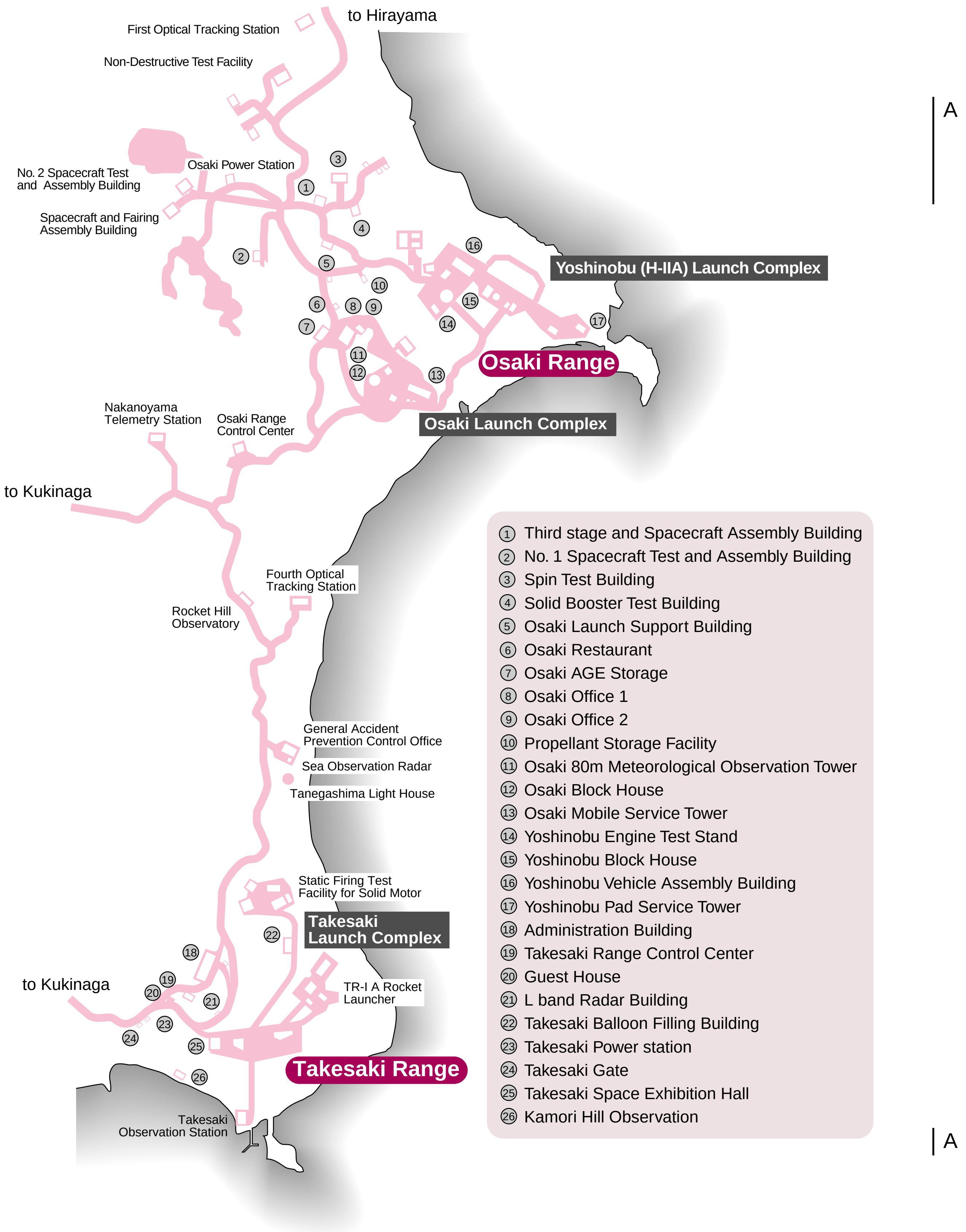
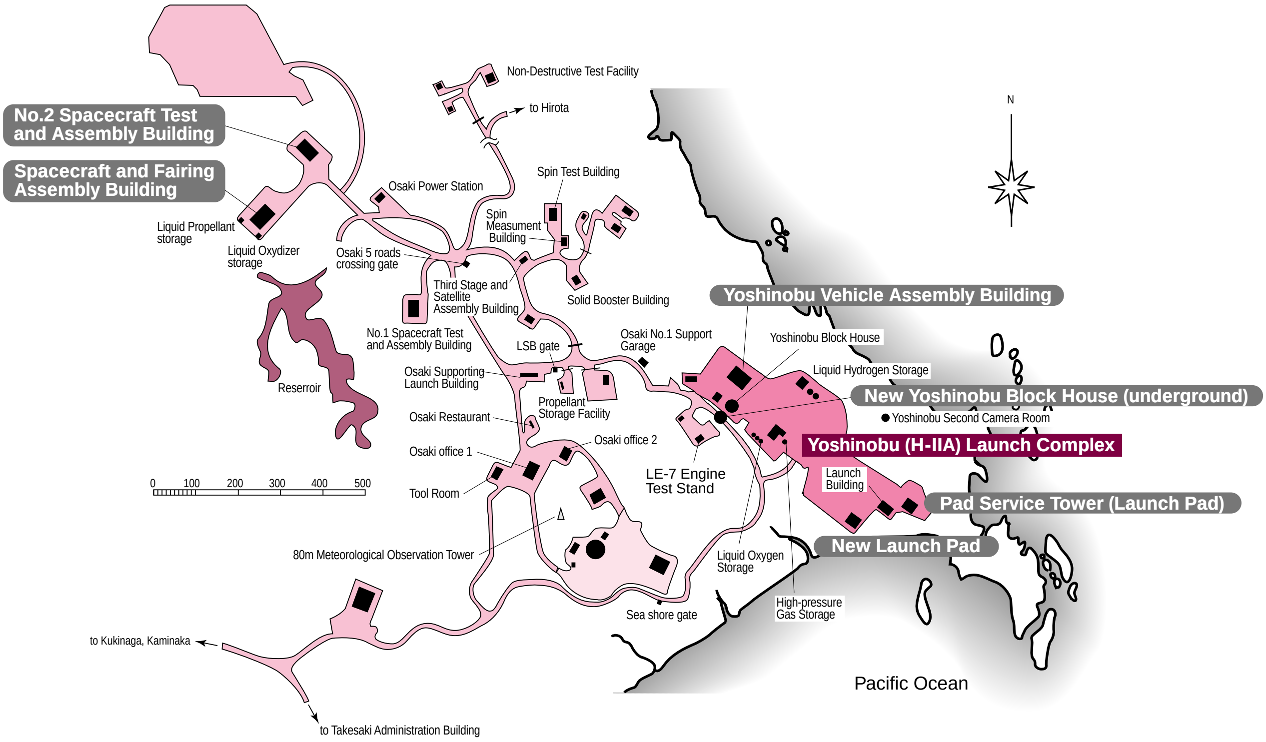


Figure A2.2.1 Location of Facilities of Tanegashima Space Center

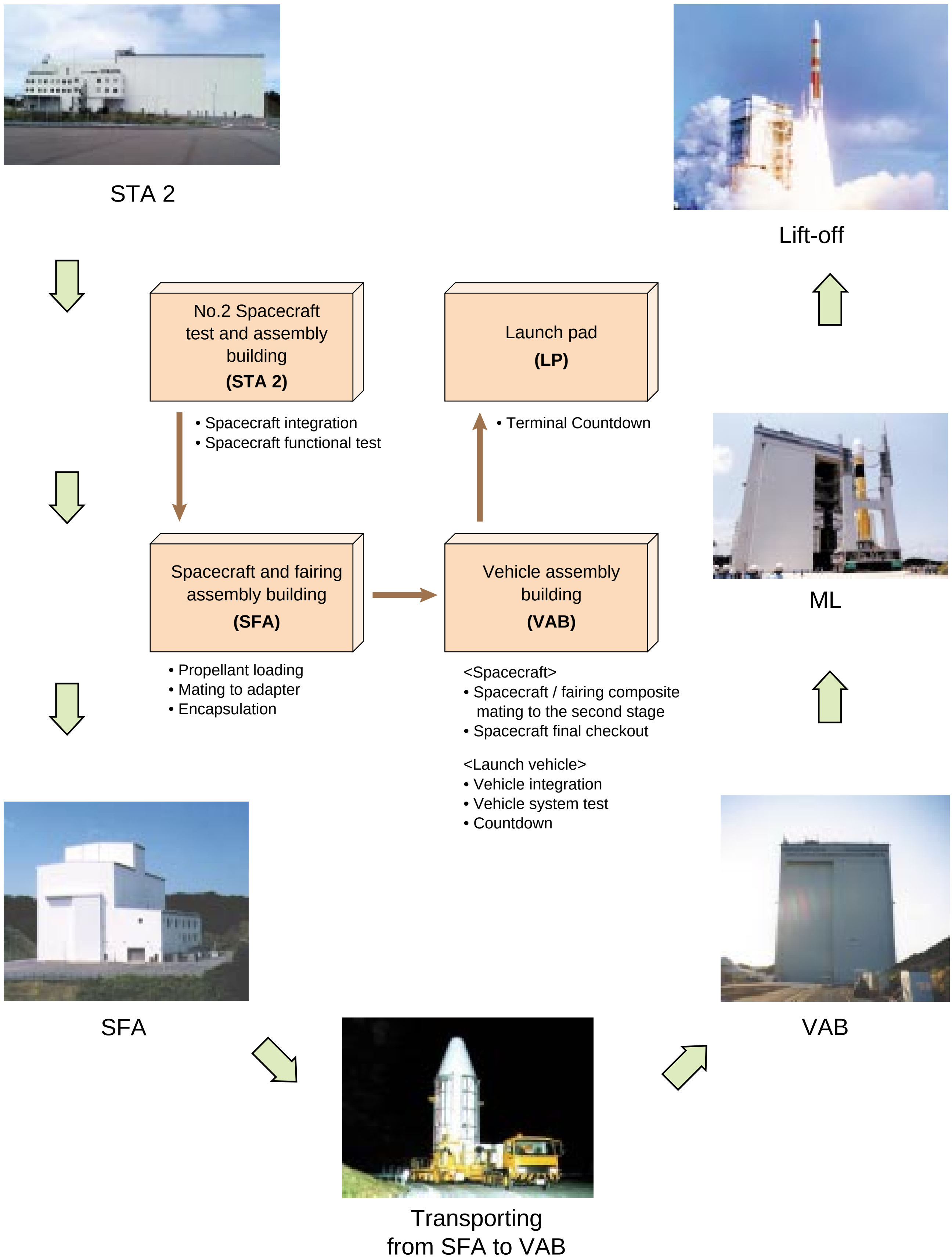


A

A

Figure A2.2.2 Location of major facilities in Osaki Launch Range

Figure A2.2.3 Overview of New Yoshinobu Launch Complex | C



B

B

A

Figure A2.2.4 H-IIA launch operations process

A

A2.3 No. 2 Spacecraft Test and Assembly Building (STA2)

A2.3.1 General

The STA2 is used for a series of operations for a large spacecraft to be loaded into the H-IIA launch vehicle after delivery to the launch site. Operations include installation and adjustment of ground support equipment (GSE) validation, preparation for assembling the spacecraft after delivery, alignment measurement after assembly, functional tests, RF link test with Masuda Tracking and Communication Station, charging and discharging of batteries, weight measurement, and preparation for transportation into SFA.

This building has two air lock rooms, two spacecraft preparation halls, two GSE storage rooms and three checkout rooms so that checkouts of two (or three in maximum) spacecraft are conducted individually. A

During Phase 3 operations, the control and the checkout of the spacecraft, which is loaded onto the launch vehicle, are conducted by remote control from this building.

Specifications of the first floor section are as follows.

a) Cleanliness : class 100,000

b) Temperature : 22 ± 3 °C

c) Humidity : 50 ± 10 % RH

e) Power supply : Frequency_60 Hz

Panel capacity_Table A2.3.1

Location of outlets - shown in H-IIA Payload-Related Facilities and GSE Manual A

A2.3.2 Main Structure and Functions of STA2

The STA2 is a four-storied, steel-frame construction building of about 8,580 m^2 floor space. (Total area of first floor is 4,550 m^2)

The first floor section contains mainly two air lock rooms, two spacecraft preparation halls, two GSE storage rooms, three checkout rooms, control room, unpacking room, air conditioning equipment room 1, two cloak room, rest room, entrance, reception, stairs (first to fourth floor), outdoor ladder (first floor to the roof), and two high-pressure gas storages. A

The second floor section contains five office rooms, two meeting rooms, two storage rooms, kitchenette, rest room, tour path for visitors and so on. A

The third floor section contains eight office rooms, two meeting rooms, two storage rooms, kitchenette, rest room, warehouse and two electric power rooms.

The fourth floor section contains air-conditioning room (#2).

Figure A2.3.1 shows the STA2.

Figure A2.3.1 No. 2 Spacecraft Test and Assembly Building (STA2)

Table A2.3.1 Summary of power supply (STA2) (1/2)

A

Install place	Name	Power				Quantity	Remarks
Checkout room 1	P-2 distribution board	AC 210 V	3ϕ	4 W	100 A	1	Uninterruptable *CVCF system
		AC 210 V	3ϕ	3 W	50 A	1	
		AC 100 V	1ϕ	2 W	50 A *	8	
		AC 115 V	1ϕ	2 W	50 A *	2	
		AC 115 V	1ϕ	2 W	20 A *	4	
		AC 210 V	1ϕ	2 W	50 A	5	
	P-2-1 distribution board	AC 480 V	3ϕ	4 W	100 A	1	
		AC 210 V	3ϕ	4 W	100 A *	1	
		AC 210 V	3ϕ	4 W	50 A *	2	
		AC 115 V	1ϕ	2 W	30 A *	3	
		AC 115 V	1ϕ	2 W	20 A *	3	
	Wall outlet	AC 100 V	1ϕ	2 W	20 A x 2	9	
	Grounding	First class grounding terminal Terminal box				4	
Checkout room 2	P-5 distribution board	AC 210 V	3ϕ	4 W	50 A	2	Uninterruptable *CVCF system
		AC 210 V	3ϕ	4 W	100 A	2	
		AC 100 V	3ϕ	3 W	100 A	5	
		AC 100 V	3ϕ	3 W	50 A	2	
		AC 100 V	3ϕ	3 W	20 A	7	
		AC 480 V	3ϕ	4 W	75 A	1	
		AC 100 V	1ϕ	2 W	50 A *	8	
		AC 115 V	1ϕ	2 W	50 A *	5	
		AC 115 V	1ϕ	2 W	20 A *	9	
	Wall outlet	AC 115 V	1ϕ	2 W	20 A x 2	6	
		AC 100 V	1ϕ	2 W	20 A x 2	7	
	Grounding	First class grounding terminal Terminal box				3	
Checkout room 3	P-6 distribution board	AC 210 V	3ϕ	4 W	100 A	2	Uninterruptable *CVCF system
		AC 210 V	3ϕ	4 W	50 A	2	
		AC 100 V	1ϕ	3 W	100 A	5	
		AC 480 V	3ϕ	4 W	75 A	1	
		AC 100 V	1ϕ	2 W	50 A *	8	
		AC 115 V	1ϕ	2 W	50 A *	6	
		AC 115 V	1ϕ	2 W	20 A *	8	
	Wall outlet	AC 115 V	1ϕ	2 W	20 A x 2	5	
		AC 100 V	1ϕ	2 W	20 A x 2	6	
	Grounding	First class grounding terminal Terminal box				3	

Table A2.3.1 Summary of power supply (STA2) (2/2)

A

Install place	Name	Power				Quantity	Remarks
Spacecraft Preparation Hall 1	P-1 distribution board	AC 210 V	3ϕ	4 W	100 A	1	
		AC 210 V	3ϕ	3 W	100 A	1	
		AC 210 V	3ϕ	3 W	50 A	1	
		AC 210 V	1ϕ	3 W	50 A	2	
		AC 210 V	1ϕ	3 W	20 A	4	
		AC 100 V	1ϕ	3 W	100 A	2	
		AC 100 V	1ϕ	3 W	50 A	9	
	P-1-1 distribution board	AC 210 V	3ϕ	4 W	60 A *	2	* For refrigerator
		AC 115 V	3ϕ	4 W	20 A	6	
		AC 100 V	1ϕ	3 W	50 A	2	
	Wall outlet	AC 210 V	3ϕ	3 W	30 A	4	
		AC 115 V	1ϕ	2 W	20 A x 2	4	
		AC 100 V	1ϕ	2 W	20 A x 2	4	
	Grounding	First class grounding terminal Terminal box				4	
Spacecraft Preparation Hall 2	P-7 distribution board	AC 210 V	3ϕ	4 W	100 A	1	
		AC 210 V	1ϕ	3 W	50 A	2	
		AC 115 V	3ϕ	4 W	50 A	2	
		AC 115 V	3ϕ	4 W	20 A	8	
		AC 100 V	1ϕ	3 W	50 A	7	
		AC 100 V	1ϕ	3 W	20 A	1	
		AC 100 V	1ϕ	3 W	150 A	1	
		AC 100 V	1ϕ	3 W	100 A	2	
	Wall outlet	AC 210 V	3ϕ	3 W	30 A	5	
		AC 115 V	1ϕ	2 W	20 A x 2	7	
		AC 100 V	1ϕ	2 W	20 A x 2	8	
	Grounding	First class grounding terminal Terminal box				4	

A2.4

Spacecraft and Fairing Assembly Building (SFA)

A2.4.1

General

The SFA is used for performing some Phase 2 and Phase 3 operations for a large spacecraft to be loaded into the H-IIA launch vehicle, following Phase 1 operations.

Phase 2 and Phase 3 operations performed in the SFA include: unpacking and visual inspection of the spacecraft after transportation into the SFA; transportation of the GSE used in the SFA and its installation and adjustment; high-pressure leak tests of spacecraft propellant systems; preparation of filling equipment; filling and pressurization of propellant in the spacecraft; storage of filling equipment; installation of pyrotechnics in the spacecraft; final fitting out of solid propellant motor and its installation on the spacecraft; mating of the spacecraft and payload adapter, the spacecraft and fairing; and preparation for transporting the spacecraft and fairing composite.

In addition to the above spacecraft-related operations, preparation and inspection of the fairing carried into this building are also performed.

The walls between the spacecraft-fairing assembly hall, filling and assembly hall, and checkout room are explosion-proof. In addition, the spacecraft fairing assembly hall, filling and assembly hall, and filling equipment room are equipped with emergency exhaust equipment, propellant washing and draining system, gas sensors, fire alarms, body shower and eye washer, and emergency exits for coping with accidents in handling the propellant. | A

Specifications of the first floor are as follows.

a) Cleanliness : class 100,000

b) Temperature : normal operation and after encapsulation 21 ± 3 °C
propellant loading 21 ± 1 °C
battery charging 18 ± 3 °C

c) Humidity : 45 ± 5 % RH | A

e) Power supply : Frequency 60 Hz
Panel capacity - Table A2.4.1
Location of outlets - shown in H-IIA Payload-Related Facilities and GSE Manual | A

A2.4.2

Main Structure and Functions of SFA

The SFA is a three-storied steel-frame building of 4,010 m^2 floor space.

The first floor contains air locks (1 and 2), spacecraft - fairing assembly hall, filling and assembly hall, filling equipment room, GSE storage room, checkout room, preparation room (#1 and #2), equipment storage room, pump room, | A

entrance hall, rest rooms, showers, visitor's path (entrance), high-pressure gas storage, and the open shed at the outside.

The second floor contains air-conditioning room #1 and the transformation room. The third floor section contains the visitor's path and air-conditioning room #2.

Figure A2.4.1 shows the spacecraft & fairing assembly building.

Figure A.2.4.1 Spacecraft and Fairing Assembly Building (SFA)

Table A2.4.1 Summary of power supply (SFA)

A

Installed Place	Name	Power				Quantity	Remarks
Checkout room	P-1 distribution board	AC200V	3φ	3 W	50 A	2	Uninterruptable *CVCF system
		AC100V	1φ	2 W	50 A*	1	
		AC100V	1φ	2 W	20 A*	4	
		AC115V	1φ	2 W	50 A*	2	
		AC115V	1φ	2 W	20 A*	4	
	P-2 distribution board	AC200V	3φ	3 W	50 A	2	Uninterruptable *CVCF system
		AC100V	1φ	2 W	50 A*	1	
		AC100V	1φ	2 W	20 A*	4	
		AC115V	1φ	2 W	50 A*	2	
		AC115V	1φ	2 W	20 A*	4	
		AC200V	3φ	3W	100 A	1	
	Wall outlet	AC100V	1φ	2W	20A x 2	8	
	Grounding	First class grounding terminal Terminal box				3	
Spacecraft-fairing assembly hall	Wall outlet (Explosion-proof socket)	AC200V	3φ	4 W	30 A	4	
		AC100V	1φ	3 W	20 A	14	
		AC115V	1φ	3 W	20 A	3	
	Grounding	First class grounding terminal Terminal box				12	

A2.5 Third Stage and Spacecraft Assembly Building (TSA)

A2.5.1 General

The TSA is used for a series of operations for a small spacecraft and dual launch spacecraft to be loaded into the H-IIA launch vehicle after delivery to the launch site. Operations include filling and pressurization of the spacecraft propellant, pyrotechnics installation, solid propellant motor installation.

Specifications of the first floor section are as follows.

a) Cleanliness : class 100,000
b) Temperature : 21 ~ 25 °C
c) Humidity : 40 ~ 60 % RH
e) Power supply : Frequency 60 Hz
Panel capacity - Table A2.5.1
Location of outlets - shown in H-IIA Payload-Related Facilities and GSE Manual (in coordinating NASDA-HDBK-T.B.D.)

A2.5.2 Main Structure and Functions of TSA

The TSA is a two storied reinforced concrete building. RF test antenna (directed to STA1) is provided on the rooftop.

It consists of an airlock room, checkout room, spacecraft fairing assembly hall, two cloak room, rest room, two air shower room, comunications equipment room, outdoor high-pressure gas cylinder storage.

Figure A2.5.1 shows the TSA.

A

Figure A.2.5.1 Third Stage and Spacecraft Assembly Building (TSA)

Table A2.5.1 Summary of power supply (TSA)

A

Install place	Name	Power				Quantity	Remarks
spacecraft - fairing assembly hall	Wall outlet	AC 200 V	3φ	4 W	30 A	7	Explosionproof socket
		AC 100 V	1φ	3 W	30 A	4	
	Graounding	First class grounding terminal Terminal box				3	
Checkout room	L-2 distribution boad	AC 200 V	3φ	3 W	30 A	1	* Can switch to 200 V
		AC 100 V*	1φ	3 W	20 A	4	
		AC 200 V	1φ	2 W	20 A	1	
	Wall outlet	AC 100 V	1φ	2 W	15 A x 2	9	
		AC 200 V	3φ	3 W	30 A	1	

A2.6 Yoshinobu Vehicle Assembly Building (VAB)

A2.6.1 General

The VAB is a facility for assembling, inspecting and adjusting all components of the H-IIA vehicle (the first stage, the second stage, the SRB-A) transported from each factory. And the mating of the spacecraft/fairing composite onto the launch vehicle and final checkout of the spacecraft and the launch vehicle are also conducted in this building prior to the launch day.

The spacecraft / fairing composite is carried into the VAB through the spacecraft carrying door loaded on the transportation dolly, and hoisted on the top of the second stage using the crane equipped in the VAB.

To maintain environments inside the VAB, the air conditioning equipment is fully furnished. Especially, upper stories of the VAB, in which spacecraft operations are performed, are maintained in a cleanliness of the class 100,000. Environments inside the fairing are controlled at 10 – 25 ± 2 °C by supplying the air through the air conditioning duct. Cleanliness of this air is class 5,000.

The GSE of the spacecraft is set in the pressurized room inside the ML, and control and monitor of the spacecraft is conducted by remote control from the STA2.

On the launch day, the slide door is opened, and then the ML on which the spacecraft / vehicle is loaded is transferred to the LP. During transfer, conditioned air is supplied to the spacecraft.

Figure A2.6.1 and Figure A2.6.2 show a general view and a sectional view of the VAB respectively.

The VAB has two pressurization rooms respectively at 9th, 10th and 11th floor. Specifications of power supply for spacecraft operations in pressurization rooms at 9th and 10th floors are as follows.

a) Frequency_60 Hz

b) Panel capacity_Table A2.6.1

Location of outlets - shown in H-IIA Payload-Related Facilities and GSE Manual (in coordinating NASDA-HDBK-T.B.D.)

A2.6.2 Main Structure and Function of VAB

The elevator type floor is equipped in upper stories to access the spacecraft; it is adjustable to required heights.

Each level is equipped with AGE storage space, air-conditioning equipment and so on. And for safety and accident prevention during operations, temperature sensors, smoke sensors, fire alarms, communications devices, announcement systems, face / body showers, and evacuation escape routes are provided at required places on each floor.

B

Figure A2.6.1 Yoshinobu Vehicle Assembly Building (VAB)

A

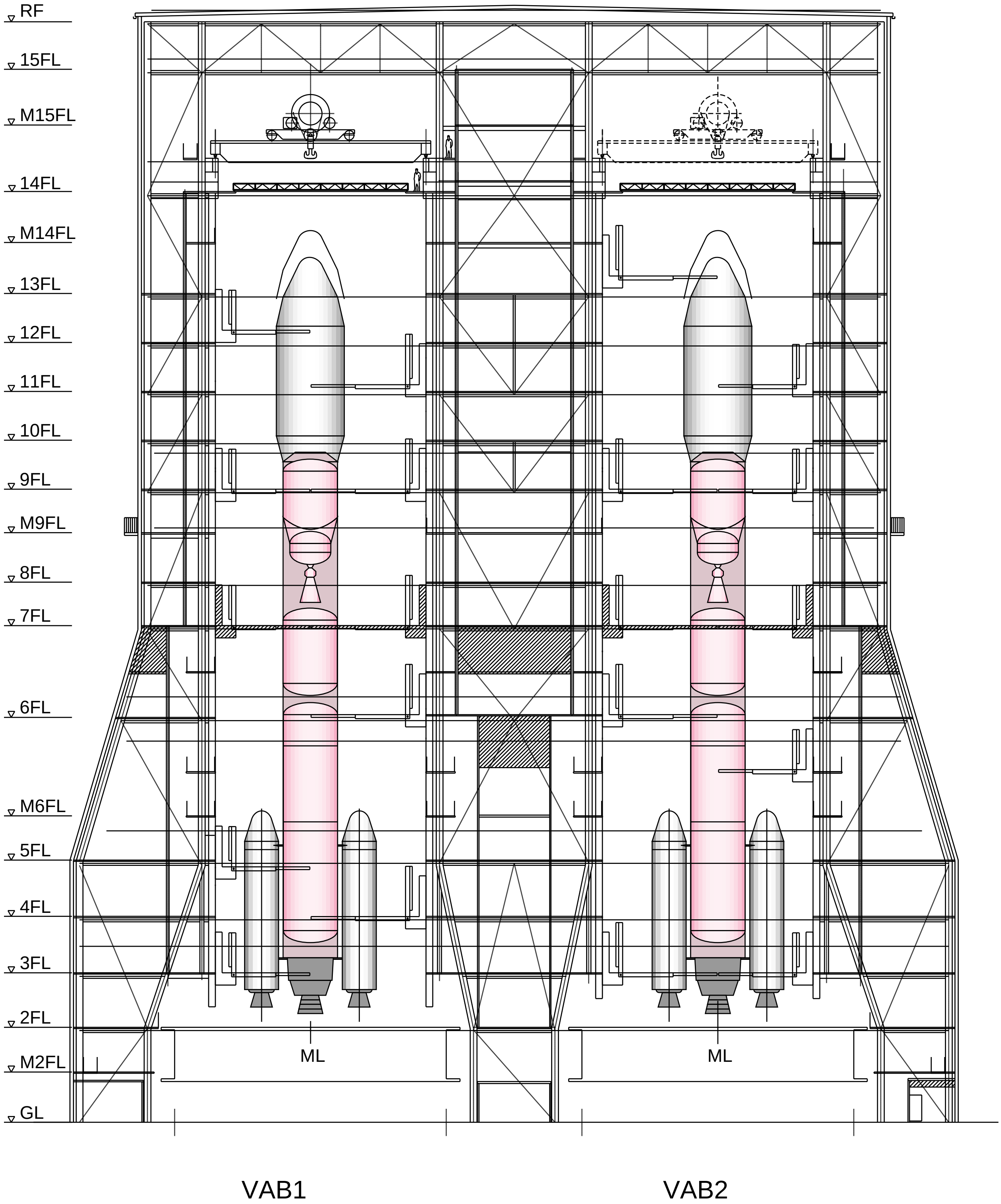


Figure A2.6.2 Yoshinobu Vehicle Assembly Building (VAB, sectional view)

Table A2.6.1 Summary of power supply (VAB 9th and 10th floor) A

A

Install place	Name	Power				Quantity	Remarks
Payload Checkout Room 1	Spacecraft AGE Power supply equipment (VAB1)	AC210V	60 A	3ϕ	4 W	1	Uninterruptable *CVCF system
		AC220V	30 A	1ϕ	2 W	1	
		AC220V	20 A	1ϕ	2 W	1	
		AC210V	30 A	1ϕ	2 W	1	
		AC115V	20 A	1ϕ	2 W	2	
		AC110V	20 A	1ϕ	2 W	2	
Payload Checkout Room 2	Spacecraft AGE Power supply equipment (VAB2)	AC210V	60 A	3ϕ	4 W	1	Uninterruptable *CVCF system
		AC220V	30 A	1ϕ	2 W	1	
		AC220V	20 A	1ϕ	2 W	1	
		AC210V	30 A	1ϕ	2 W	1	
		AC115V	20 A	1ϕ	2 W	2	
		AC110V	20 A	1ϕ	2 W	2	

A2.7 Movable Launcher (ML) | A

A2.7.1 General | A

The ML is a movable facility, on which the spacecraft and the launch vehicle are loaded in the VAB, and which carries them to the launch pad on the launch day and functions as the launch table at the launch pad as it is.

The GSE of the spacecraft is set in the pressurized room inside the ML and has the interface with the spacecraft through the umbilical line equipped in the umbilical mast of the ML.

Figure A2.7.1 shows a general view of the ML. | A

A2.7.2 Main structure and functions of ML | A

(1) Launch table

The H-IIA launch vehicle is set on the launch settling table of the launch table. The first and second stages of the launch vehicle are loaded with propellants though the launch settling table and the umbilical mast respectively.

For the spacecraft inside the payload fairing, electrical signals and power are sent through the umbilical cable and conditioned air is supplied through the air conditioning duct. The umbilical cable and the air conditioning duct are equipped in the umbilical mast on the launch table.

(2) Dolly (Movable Launcher Transporter) | A

The dolly is equipment which carries the launch table from the VAB to the launch pad. The dolly has wheels and can move anywhere and rotate. | C

(3) Pressurization room

The pressurization room is inside the ML and the GSE of the spacecraft is set in this room. Specifications of this room are as follows.

a) Area : 20 m^2 (per each spacecraft)
b) Temperature : 18 - 28 °C
c) Humidity : 40 - 60 % RH
d) Power supply : Frequency_50 Hz or 60 Hz | C
Panel capacity_Table A2.7.1 | A
Location of outlets - shown in H-IIA Payload-Related Facilities and GSE Manual (in coordinating NASDA-HDBK-T.B.D.) | A

B

Figure A2.7.1 Movable Launcher (ML)

A

Table A2.7.1 Summary of power supply (ML) A

Place	Power					Quantity	Remarks
ML Payload GSE Room 1	AC100V	50 A	1ϕ	2 W	60 Hz	1	
	AC100V	20 A	1ϕ	2 W	60 Hz	2	
	AC115V	50 A	1ϕ	2 W	60 Hz	1	
	AC115V	20 A	1ϕ	2 W	60 Hz	2	
	AC210V	50 A	1ϕ	2 W	60 Hz	2	
	AC208V	50 A	3ϕ	4 W	60 Hz	1	
	AC208V	50 A	3ϕ	4 W	60 Hz	1	
	AC220V	30 A	1ϕ	2 W	60 Hz	2	
	AC220V	50 A	1ϕ	2 W	60 Hz	1	
	AC220V	10 A	1ϕ	2 W	60 Hz	1	
	AC220V	30 A	1ϕ	2 W	50 Hz	2	
ML Payload GSE Room 2	AC100V	50 A	1ϕ	2 W	60 Hz	1	
	AC100V	20 A	1ϕ	2 W	60 Hz	2	
	AC115V	50 A	1ϕ	2 W	60 Hz	1	
	AC115V	20 A	1ϕ	2 W	60 Hz	2	
	AC210V	50 A	1ϕ	2 W	60 Hz	2	
	AC208V	50 A	3ϕ	4 W	60 Hz	1	
	AC208V	50 A	3ϕ	4 W	60 Hz	1	
	AC220V	30 A	1ϕ	2 W	60 Hz	2	
	AC220V	50 A	1ϕ	2 W	60 Hz	1	
	AC220V	10 A	1ϕ	2 W	60 Hz	1	
	AC220V	30 A	1ϕ	2 W	50 Hz	2	
	AC100V	20 A	1ϕ	2 W	60 Hz	1	For Piggy-buck spacecraft

A

A2.8

Takesaki Range Control Center (RCC)

A

A2.8.1

General

A

The Takesaki RCC is responsible for control and planning of overall launch operations, communication with each launch site station, ground safety, flight safety, and weather observation.

For these operations, the center is equipped with a range control system, flight safety processing system, flight safety control system, communication system (clock system, operational intercommunication telephone system (OIS), and optical transmission system), and weather observation system.

In addition, the third floor is equipped with optical equipment (No. 2 optical observatory station) for tracking the rocket after lift-off.

A2.8.2

Main structure and functions of Takesaki RCC

A

Takesaki RCC is housed in a three-story reinforced concrete building. Its floor area is about 1,400 m^2.

The first floor of the building comprises the control room, communication and computer room, flight safety command room, planning and combination room, meeting room, visitors observation room, and office rooms. A

The second floor contains the weather observation machine room, weather observation operations room, and office rooms.

The third floor contains the optical observation room and observation camera room.

Figure A2.8.1 shows the Takesaki RCC. A

Figure A2.8.1 Takesaki Range Control Center (RCC) A

A2.9 Down range stations

A2.9.1 Ogasawara down range station

The Ogasawara down range station is located on Chichijima Island of Ogasawara village, Tokyo. The launch vehicle telemetry receiver facility, flight safety command transmitter facility, and tracking radar facility are installed there.

Figure A2.9.1 shows the Ogasawara down range station.

(1) Objective of Ogasawara down range station

The major objective of the Ogasawara down range station is to take over the tracking of the vehicle launched at Tanegashima outside the visible range of Tanegashima. For the H-IIA, it monitors the flight conditions during the period from combustion in the second stage to orbit injection using the vehicle telemetry and confirms the vehicle flight position by the tracking radar.

(2) Interface with Tanegashima

The vehicle orbit prediction information (elevation angle, azimuth and line-of-sight distance) for the antenna and radar tracking system required to capture the vehicle is calculated in real time by the flight safety processing system at the Tanegashima Range Control Center Building, based on the vehicle tracking measurement data obtained at the Tanegashima Launch Site, then supplied to the Ogasawara down range station. The collected radar tracking data and specific vehicle telemetry data are transmitted to Tanegashima in real time. The data transmission interface between Tanegashima and Ogasawara is ensured by using telecommunication satellites and domestic leased lines.

A2.9.2 Christmas down range station

The Christmas down range station is located at 157 deg. W. longitude and 2 deg. N. latitude on Christmas Island of the Republic of Kiribati about 1,000 km south of Hawaii. The launch vehicle telemetry receiver facility, INMARSAT earth station facilities and power facilities are installed there.

Figure A2.9.2 shows the Christmas down range station.

(1) Objective

The major objective of the Christmas down range station is to monitor reignition of the vehicle second stage engine (SEIG #2) and to confirm separation between the vehicle and spacecraft using telemetry. The received data are recorded and sent to Tanegashima via the INMARSAT line after tracking.

(2) Initial capturing of launch vehicle

The vehicle orbit prediction information (elevation angle and azimuth) for the antenna required to capture the vehicle is supplied from Tanegashima in real time and is based on the measurements of the vehicle from the Ogasawara down range station.

Figure A2.9.1 Ogasawara down range station

| A

Figure A2.9.2 Christmas down range station

| A

APPENDIX 3.

PAYLOAD ADAPTER

APPENDIX 3. PAYLOAD ADAPTER

11 types of payload adapter are offered. C

These are;

a) Model 937M
b) Model 937MH
c) Model 937M-SPIN
d) Model 937M-SPIN-A
e) Model 1194M
f) Model 1666M
g) Model 1666S
h) Model 2360S
i) Model 3470S
j) Model 1666MA C
k) Model 239M

In the following sections, more information on these adapters is provided.

APPENDIX 3.1 937M ADAPTER

A

The main characteristics are as follows. C

(1) Interface diameter : 945 mm
(2) Height : 900 mm
(3) Material
  a. Cone : co-cured graphite epoxy
  b. Adapter ring : Aluminum
(4) Attached system : Clamp bands
(5) Separation system : 4 springs
(6) Clamp band
  Maximum tension : 24.1 kN
(7) Maximum load per spring : 1670 N
(8) Adapter mass : 76 kg

Figure A3.1.1 shows the photograph of the 937M adapter. C
Figure A3.1.2 shows a general view of the 937M adapter .
Figure A3.1.3 shows details of the 937M adapter.
Figure A3.1.4 shows stay-out zone around the 937M adapter.
Figure A3.1.5 shows the limit load of the 937M adapter.
Figure A3.1.6 shows the spacecraft separation shock spectrum with the 937M adapter.
Figure A3.1.7 shows the limit load at separation plane of the 937M adapter.

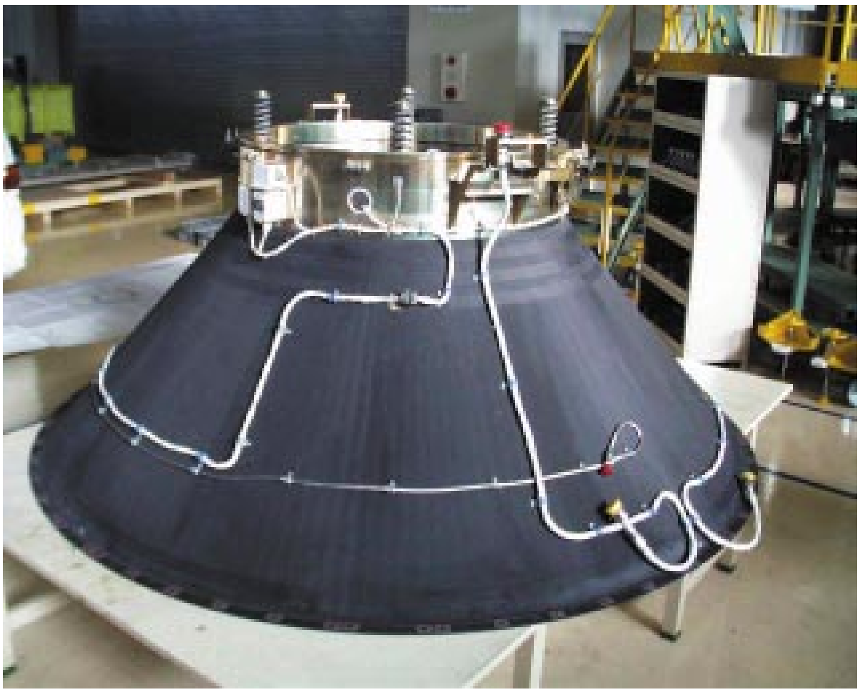
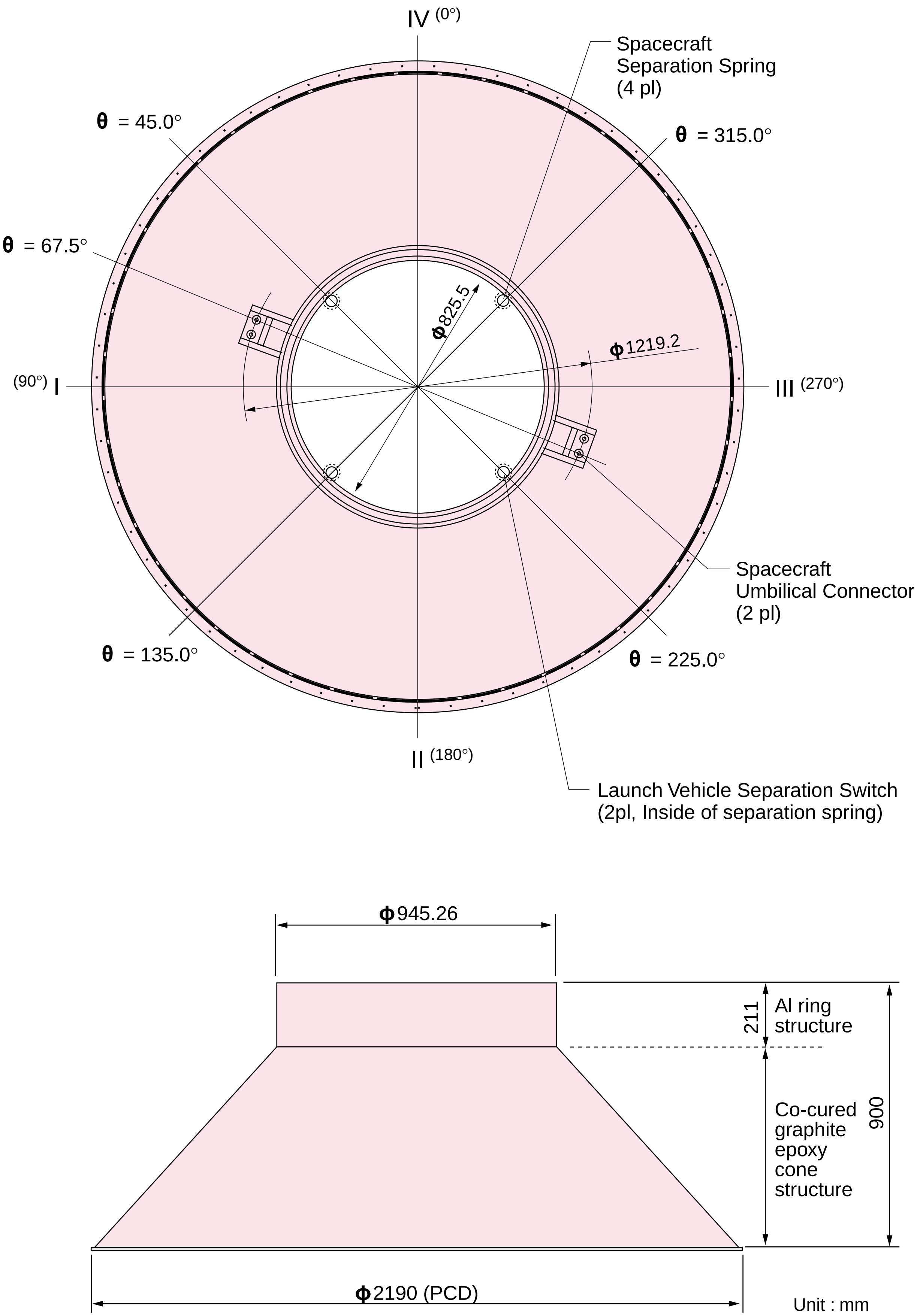
 C

Figure A3.1.1 Photograph of the 937M adapter



A
C
A
B
C

Figure A3.1.2 General view of the 937M adapter

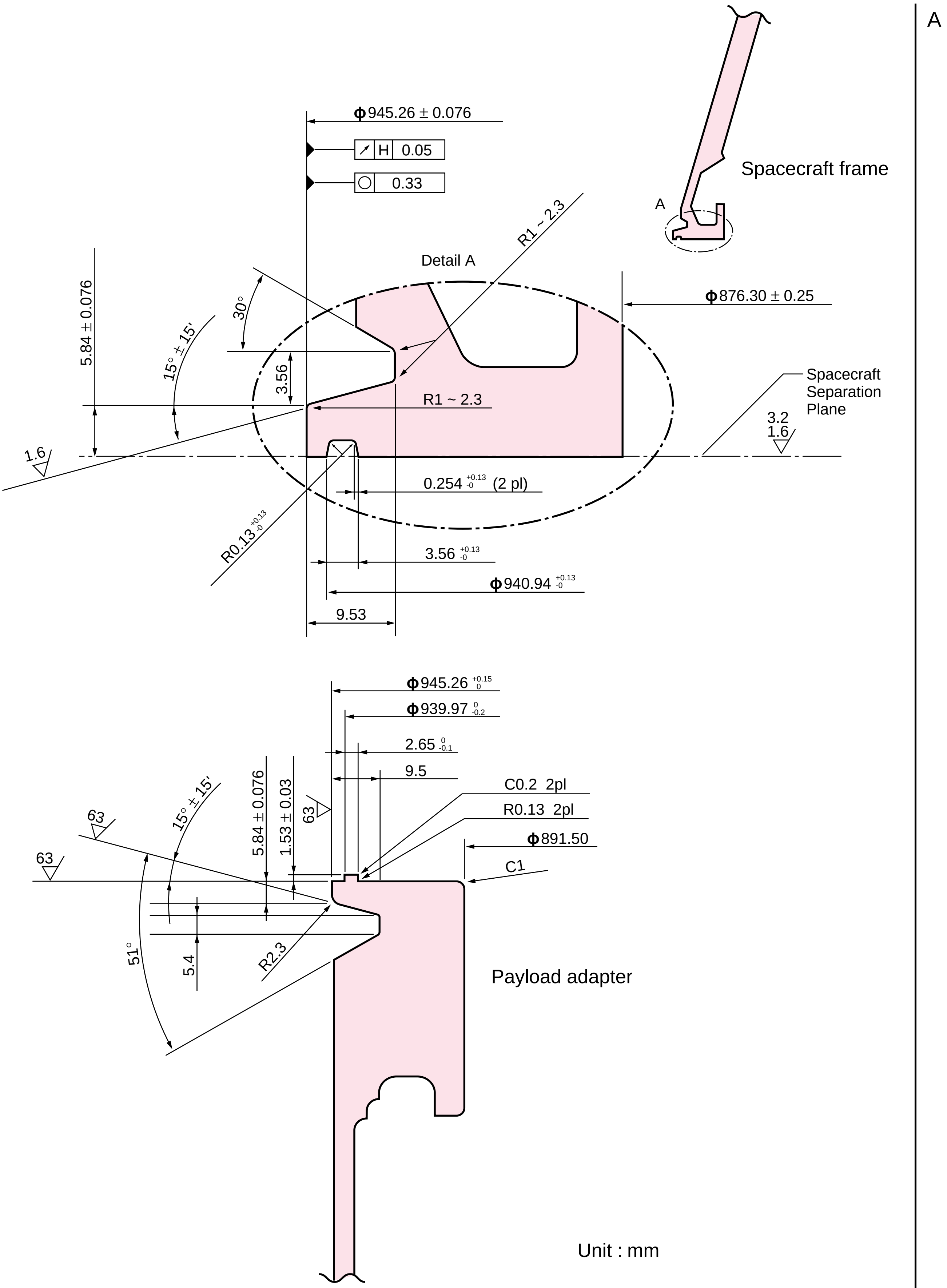
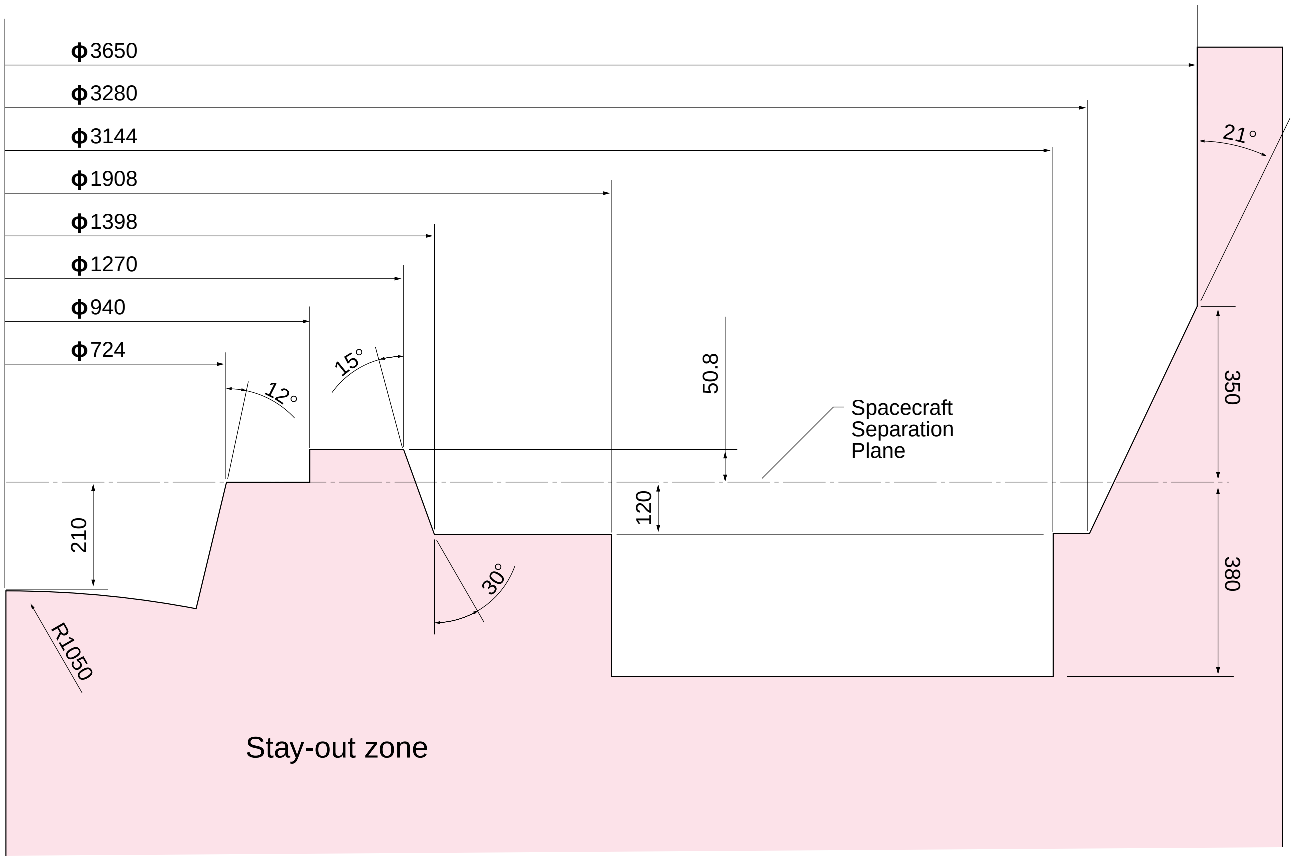


Figure A3.1.3 Details of the 937M adapter



A

C

Figure A3.1.4 Stay-out zone around the 937M adapter

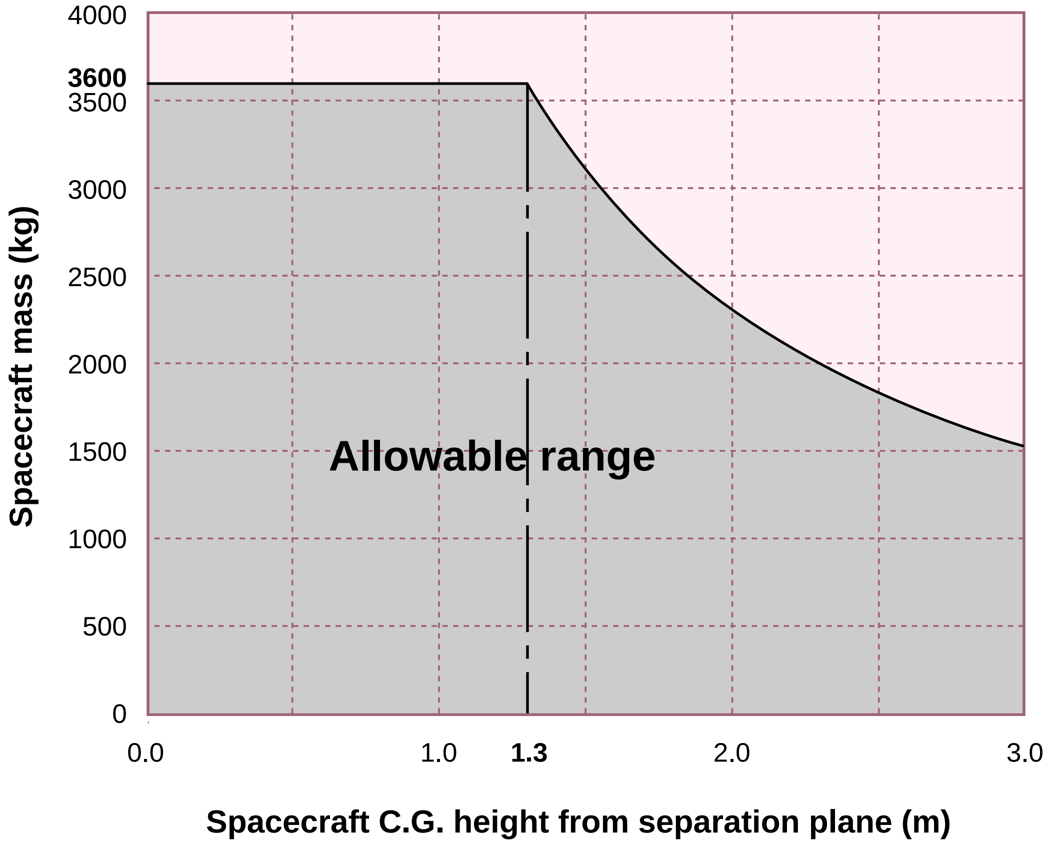
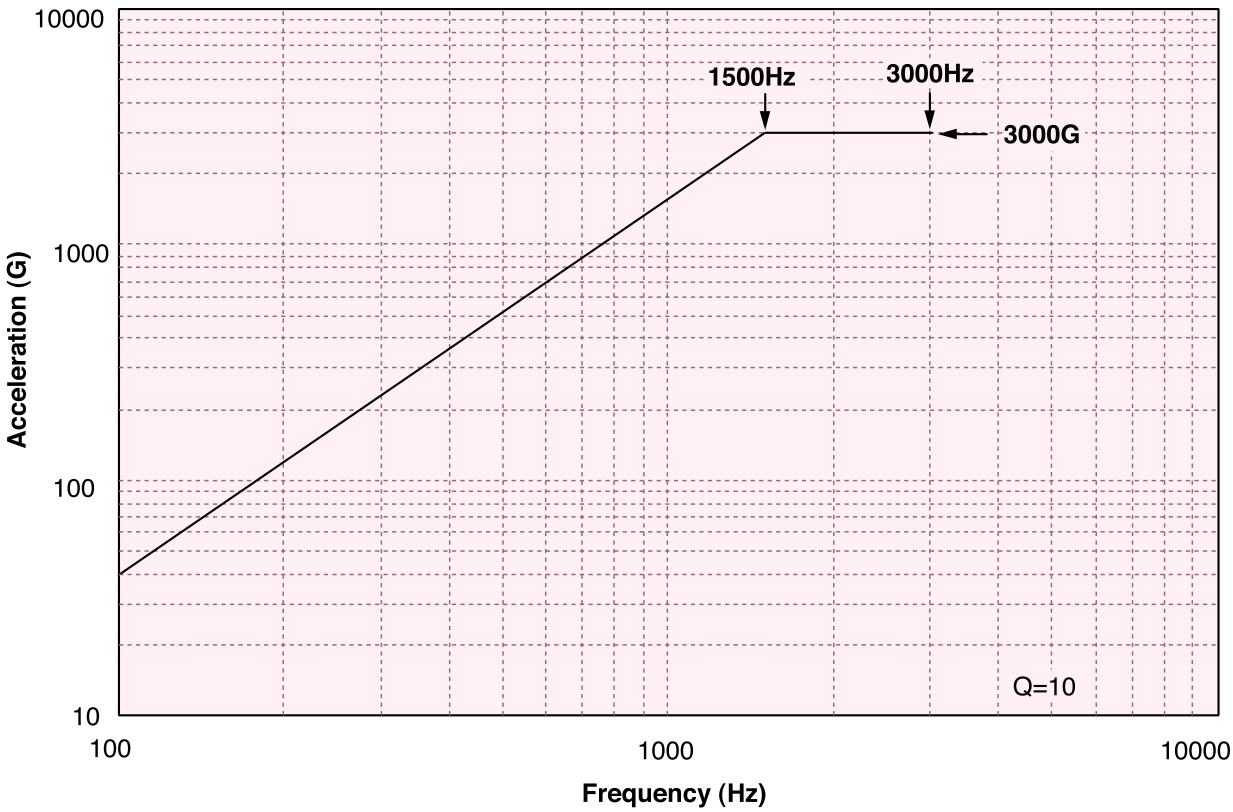


Figure A3.1.5 Limit load of the 937M adapter

A

C



G = 9.80665 (m/s^2) | B

Figure A3.1.6 Spacecraft separation shock spectrum with the 937M adapter | C

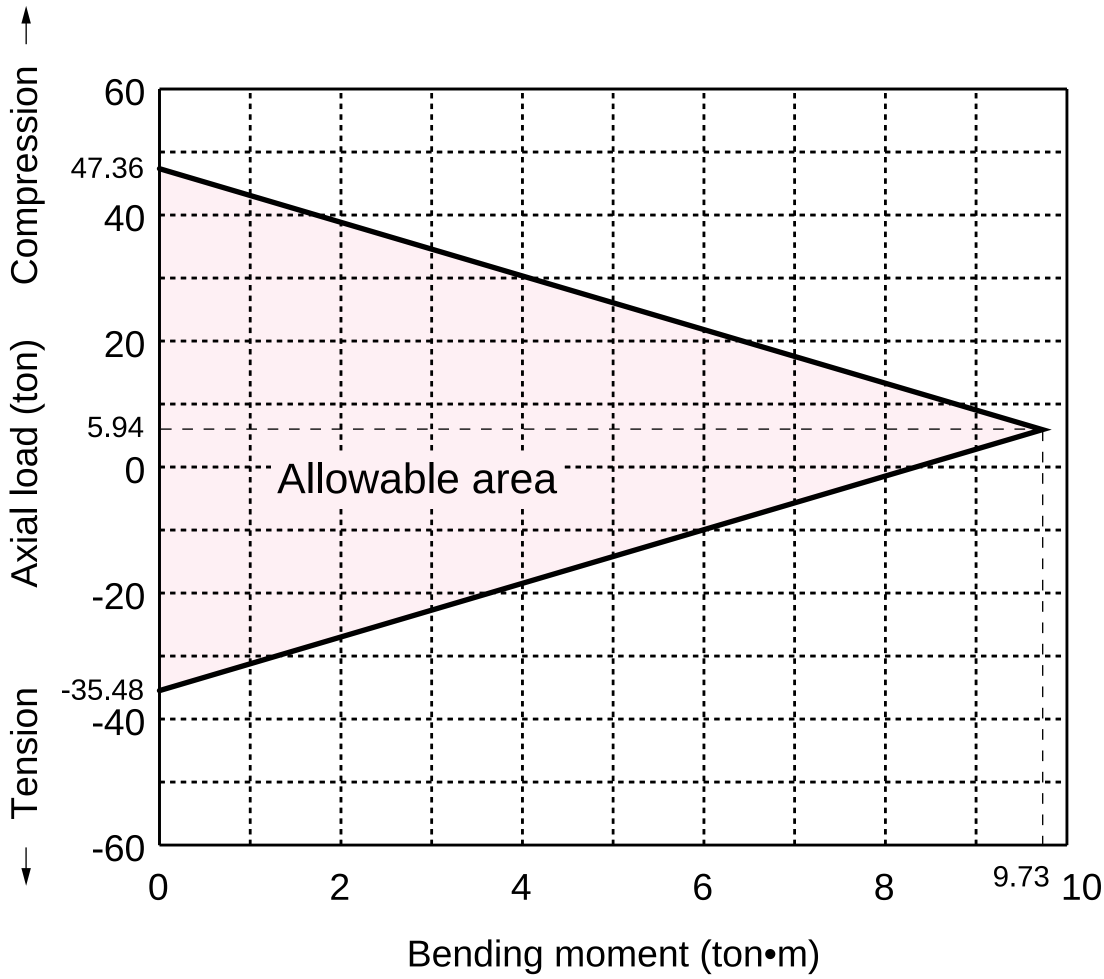


Figure A3.1.7 Limit loads at separation plane of the 937M adapter

B

C

APPENDIX 3.2 937MH ADAPTER

A

C

The main characteristics are as follows.

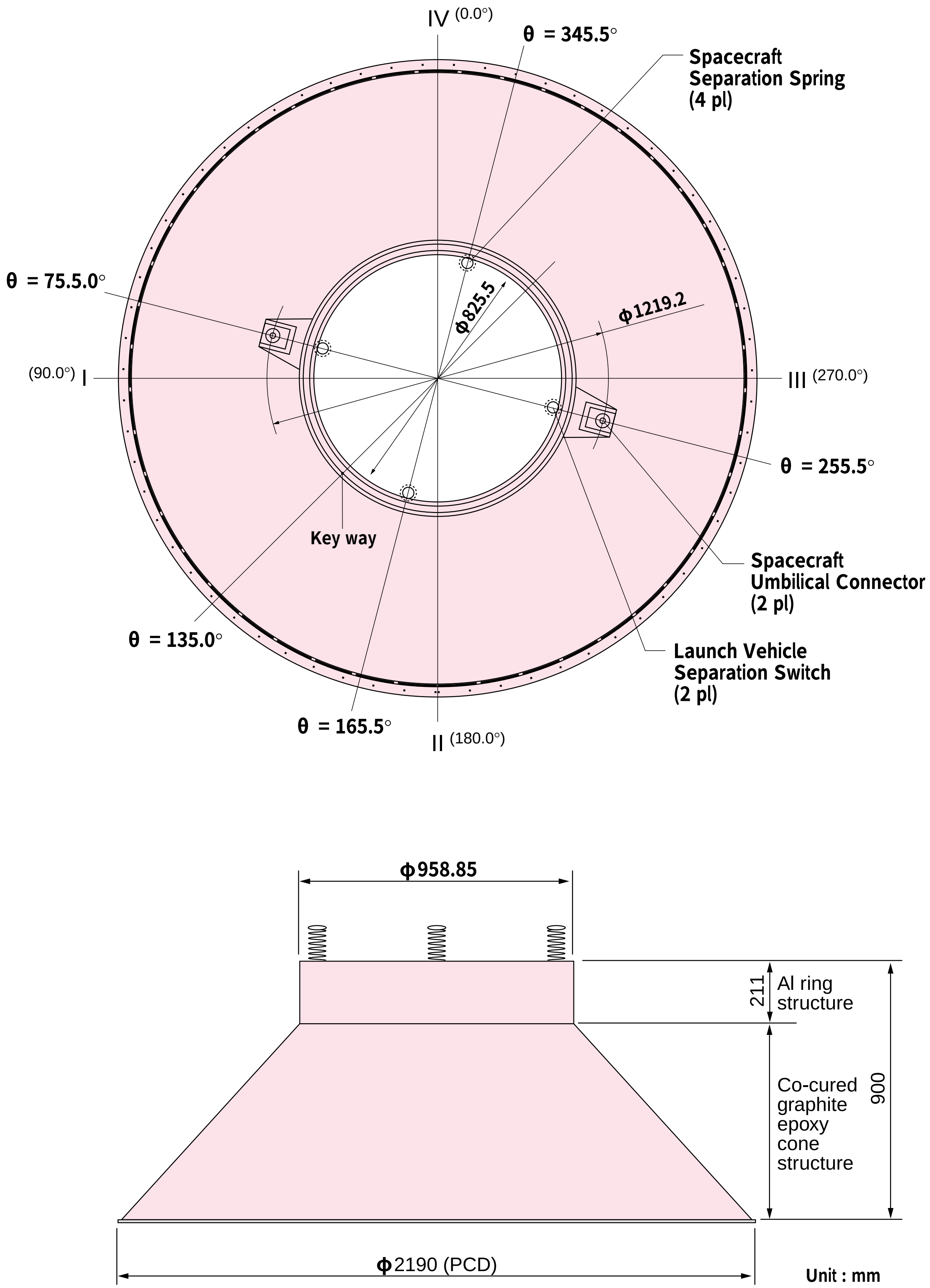
(1)	Interface diameter	: 958 mm
(2)	Height	: 900 mm
(3)	Material	
	a. Cone	: co-cured graphite epoxy
	b. Adapter ring	: Aluminum
(4)	Attached system	: Clamp bands
(5)	Separation system	: 4 springs
(6)	Clamp band	
	Maximum tension	: 14.9 kN
(7)	Maximum load per spring	: 1670 N
(8)	Adapter mass	: 80 kg

C

Figure A3.2.1 shows the photograph of the 937MH adapter.
Figure A3.2.2 shows a general view of the 937MH adapter .
Figure A3.2.3 shows details of the 937MH adapter.
Figure A3.2.4 shows stay-out zone around the 937MH adapter.
Figure A3.2.5 shows the limit load of the 937MH adapter.
Figure A3.2.6 shows the spacecraft separation shock spectrum with the 937MH adapter.
Figure A3.2.7 shows the limit load at separation plane of the 937MH adapter.

C

Figure A3.2.1 Photograph of the 937MH adapter



C
A
C

Figure A3.2.2 General view of the 937MH adapter

C

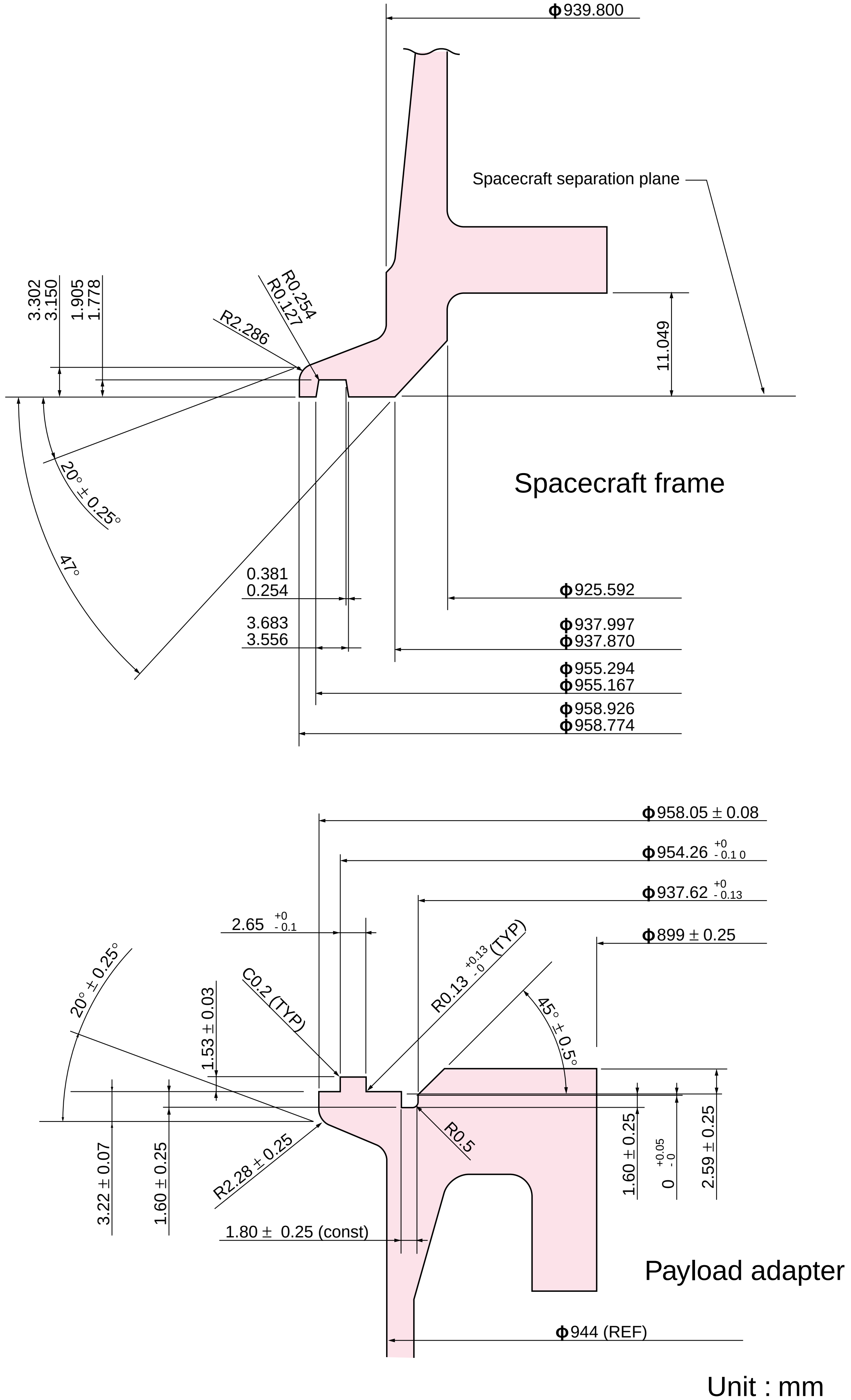


Figure A3.2.3 Details of the 937MH adapter

A

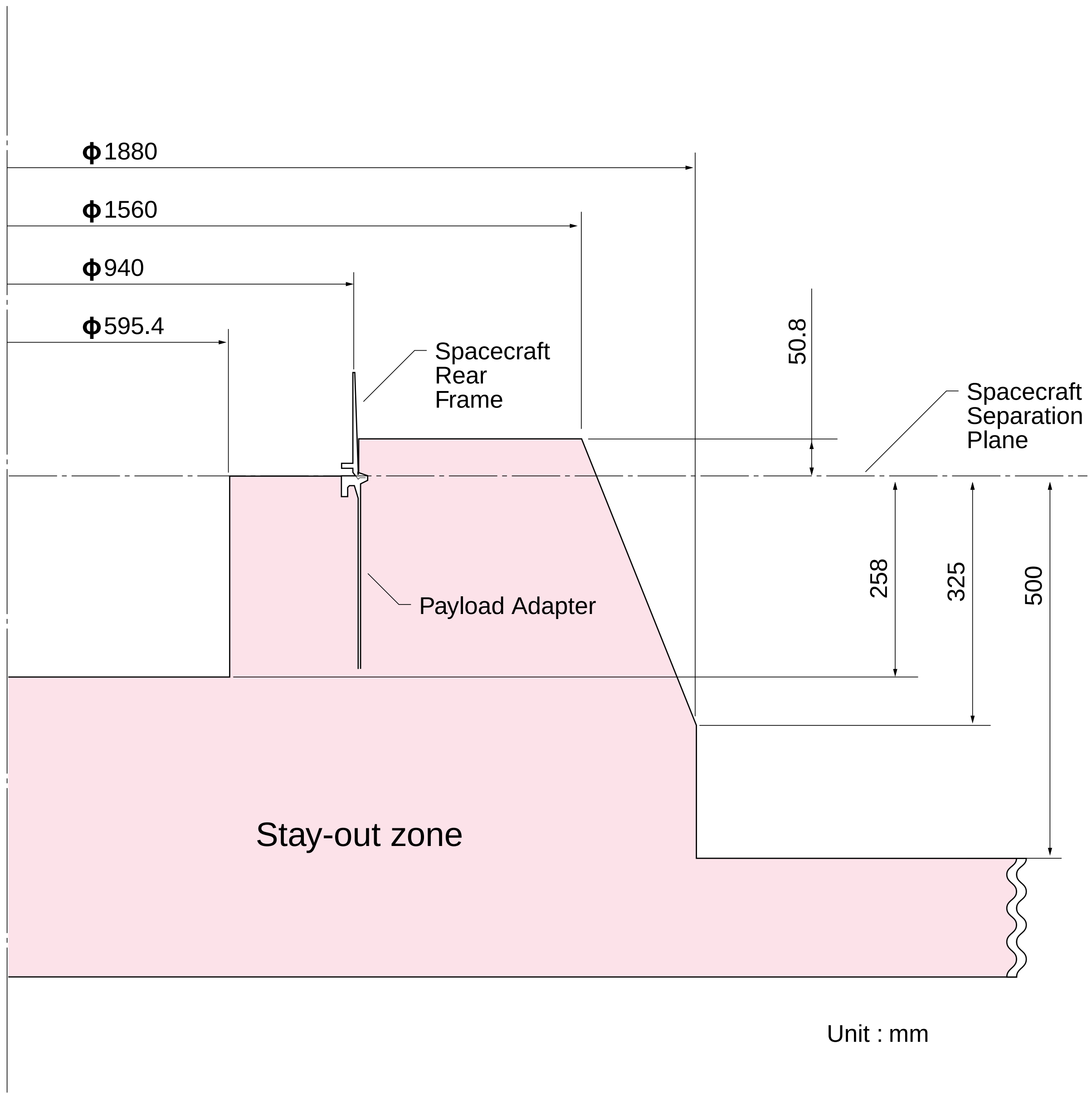


Figure A3.2.4 Stay-out zone around the 937MH adapter

C

A

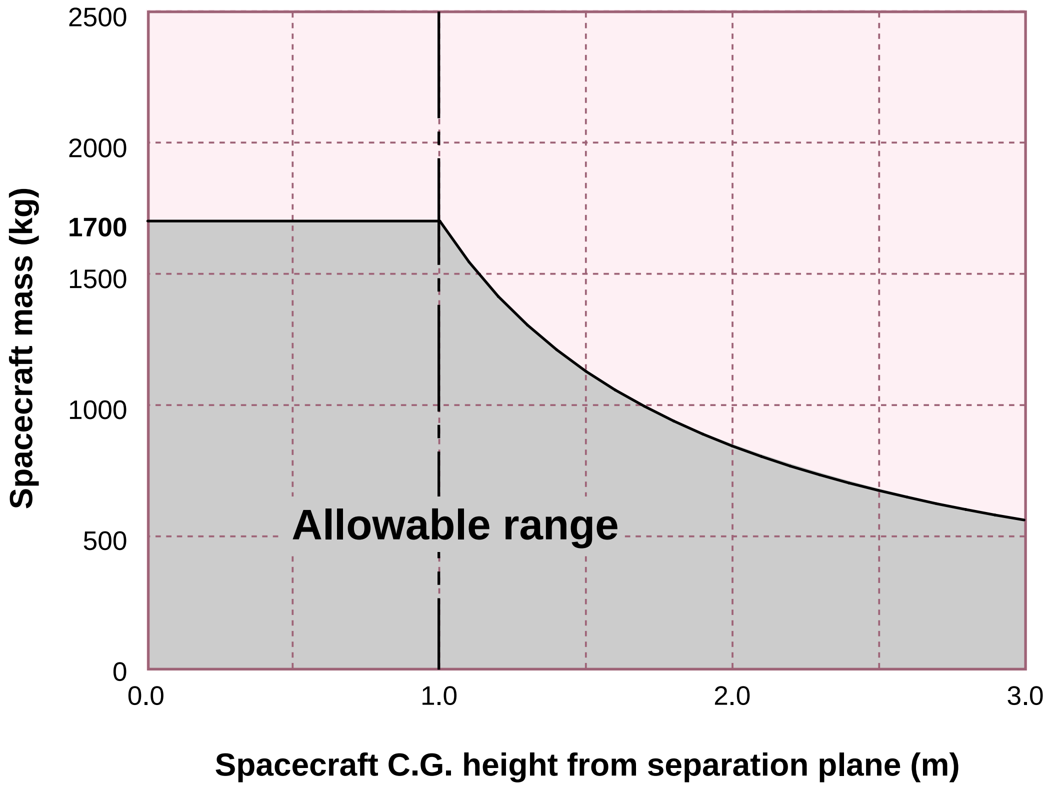


Figure A3.2.5 Limit load of the 937MH adapter

| C

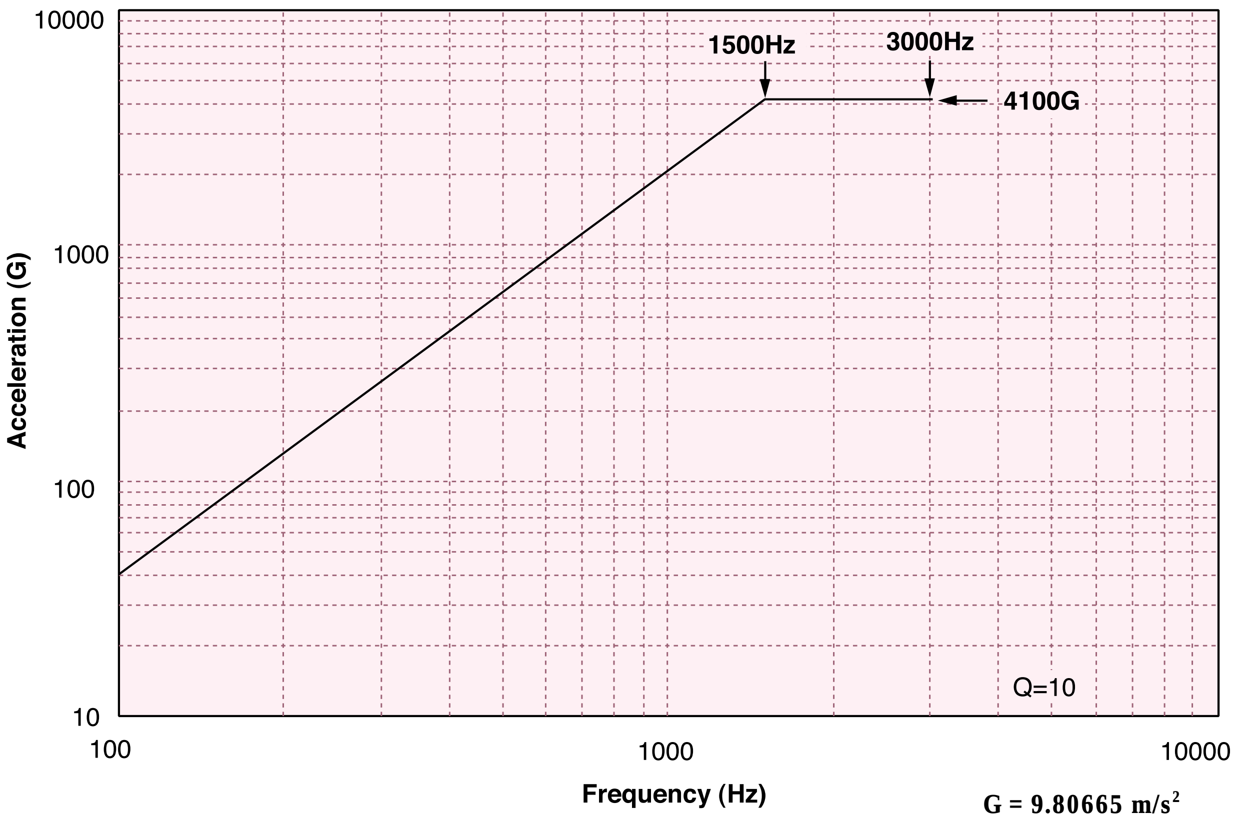
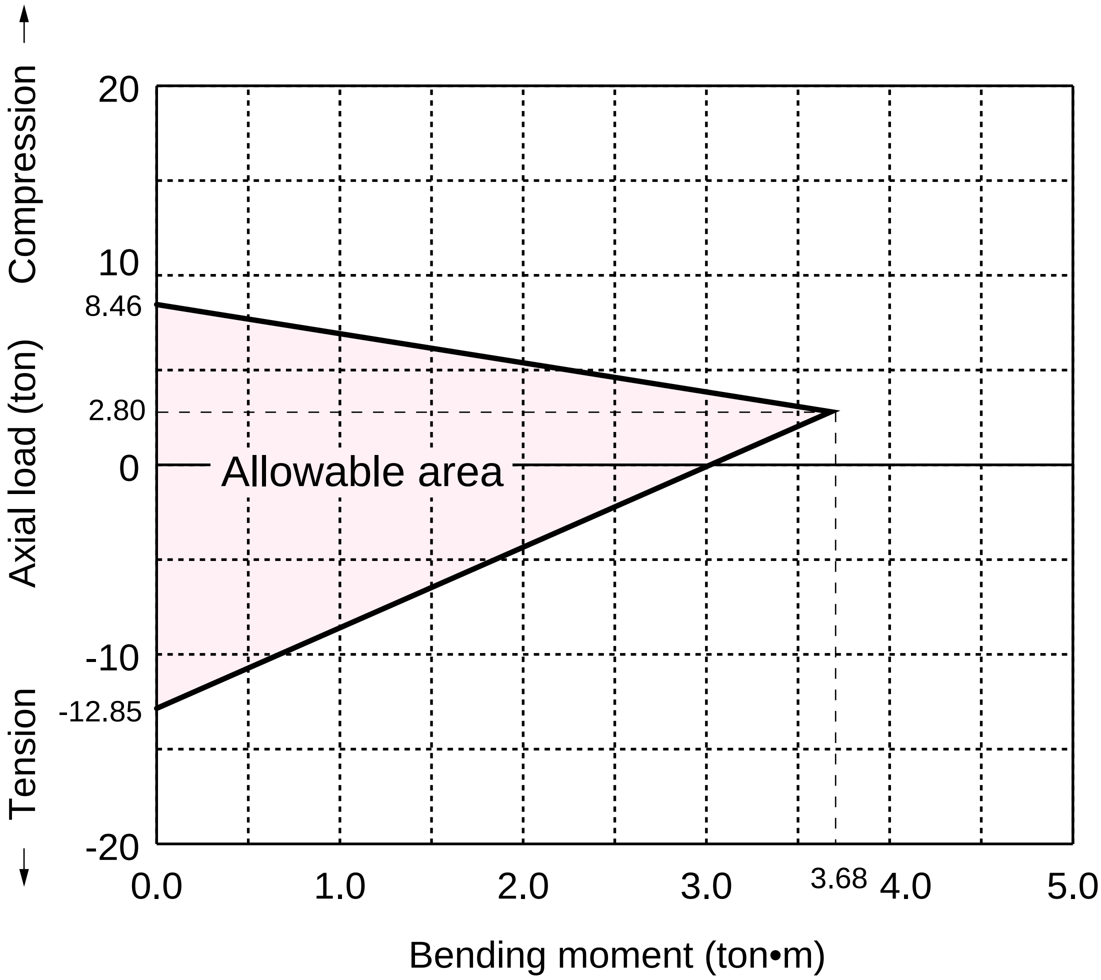


Figure A3.2.6 Spacecraft separation shock spectrum with the 937MH adapter



B

Figure A3.2.7 Limit loads at separation plane of the 937MH adapter

C

APPENDIX 3.3 937M-SPIN ADAPTER

A

C

The main characteristics are as follows.

(1) Interface diameter : 959 mm
(2) Height : 605 mm
(3) Material : Aluminum semi-monocoque
(4) Attached system : Clamp bands
(5) Separation system : 4 springs
(6) Clamp band
Maximum tension : 13.5 kN
(7) Maximum load per spring : 1670 N
(8) Adapter mass : 130 kg

The plume shield can be installed if necessary.

The adapter has a spin-table section, which can spin the spacecraft up to 100[*1] rpm with an angular acceleration of less than 10 rad/sec^2.

A

Figure A3.3.1 shows the photograph of the 937M-SPIN adapter.

Figure A3.3.2 shows a general view of the 937M-SPIN adapter.

Figures A3.3.3 and A3.3.4 show details of the 937M-SPIN adapter.

Figure A3.3.5 shows the stay-out zone around the 937M-SPIN adapter.

Figure A3.3.6 shows the limit load of the 937M-SPIN adapter.

Figure A3.3.7 shows the spacecraft separation shock spectrum of the 937M-SPIN adapter.

<Note>

*1 : In this case, the moment of inertia related to the vehicle X axis (Ixx) is less than T.B.D. kg·m·sec^2.

Figure A3.3.1 Photograph of the 937M-SPIN adapter

A

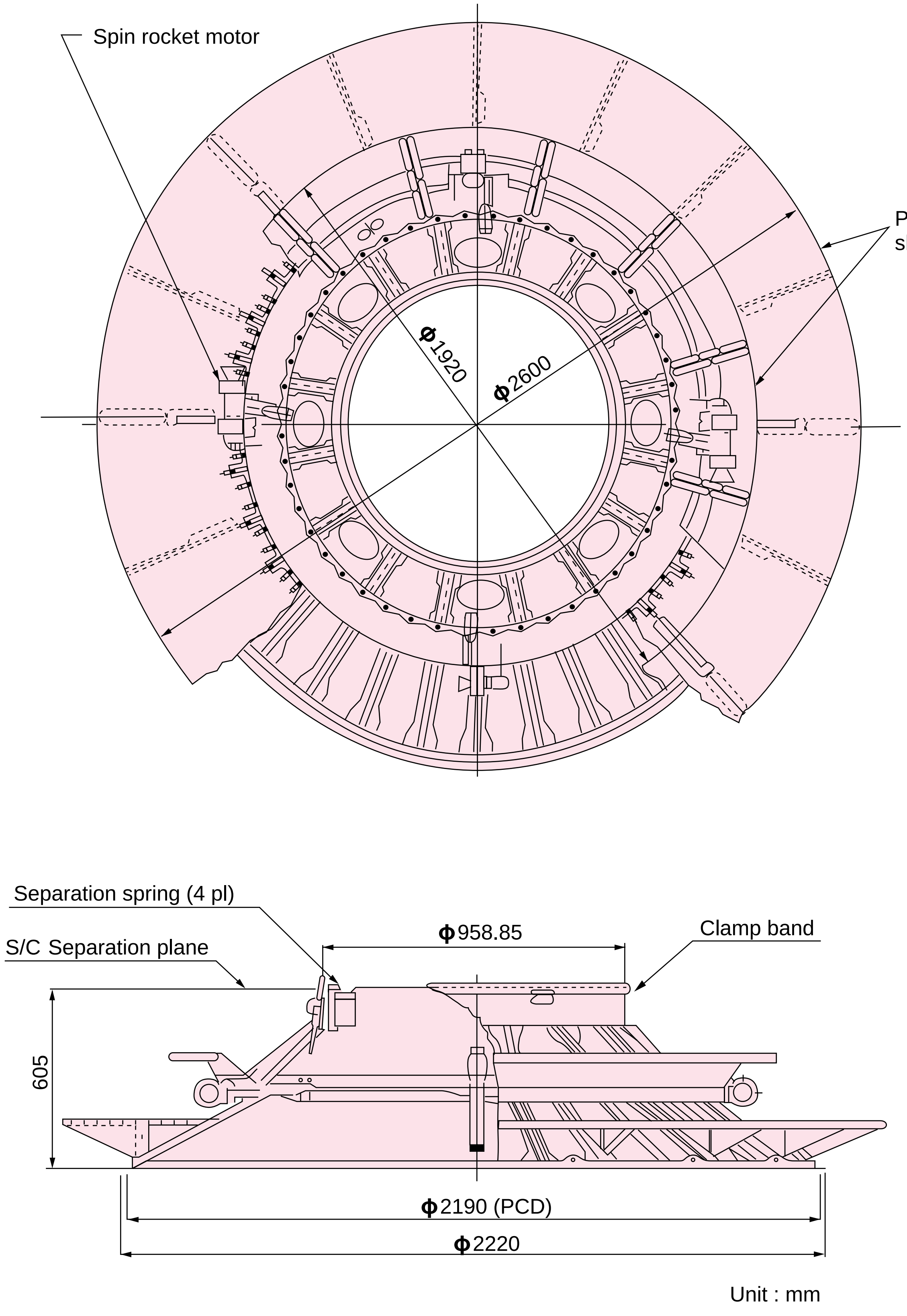
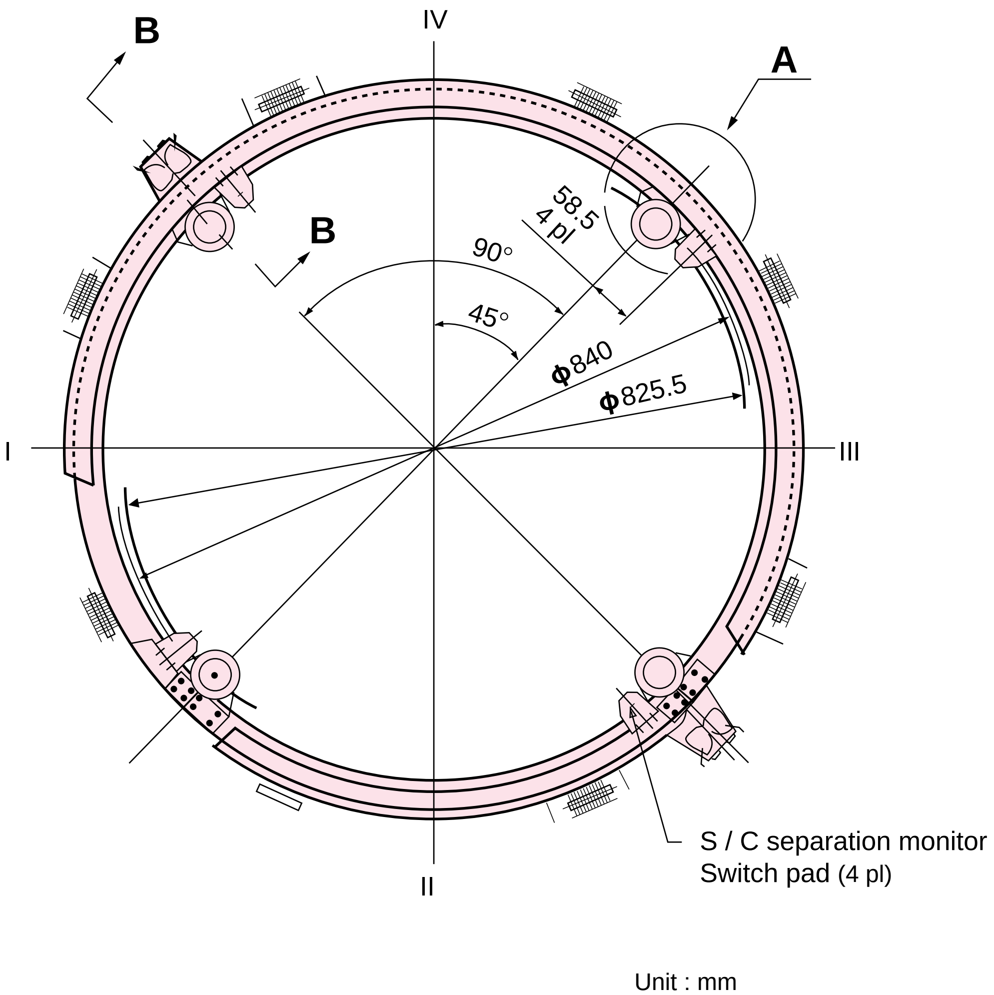


Figure A3.3.2 General view of 937M-SPIN adapter

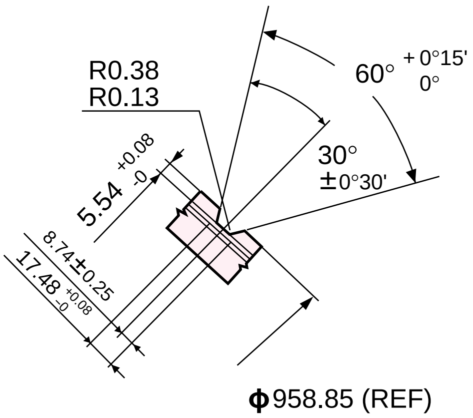
A



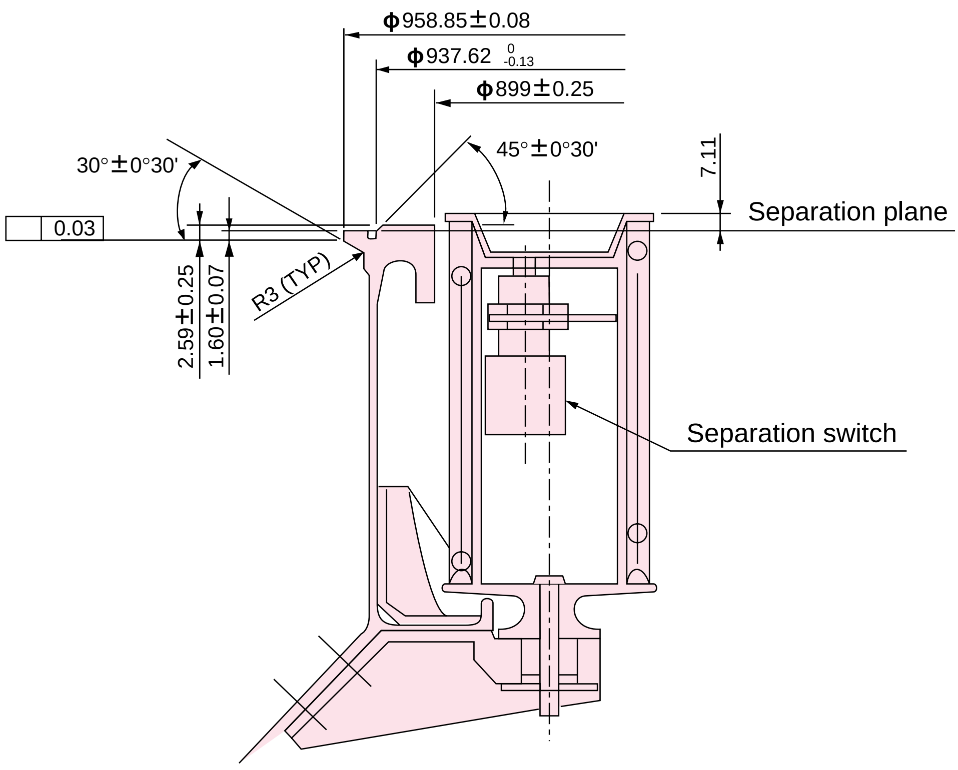
C

Figure A3.3.3 Details of 937M-SPIN adapter #1

A



Detail A

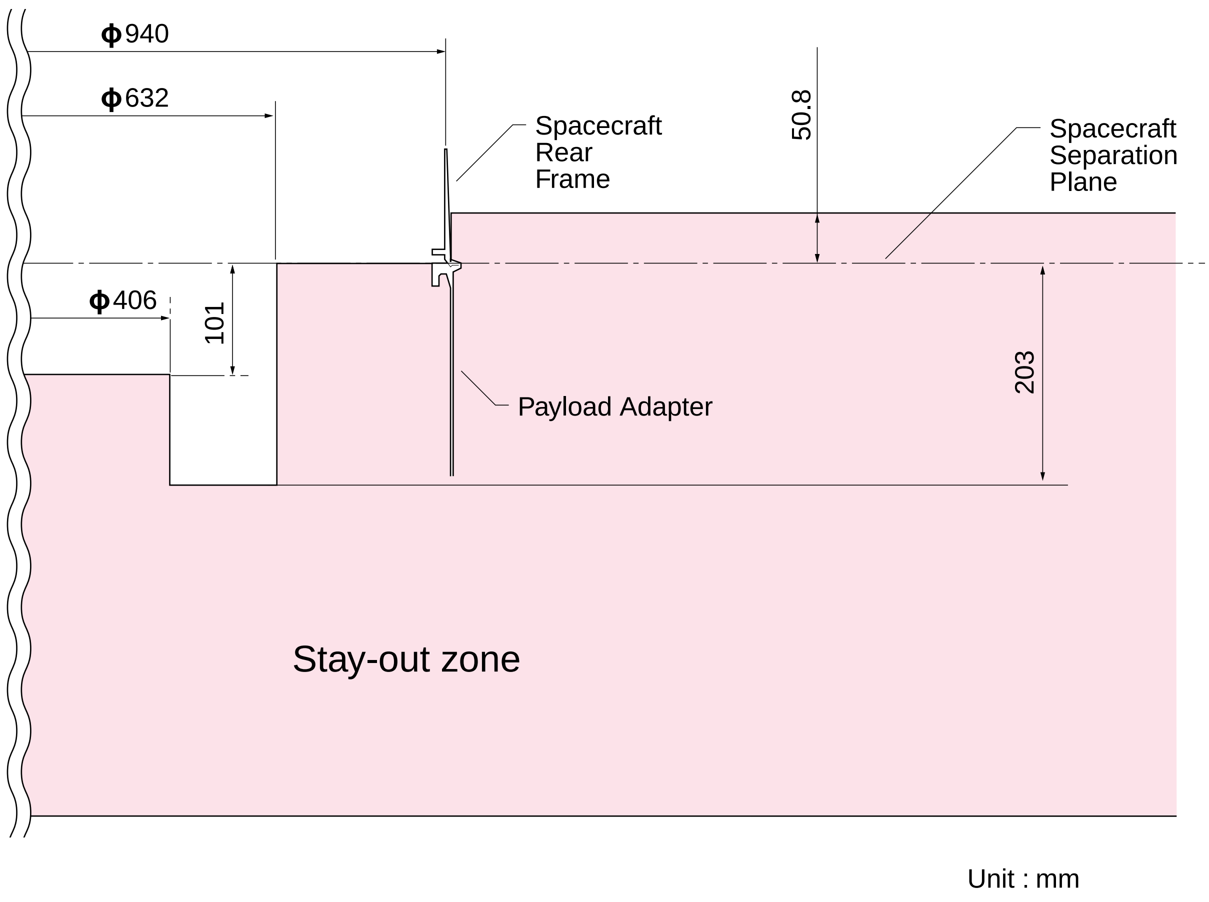


Section **B**

Unit : mm

Figure A3.3.4 Details of 937M-SPIN adapter #2

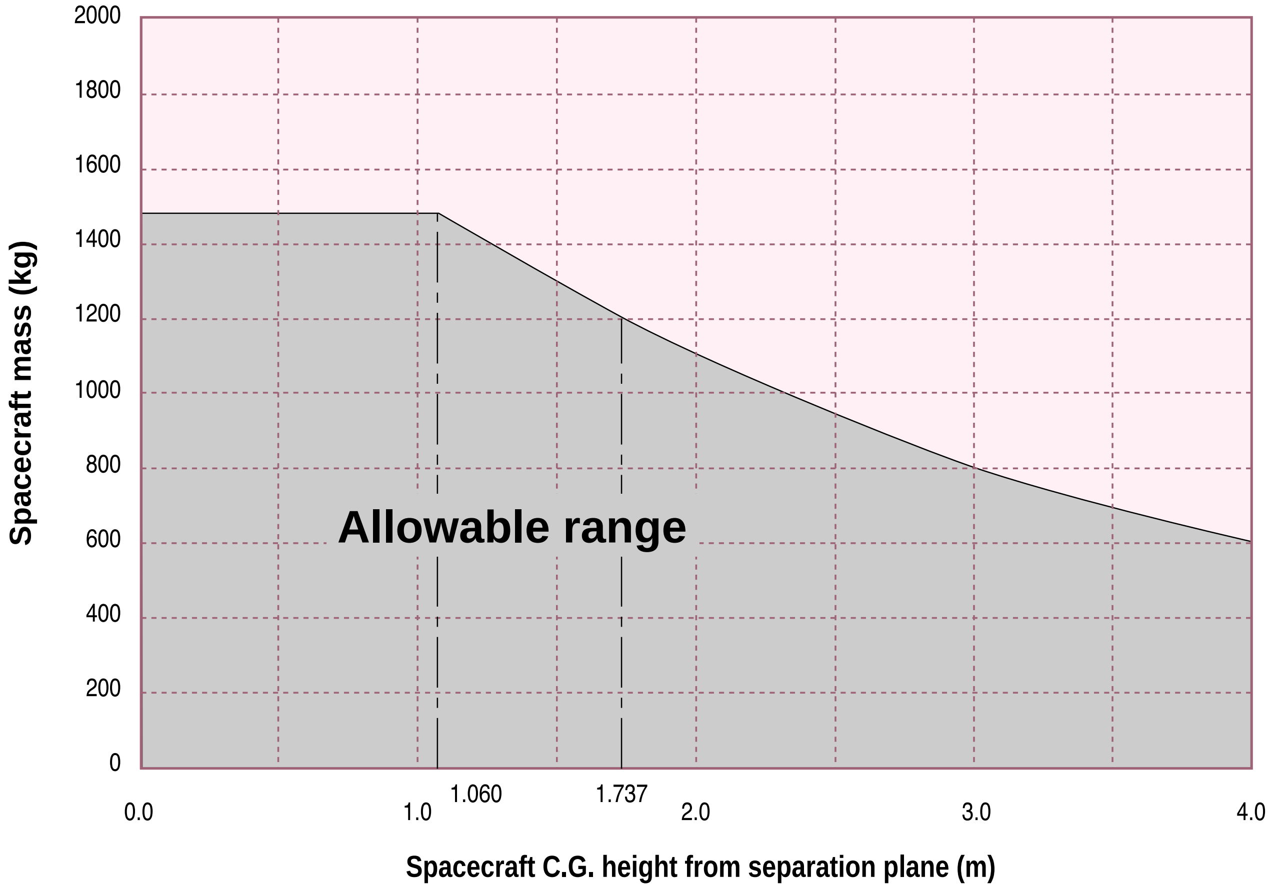
A



C

Figure A3.3.5 Stay-out zone around the 937M-SPIN adapter

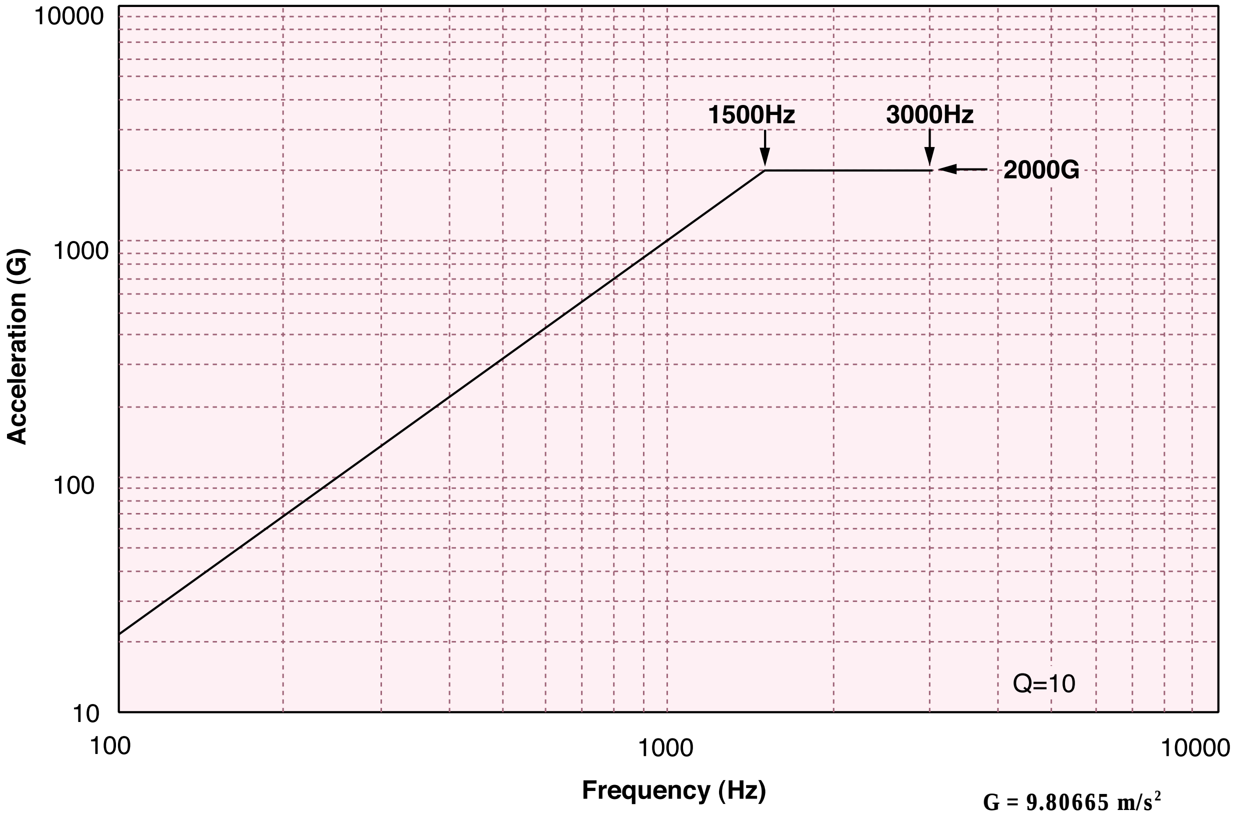
A



C

Figure A3.3.6 Limit load of the 937M-SPIN adapter

A



C

Figure A3.3.7 Spacecraft separation shock spectrum of the 937M-SPIN adapter

A

APPENDIX 3.4 937M-SPIN-A ADAPTER A

The main characteristics are as follows. C

(1) Interface diameter : 958 mm
(2) Height : 1,000 mm
(3) Material : Aluminum semi-monocoque
(4) Attached system : Clamp bands
(5) Separation system : 4 springs
(6) Clamp band
    Maximum tension : 14.9 kN
(7) Maximum load per spring : 1670 N
(8) Adapter mass : 150 kg

The plume shield can be installed if necessary. A

The adapter has a spin-table section, which can spin the spacecraft up to 50*1 C
rpm with an angular acceleration of less than 5 rad / sec^2. A

Figure A3.4.1 shows the photograph of the 937M-SPIN-A adapter. C

Figure A3.4.2 shows the general view of the 937M-SPIN-A adapter.

Figure A3.4.3 shows the details of the 937M-SPIN-A adapter.

Figure A3.4.4 shows the stay-out zone around the 937M-SPIN-A adapter.

Figure A3.4.5 shows the limit load of the 937M-SPIN-A adapter.

Figure A3.4.6 shows the spacecraft separation shock spectrum of the 937M-SPIN-A adapter.

Figure A3.4.7 shows the limit load at separation plane of the 937M-SPIN-A adapter.

<Note> A

*1 : In this case, the moment of inertia related to the vehicle X axis (Ixx) is less than T.B.D. kg·m·sec^2.

C

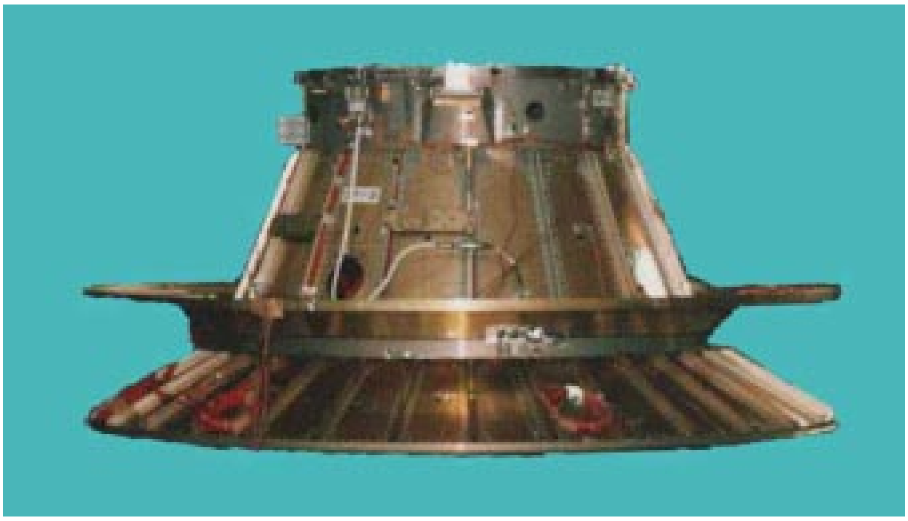
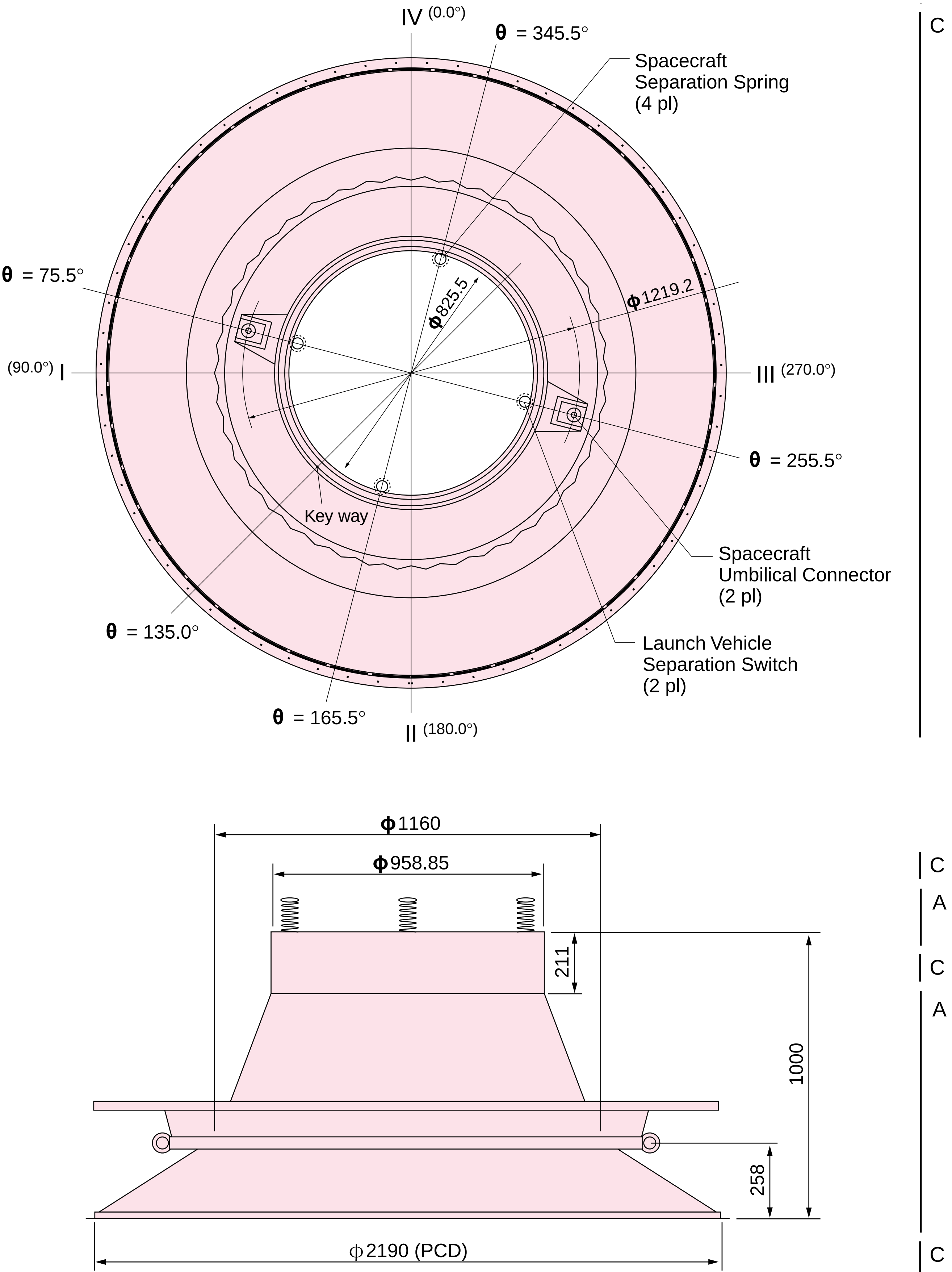
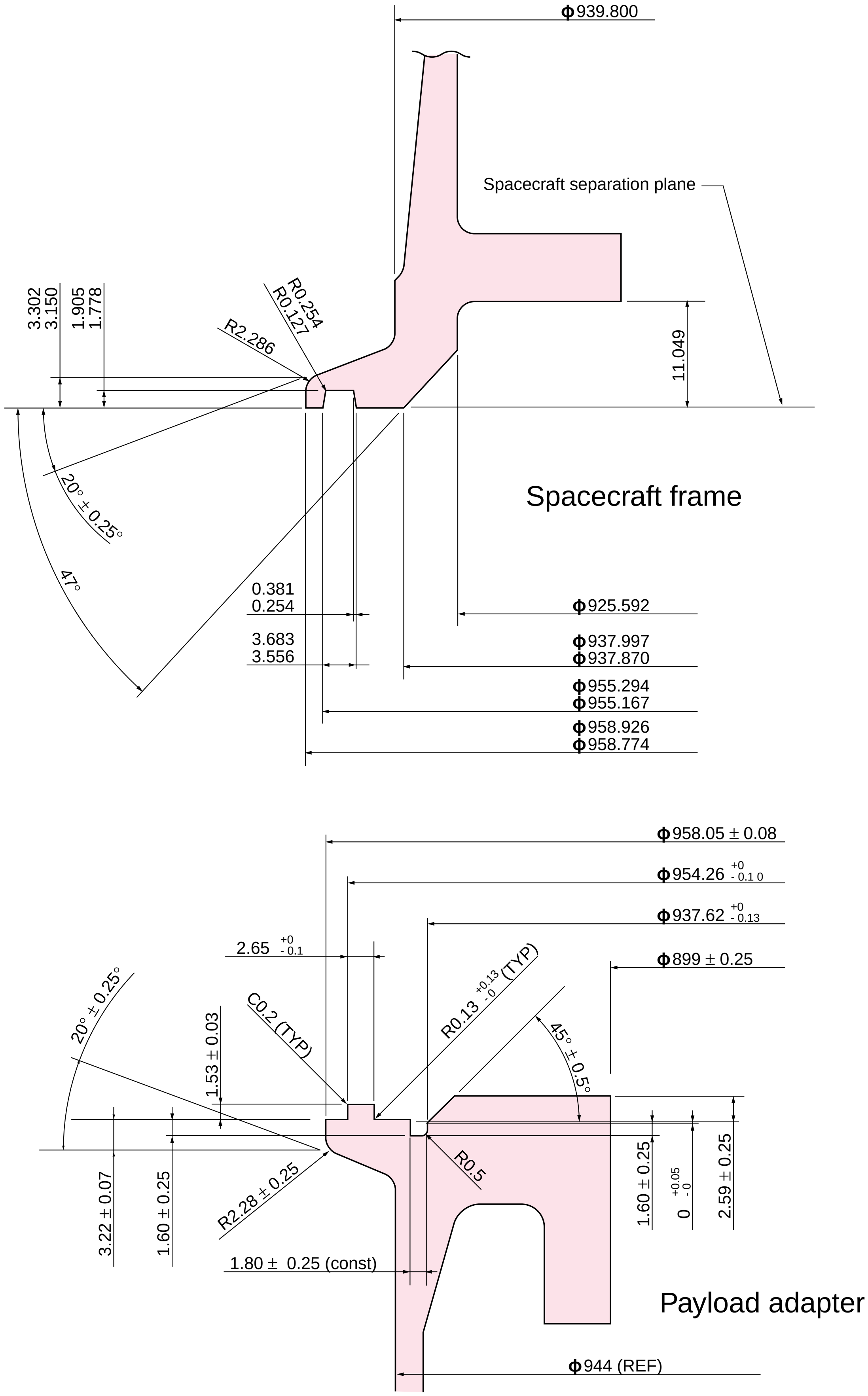


Figure A3.4.1 Photograph of the 937M-SPIN-A adapter



Unit : mm

Figure A3.4.2 General view of 937M-SPIN-A adapter



Unit : mm

Figure A3.4.3 Details of 937M-SPIN-A adapter

A
B
A
C
A
C

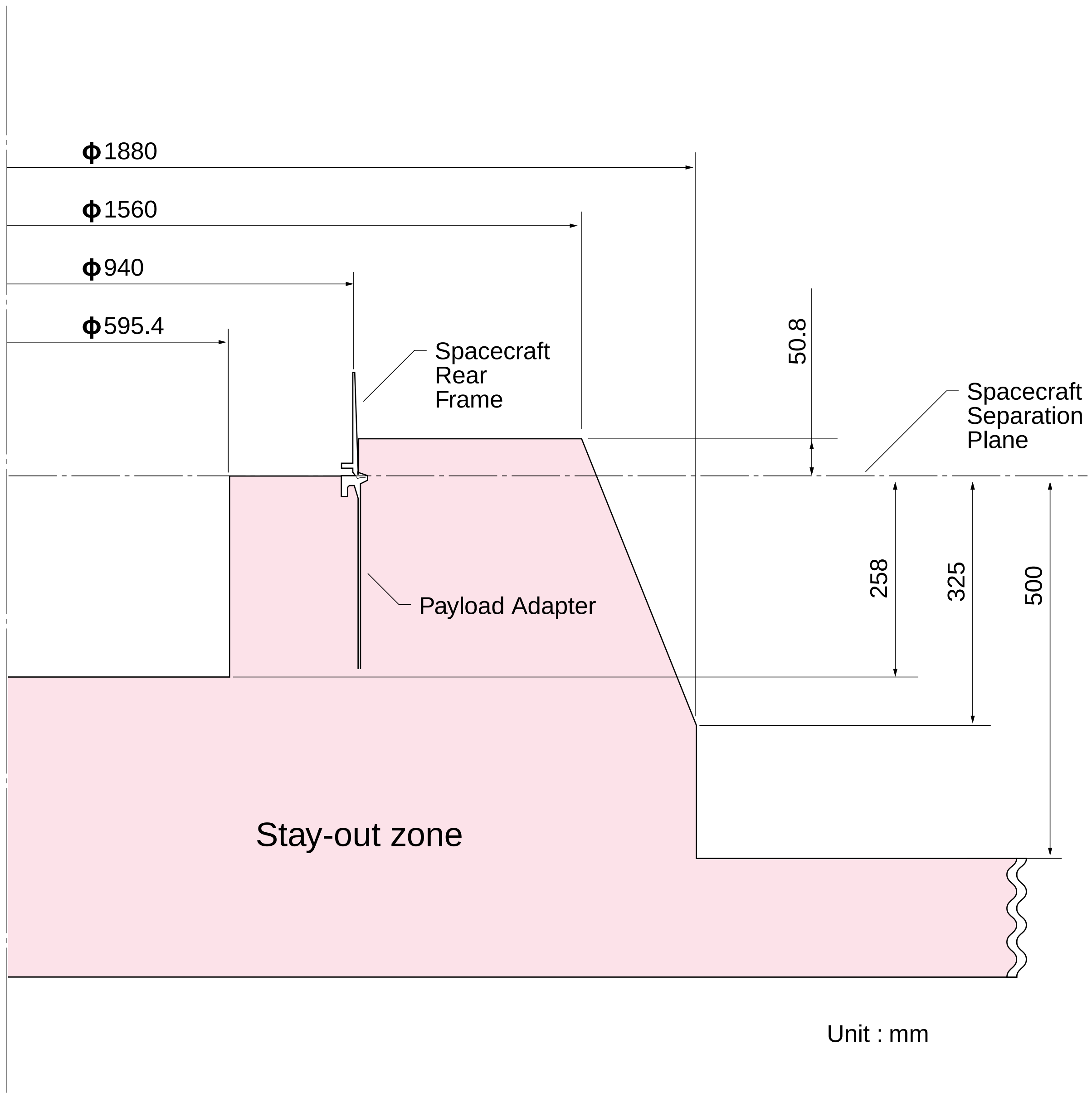


Figure A3.4.4 Stay-out zone around the 937M-SPIN-A adapter

A

C

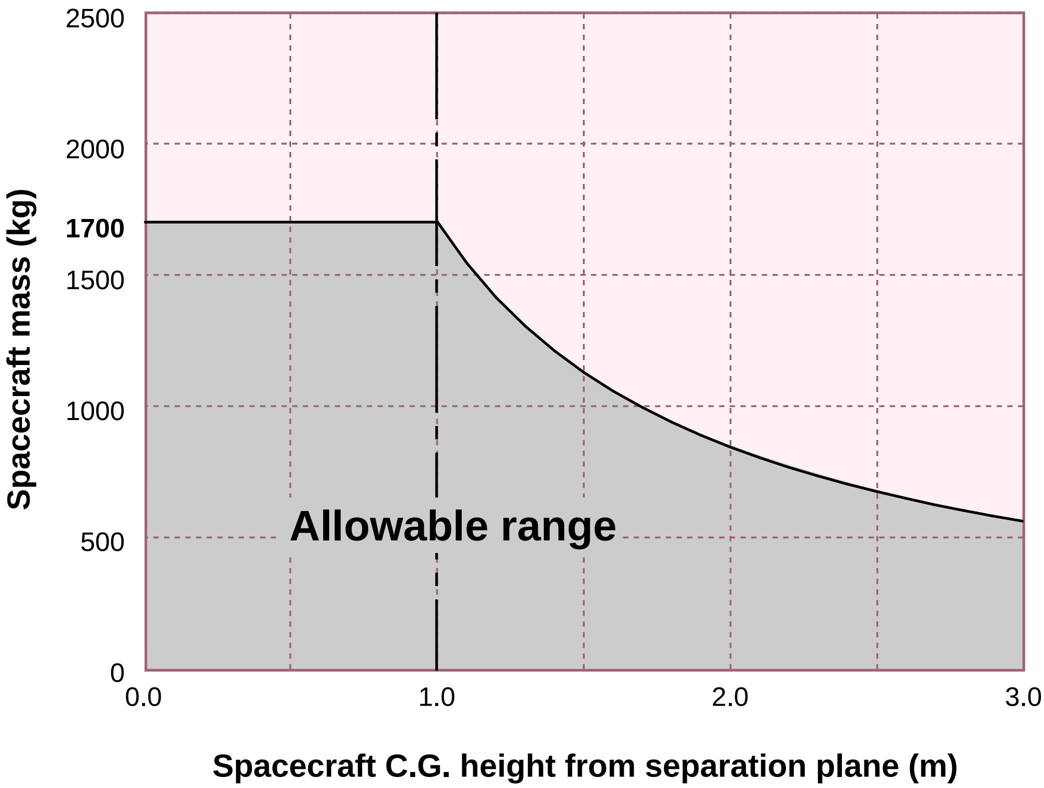
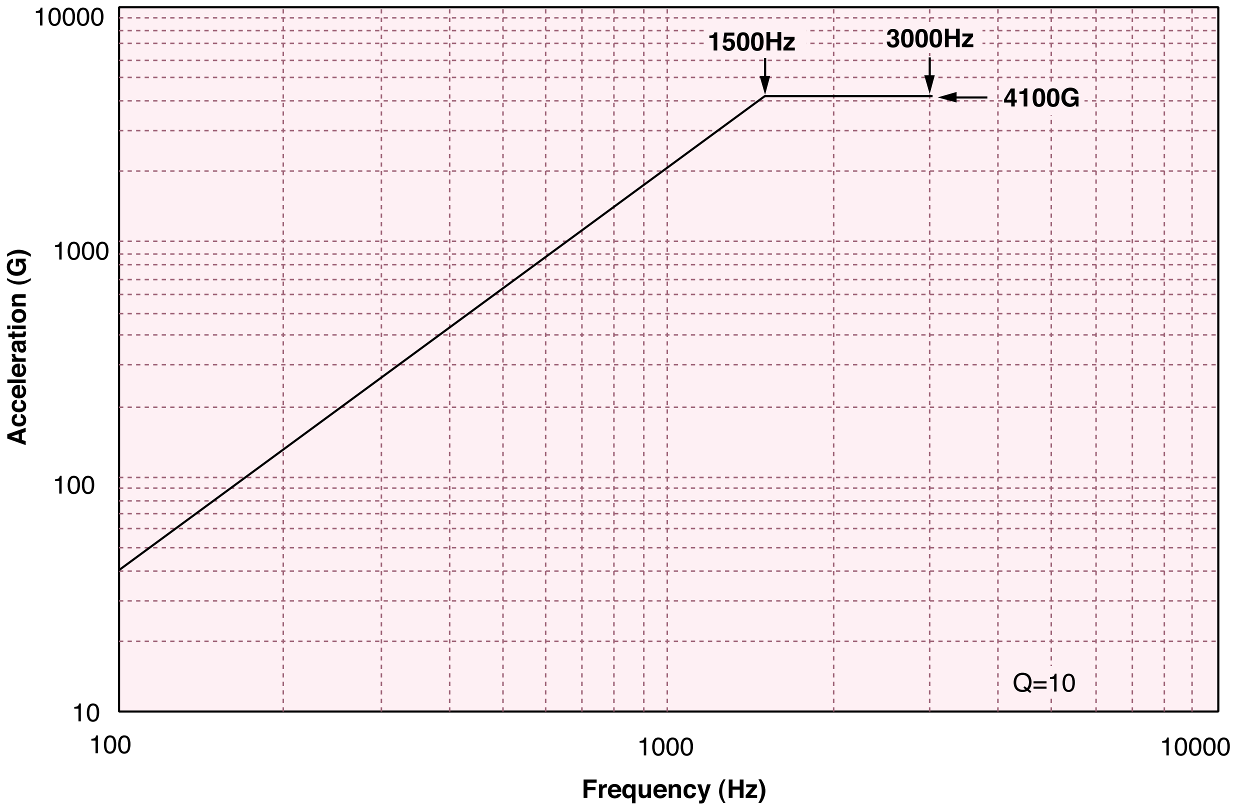


Figure A3.4.5 Limit load of the 937M-SPIN-A adapter

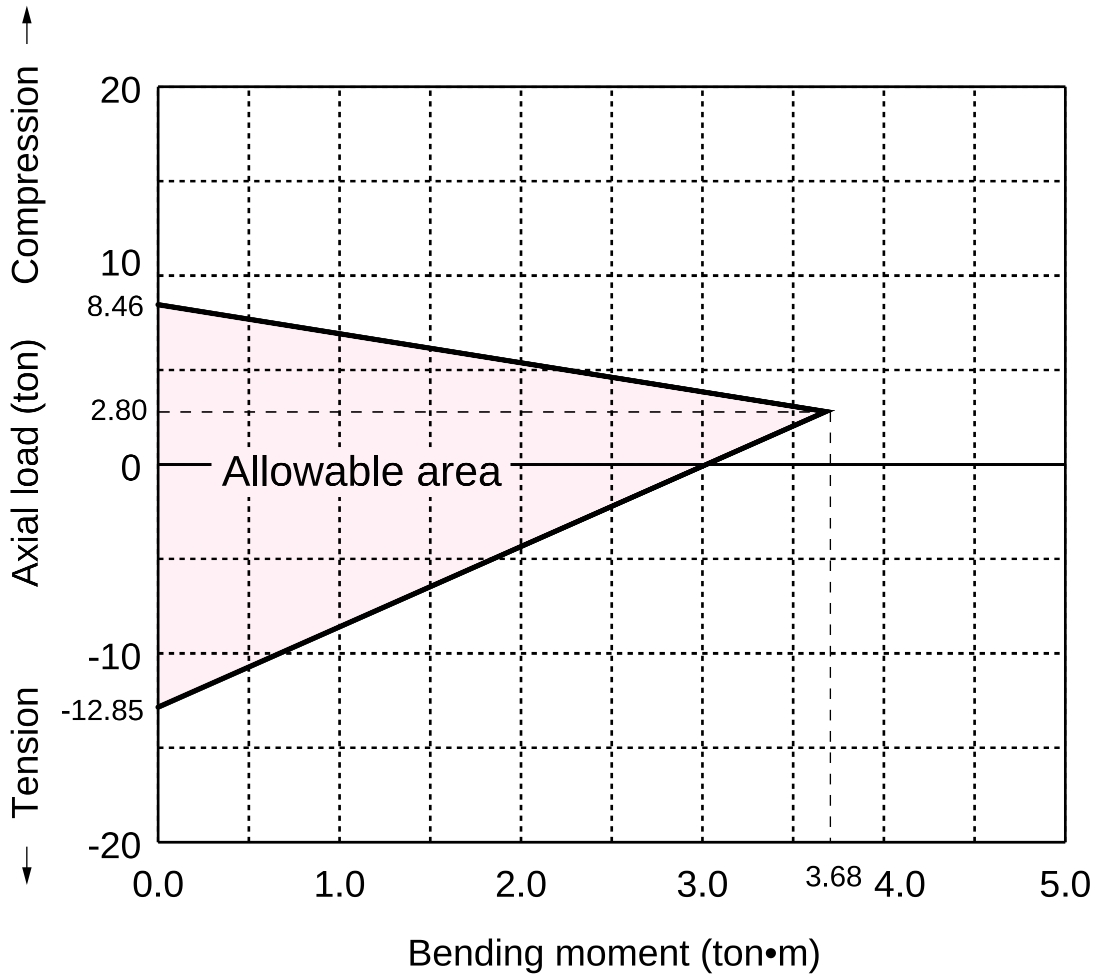
A

C



G = 9.80665 m/s^2

Figure A3.4.6 Spacecraft separation shock spectrum of the 937M-SPIN-A adapter



B

Figure A3.4.7 Limit loads at separation plane of the 937M-SPIN-A adapter

C

APPENDIX 3.5 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 system	: 4 or 8 springs
(6)	Clamp band	
	Maximum tension	: 36.8 kN
(7)	Maximum load per spring	: 1670 N
(8)	Adapter mass	: 100 kg

This adapter has two ventholes of 45 mmø (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 A3.5.1 shows the photograph of the 1194M adapter.
Figure A3.5.2 shows the general view of the 1194M adapter.
Figures A3.5.3 to A3.5.5 show the details of the 1194M adapter.
Figure A3.5.6 to A3.5.7 show the stay-out zone around the 1194M adapter.
Figure A3.5.8 shows the limit load of the 1194M adapter.
Figure A3.5.9 shows the spacecraft separation shock spectrum of the 1194M adapter.
Figure A3.5.10 shows the limit loads at separation plane of the 1194M adapter.

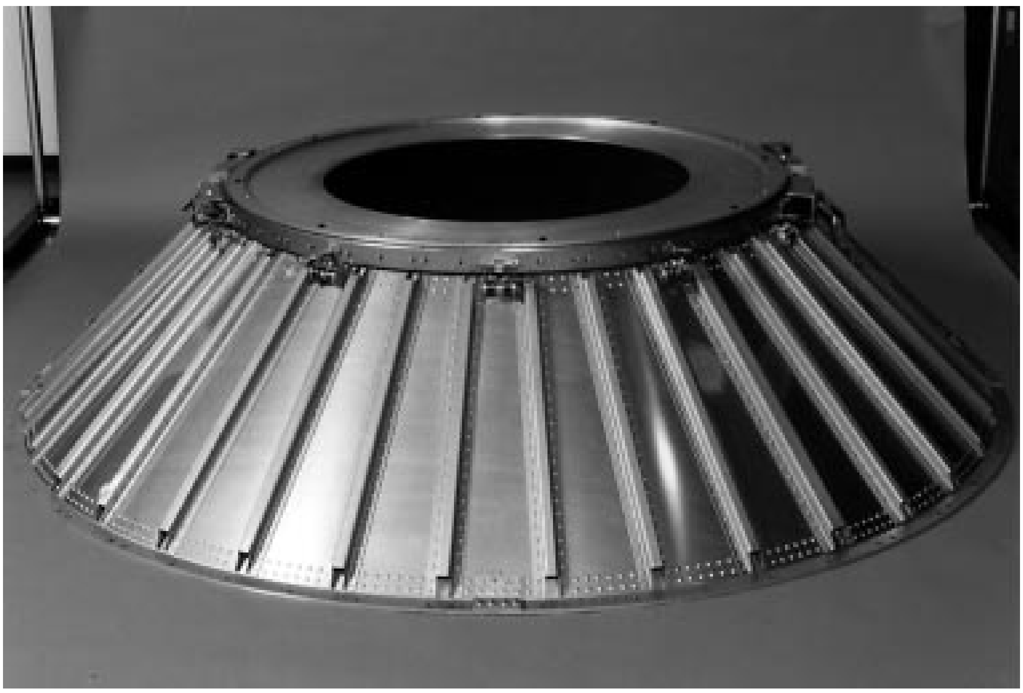
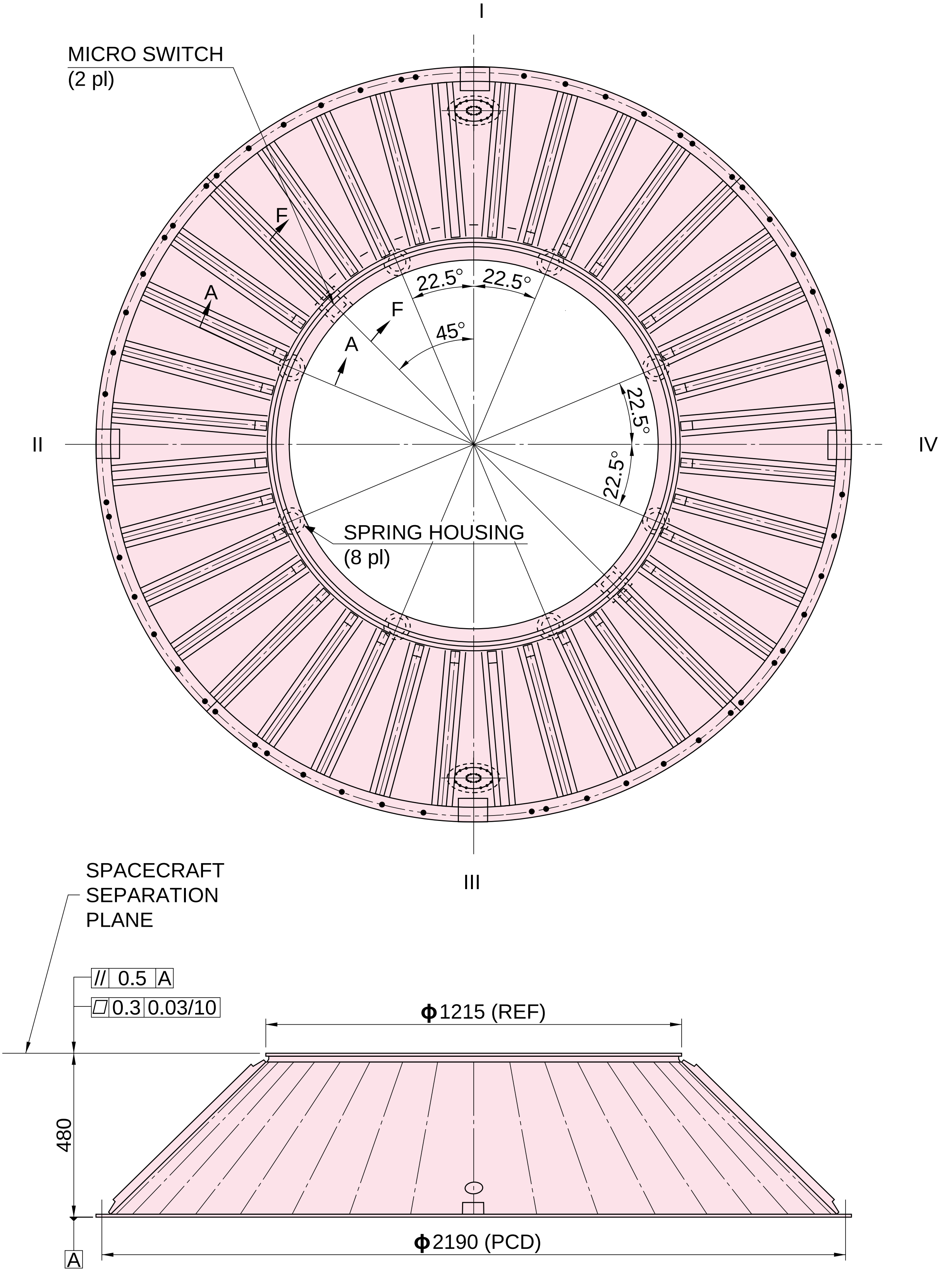


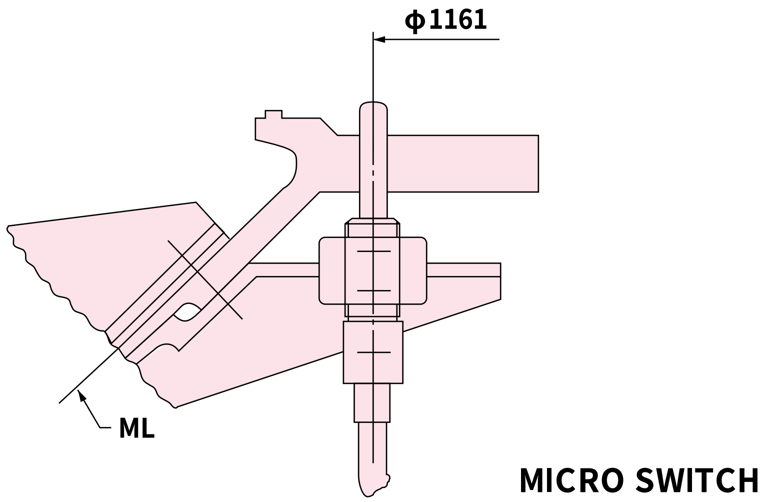
Figure A3.5.1 Photograph of the 1194M adapter



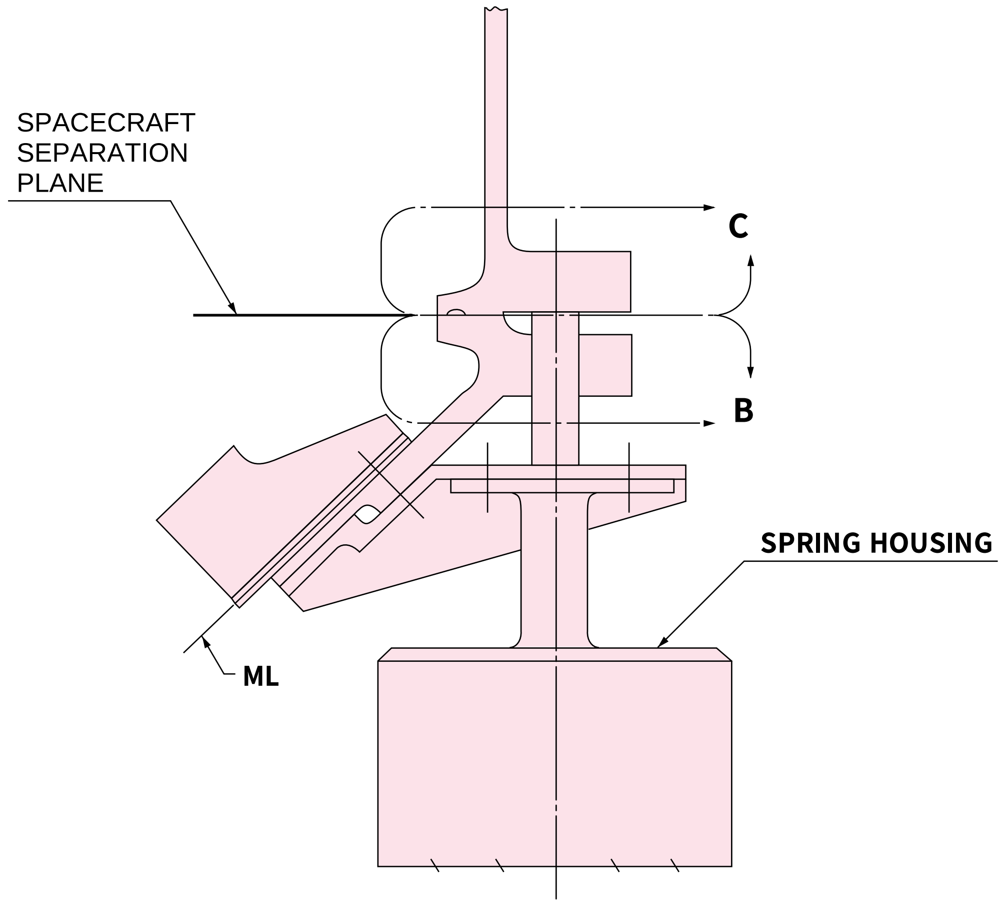
Unit : mm

Figure A3.5.2 General view of 1194M adapter

A



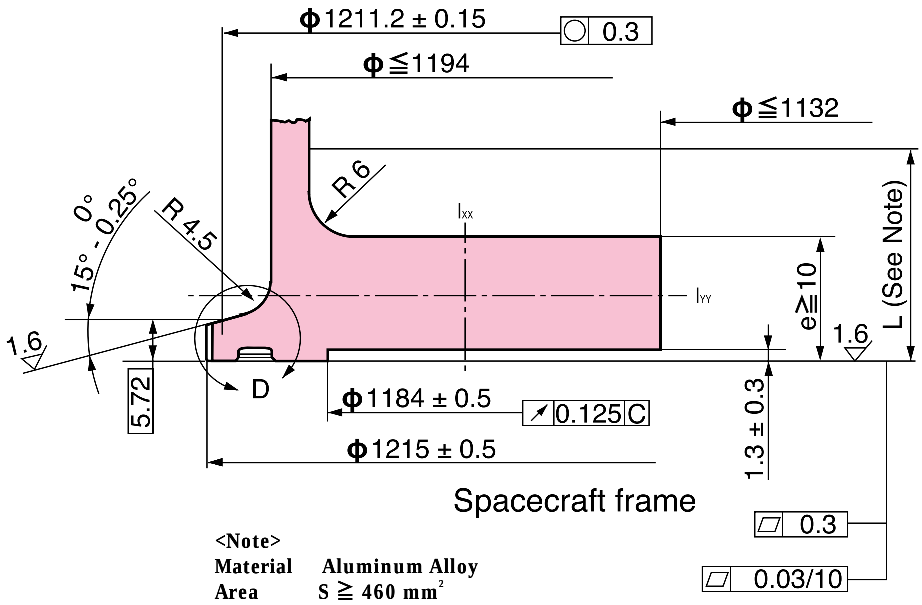
SECTION F–F



SECTION A–A

Unit : mm

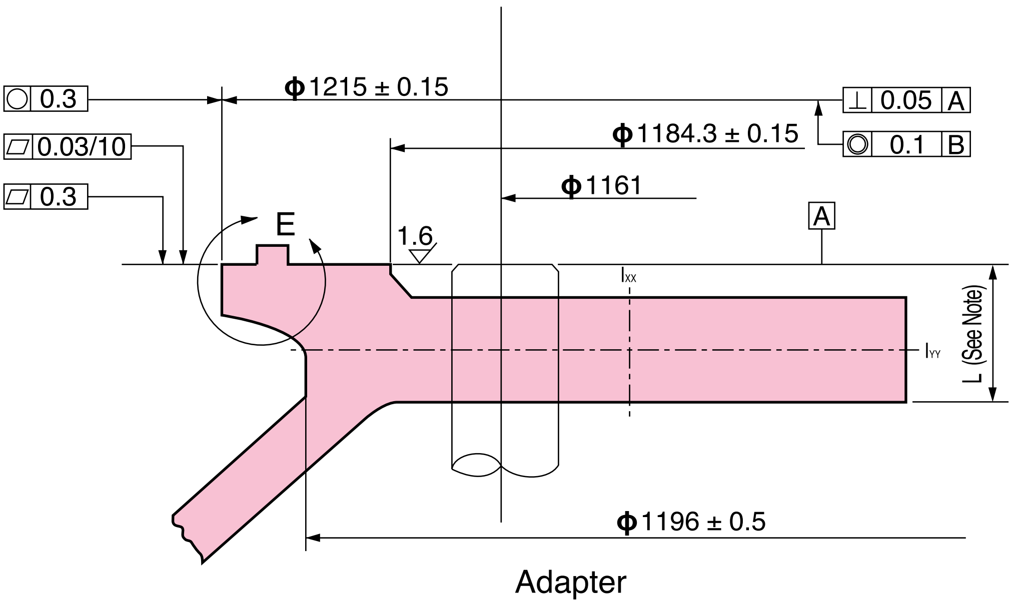
Figure A3.5.3 Details of 1194M adapter #1 (Details of separation plane) | A



<Note>

Material	Aluminum Alloy
Area	$S \geqq 460\ mm^2$
Inertia	$I_{XX} \geqq 5.1 \times 10^4\ mm^4$
	$I_{YY} \geqq 1.2 \times 10^4\ mm^4$
Applicable Length	L = 25 mm

Detail C



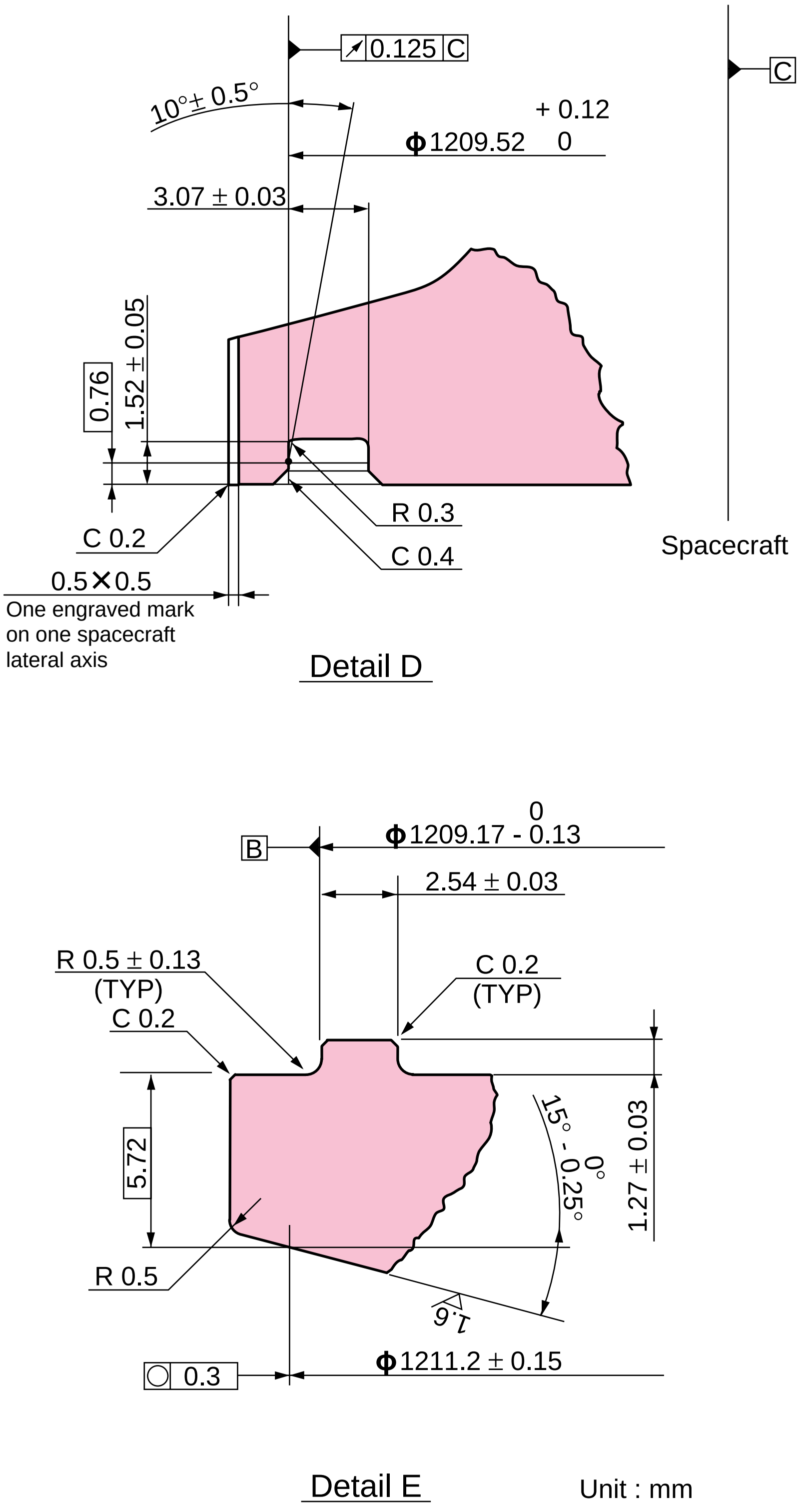
<Note>

Material	Aluminum Alloy
Area	$S = 770\ mm^2$
Inertia	$I_{XX} \geqq 2.0 \times 10^5\ mm^4$
	$I_{YY} \geqq 1.3 \times 10^4\ mm^4$
Applicable Length	L = 18mm

Detail B

Unit : mm

Figure A3.5.4 Details of 1194M adapter #2 (Cross section of frames)



A

A

Figure A3.5.5 Details of 1194M adapter #3 (Details of frames) | A

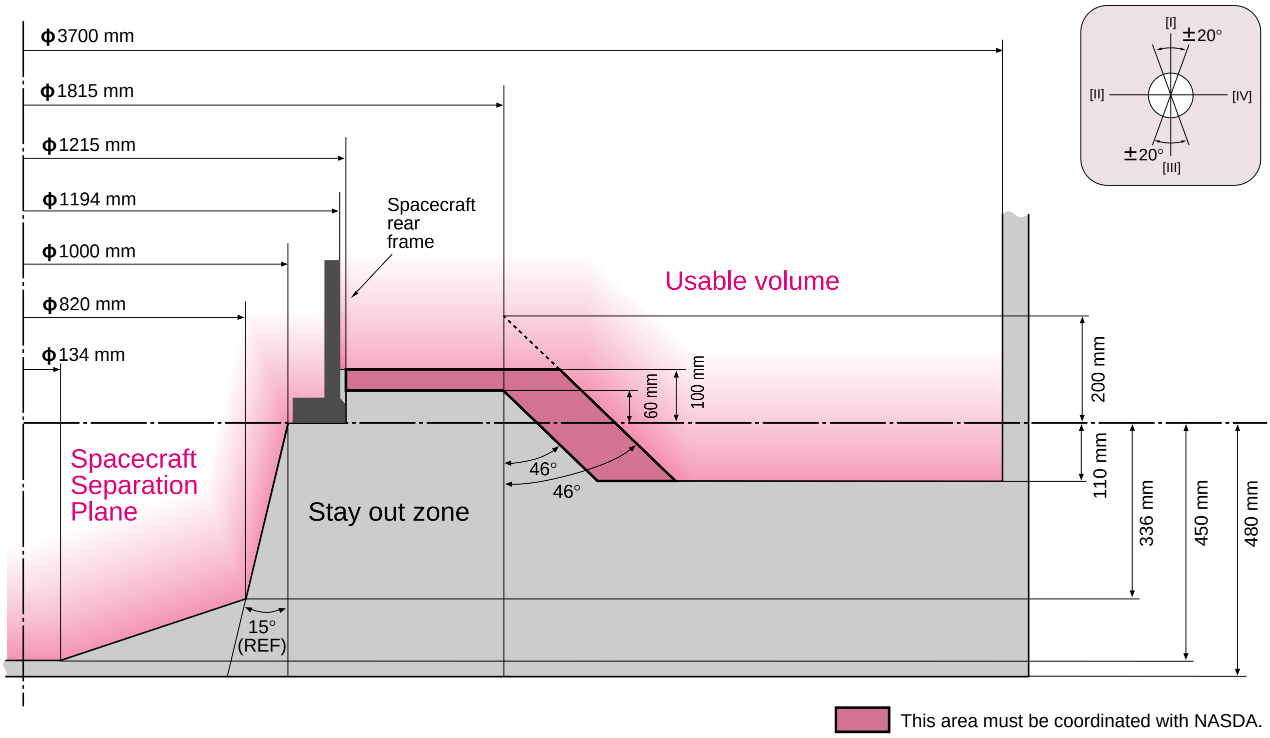


Figure A3.5.6 Stay-out zone around the 1194M adapter (I/III)

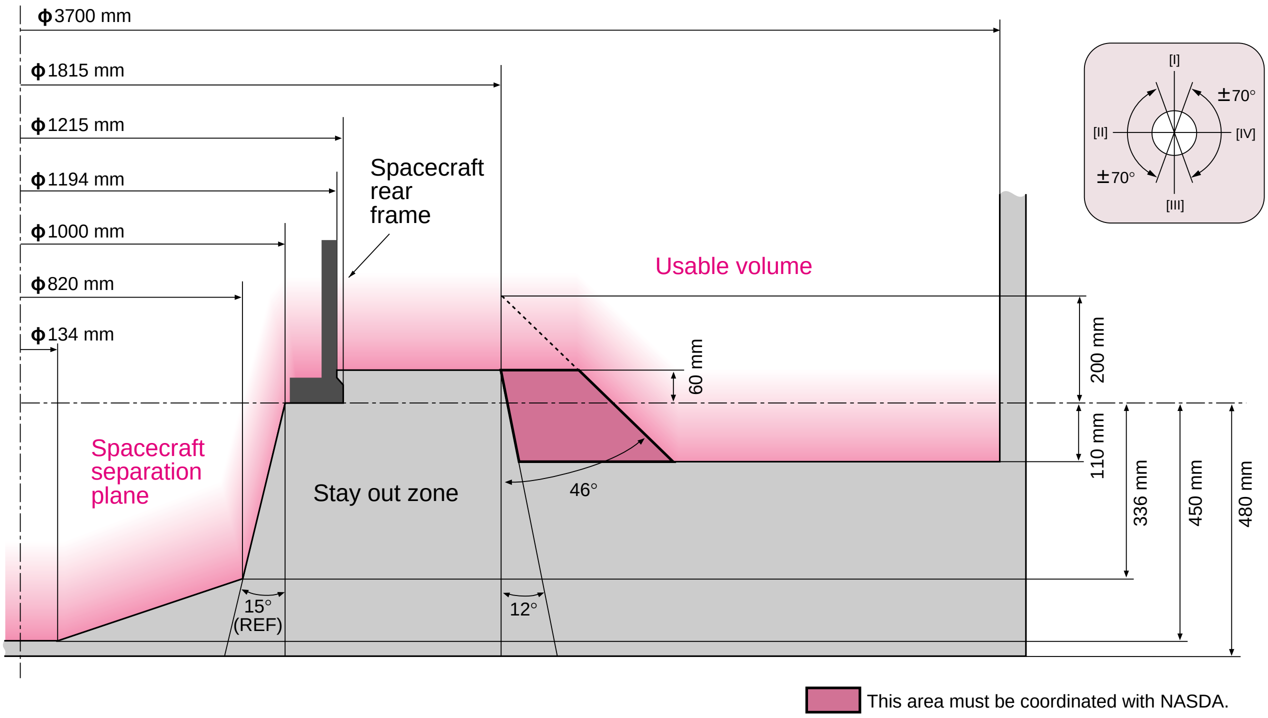


Figure A3.5.7 Stay-out zone around the 1194M adapter (II/IV)

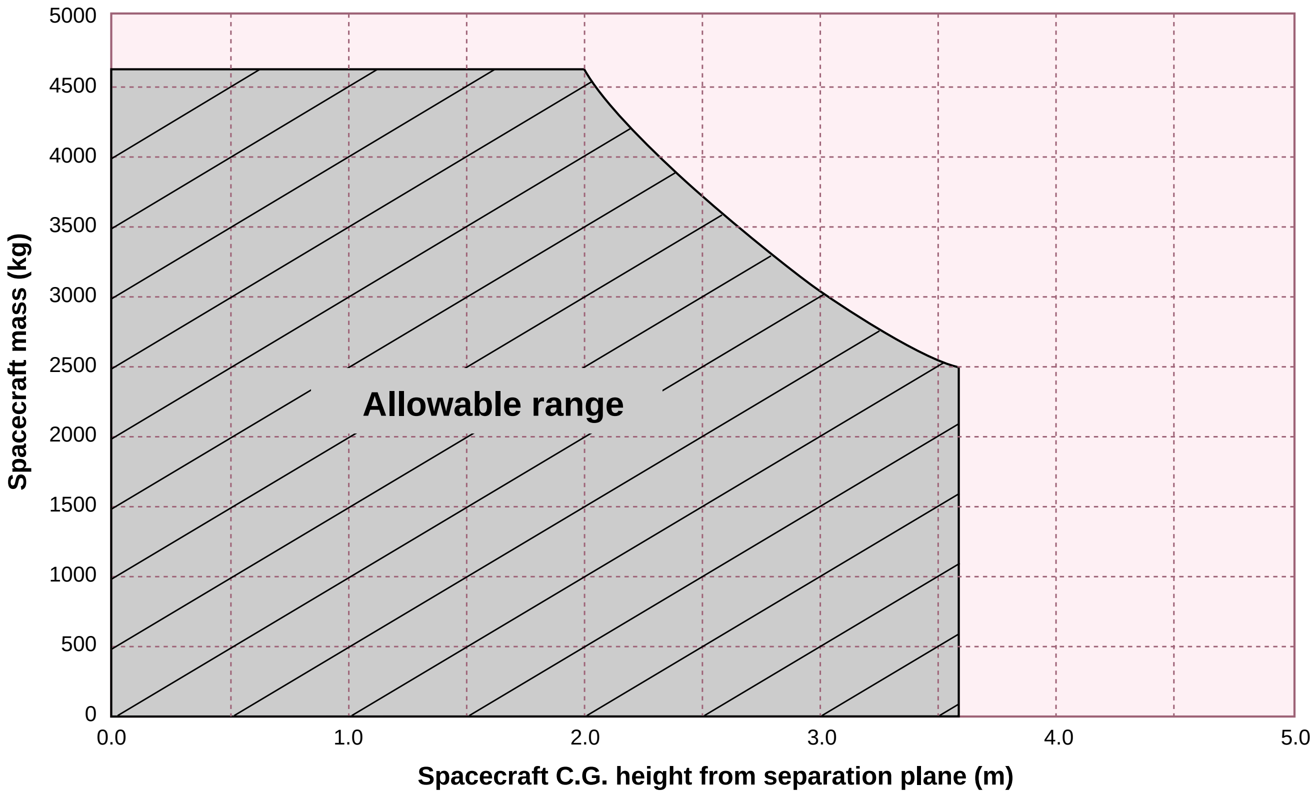


Figure A3.5.8 Limit load of the 1194M adapter

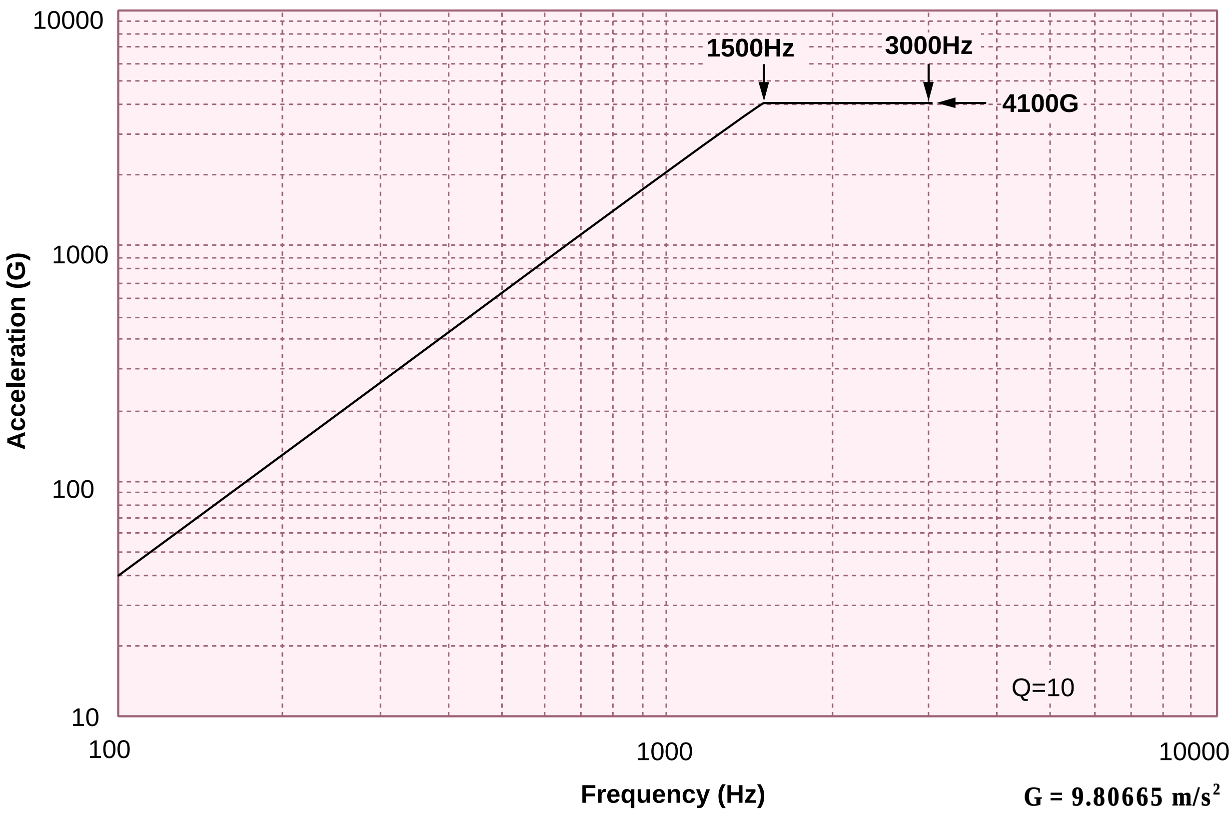


Figure A3.5.9 Spacecraft separation shock spectrum of the 1194M adapter

A
B
A

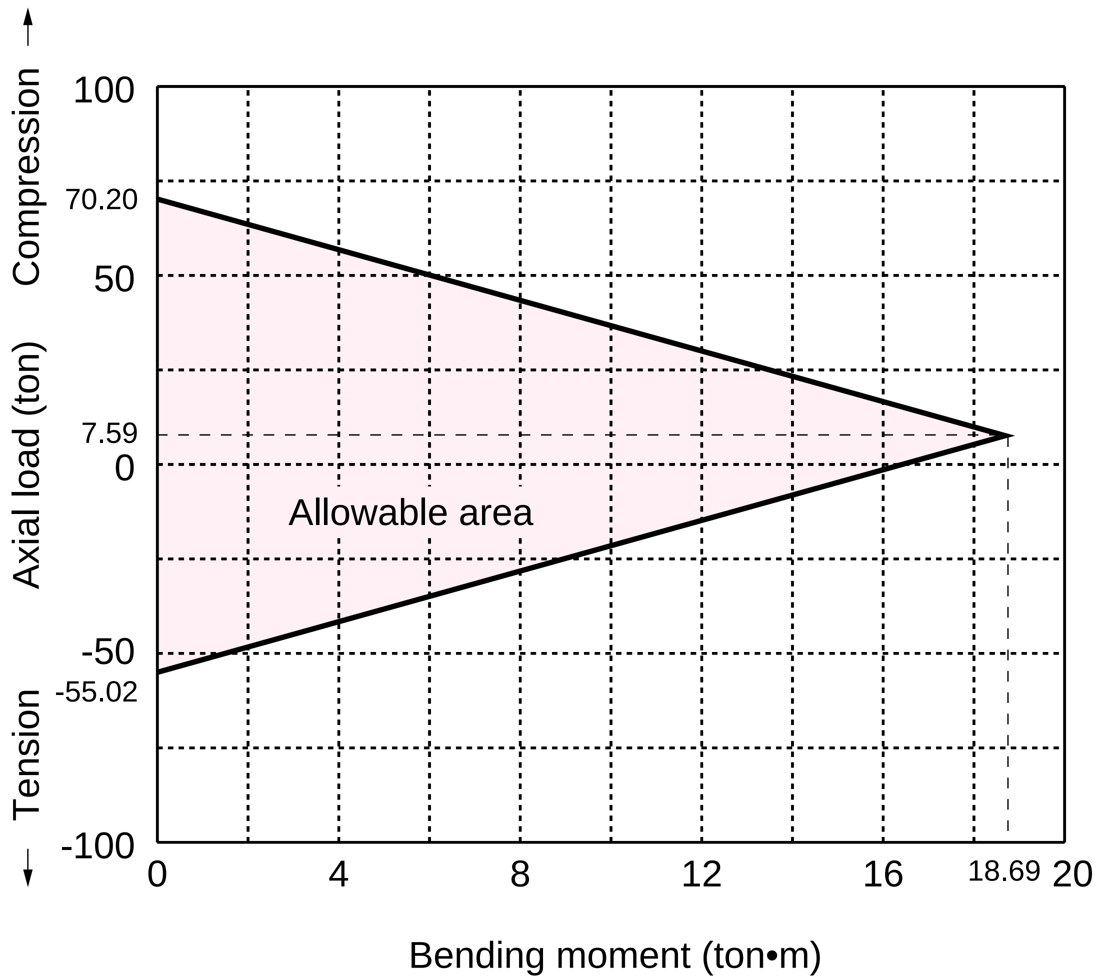


Figure A3.5.10 Limit loads at separation plane of the 1194M adapter

B

APPENDIX 3.6 1666M ADAPTER

The main characteristics are as follows.

(1) Interface diameter : 1,666 mm
(2) Height : 1,000 mm
(3) Material : Aluminum semi-monocoque
(4) Attached system : Clamp bands
(5) Separation system : 4 or 6 springs
(6) Clamp band
  Maximum tension : 36.3 kN
(7) Maximum load per spring : 1670 N
(8) Adapter mass : 100 kg

This adapter has 12 vent holes of 83 mm ϕ (5410.6 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 942.45 mm from the center of the vehicle axis.

Figure A3.6.1 shows the general view of the 1666M adapter.
Figure A3.6.2 to A3.6.4 show details of the 1666M adapter.
Figure A3.6.5 shows the stay-out zone around the 1666M adapter.
Figure A3.6.6 shows the limit load of the 1666M adapter.
Figure A3.6.7 shows the spacecraft separation shock spectrum of the 1666M adapter.
Figure A3.6.8 shows the limit load at separation plane of the 1666M adapter.

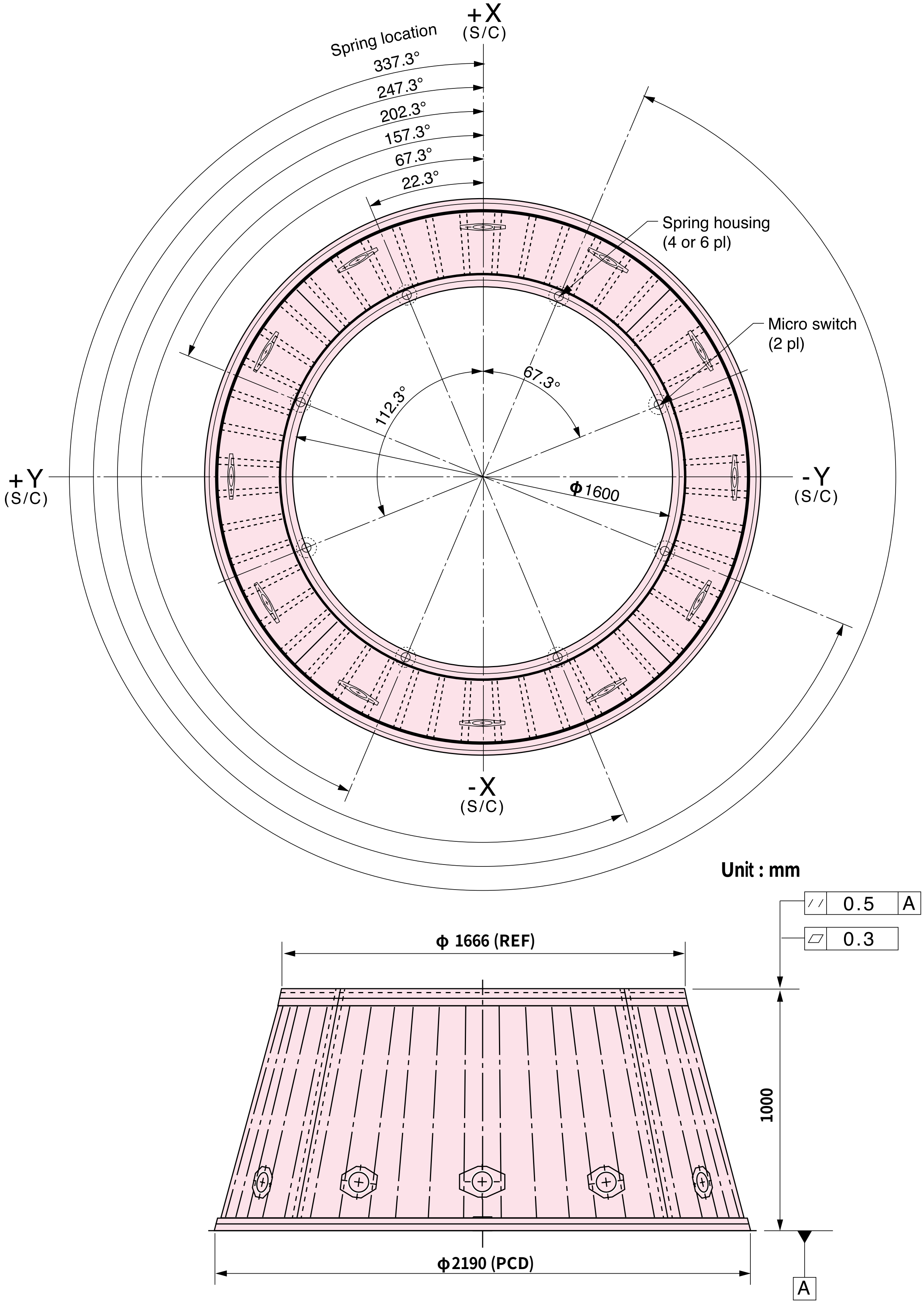


Figure A3.6.1 General view of the 1666M adapter

A
C
A
C
A

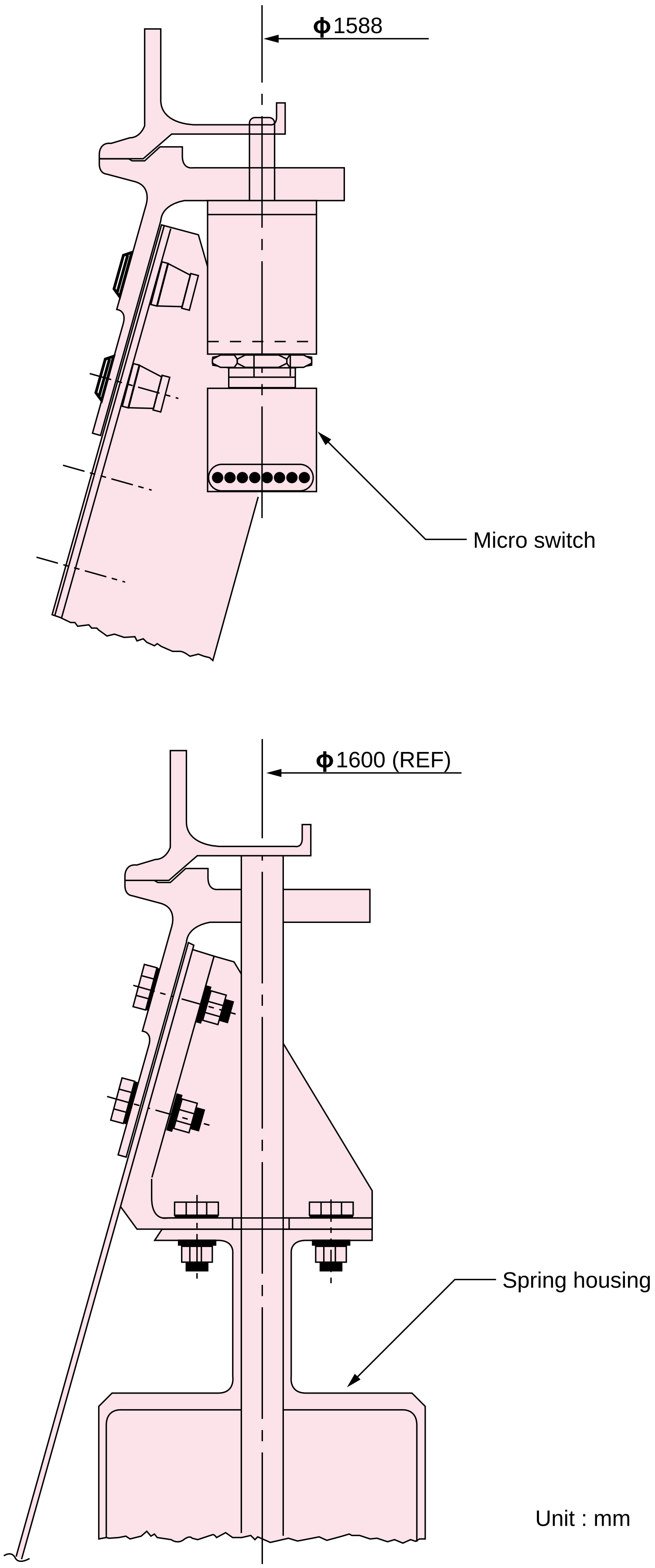


Figure A3.6.2 Details of the 1666M adapter #1

A

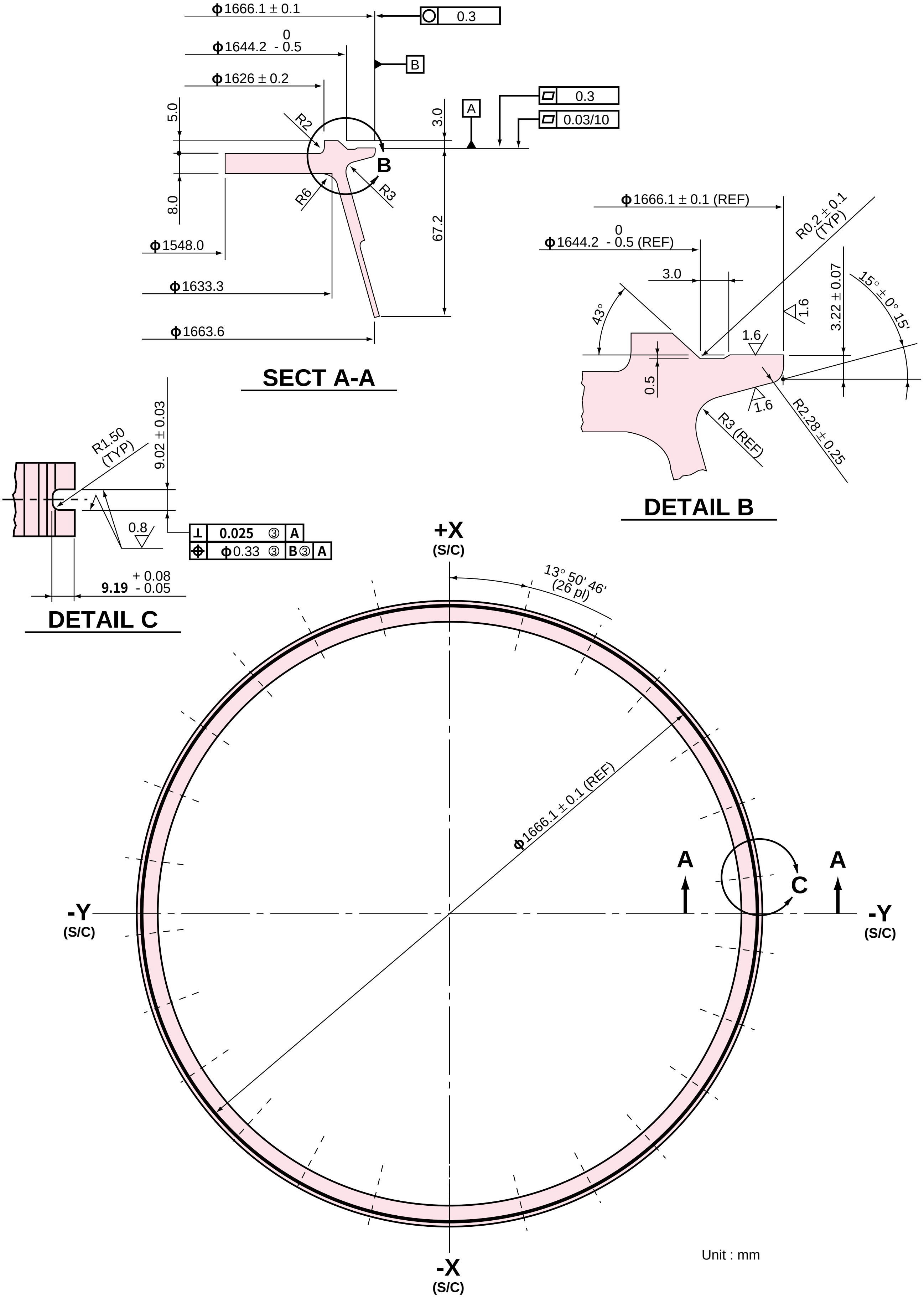


Figure A3.6.3 Details of the 1666M adapter #2

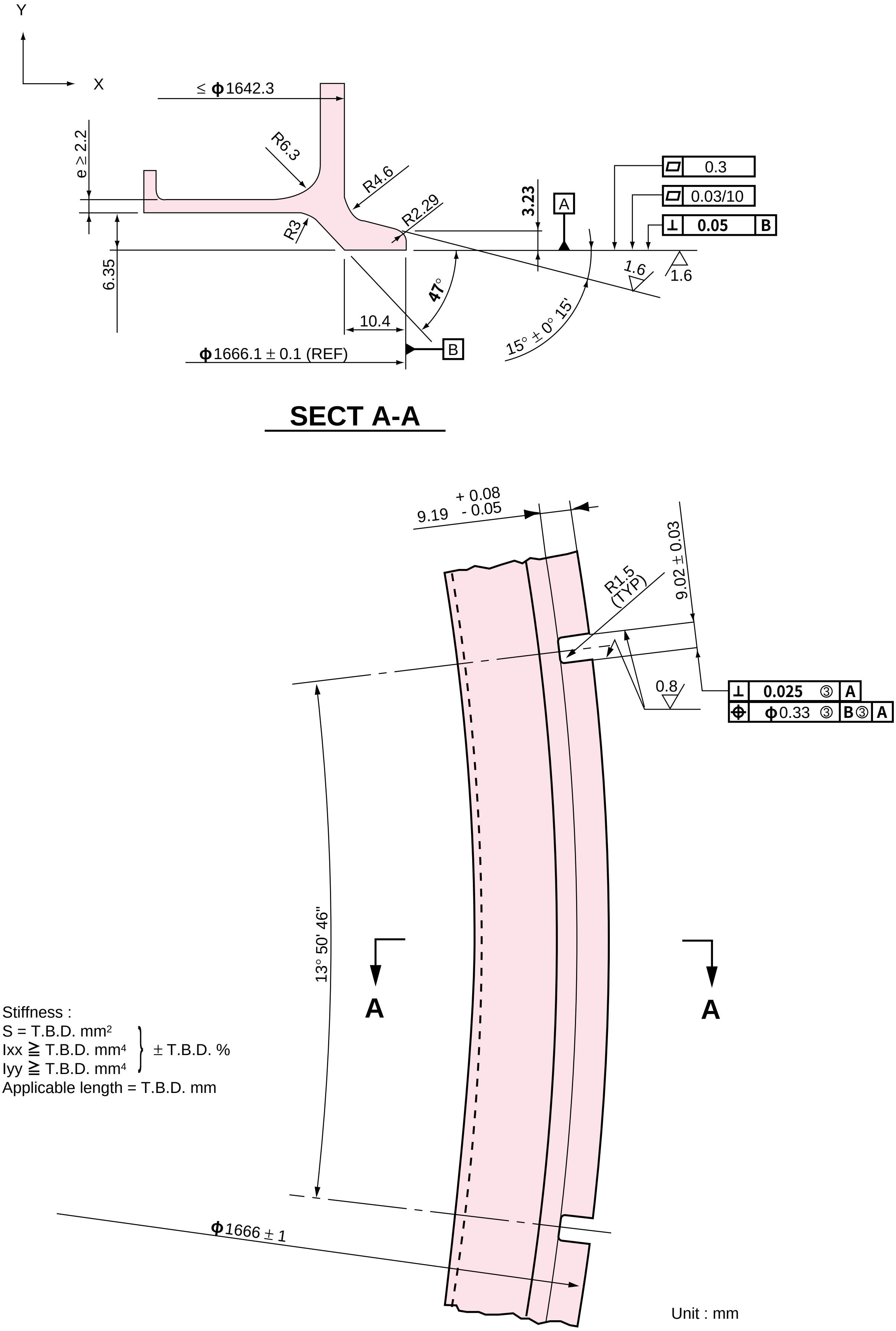


Figure A3.6.4 Details of the 1666M adapter #3 (Spacecraft frame)

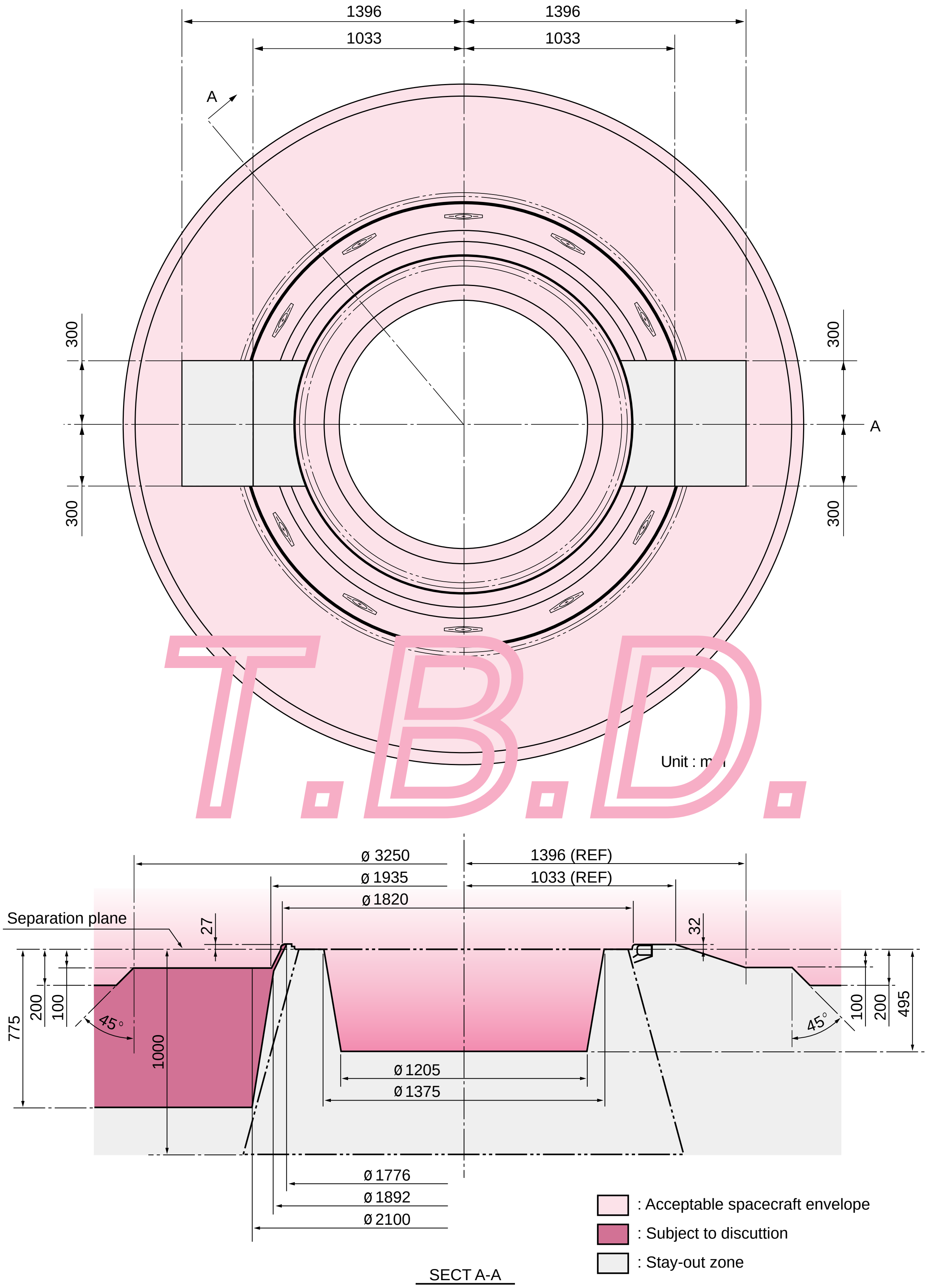


Figure A3.6.5 Stay-out zone around the 1666M adapter

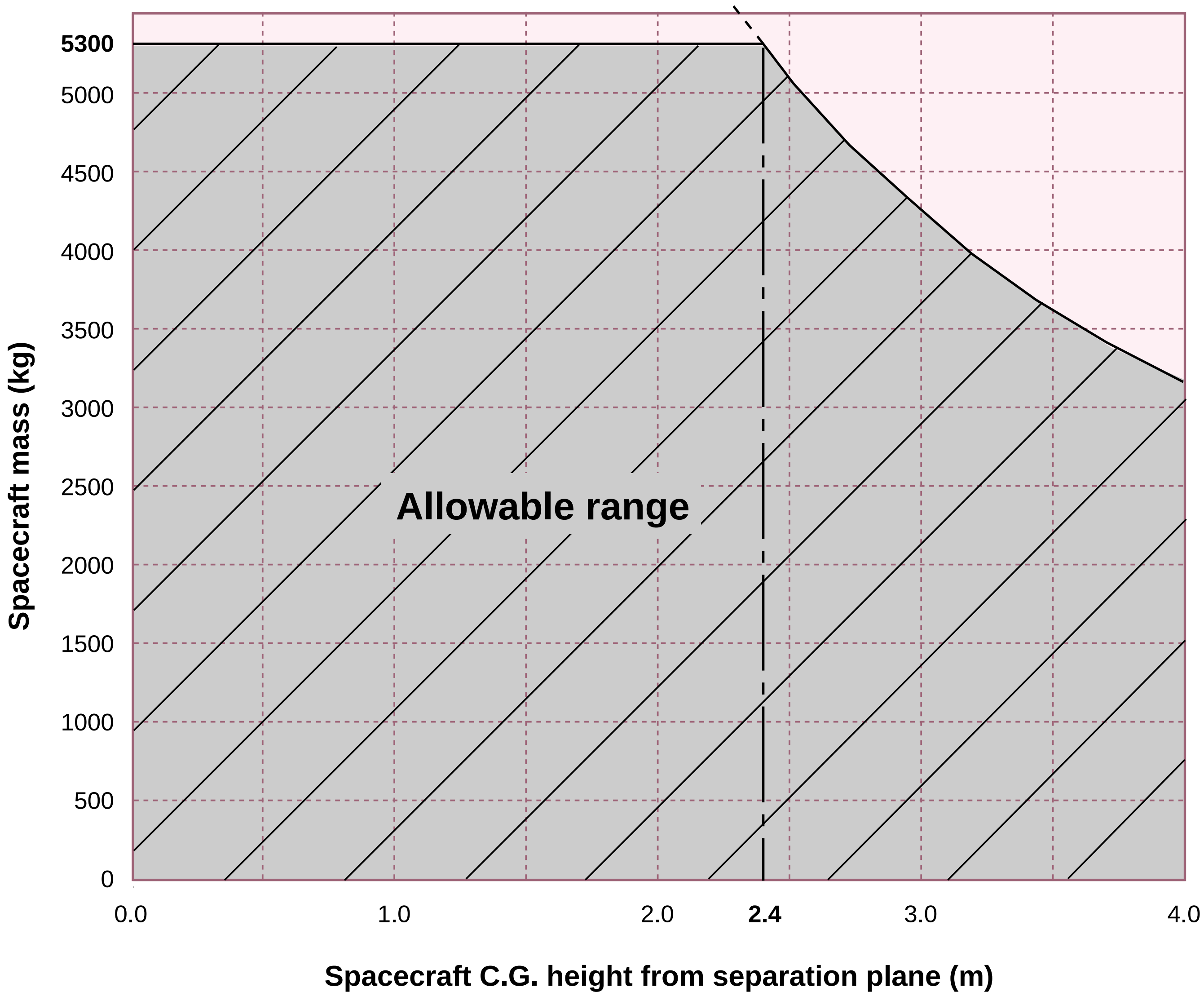


Figure A3.6.6 Limit load of the 1666M adapter

A

T.B.D.

Figure A3.6.7 Spacecraft separation shock spectrum of the 1666M adapter

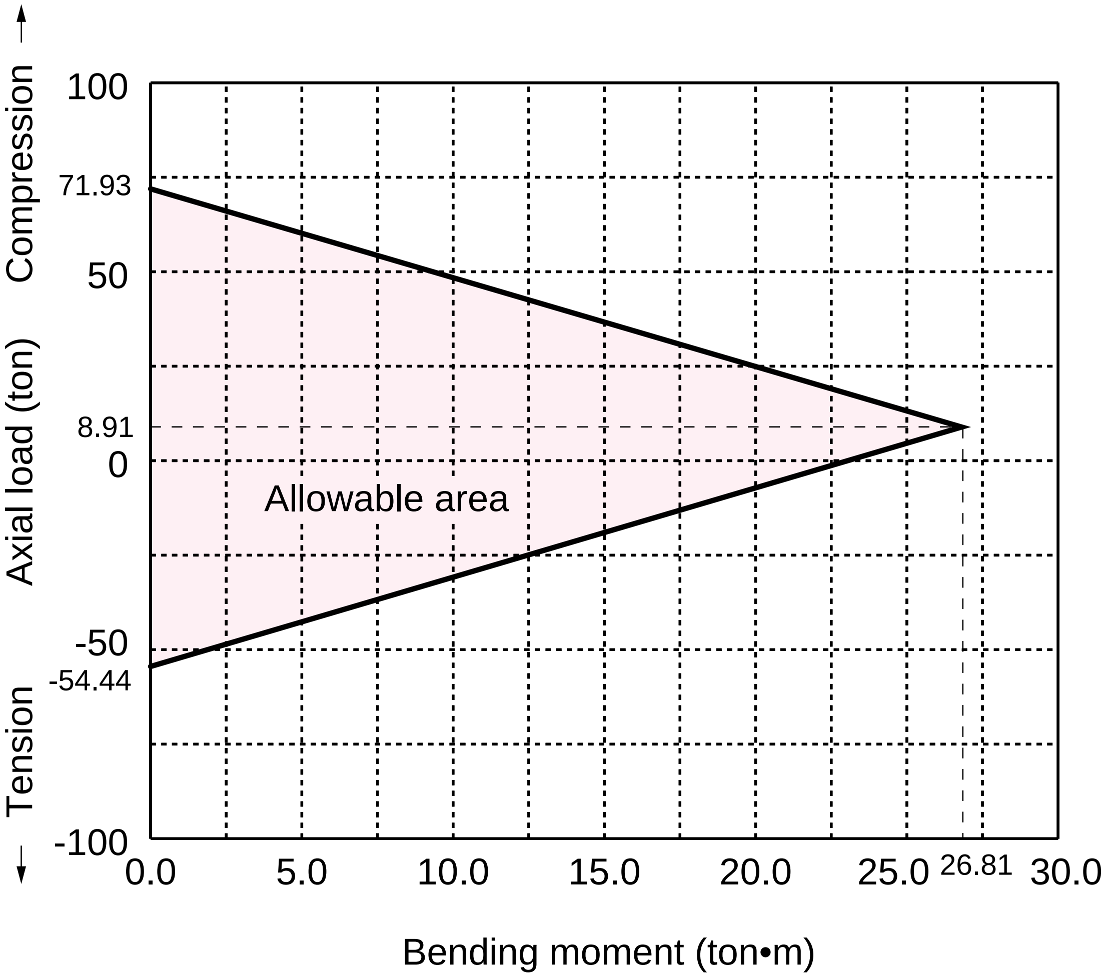


Figure A3.6.8 Limit loads at separation plane of the 1666M adapter

B

APPENDIX 3.7 1666S ADAPTER

The main characteristics are as follows.

(1) Interface diameter : 1,664 mm (PCD)
(2) Height : 1,000 mm
(3) Material : Aluminum semi-monocoque
(4) Attached system : 4 places 19.1 mm (3/4") Bolts and Separation nuts
(5) Separation system : 4 springs
(6) Separation nuts
Maximum tension : 178 kN / Bolt
(7) Maximum load per spring : 2390 N
(8) Adapter mass : 100 kg

This adapter has 8 vent holes (100 x 155 mm 4 places, 50 x 100 mm x 4 places) 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 928.9 mm from the center of the vehicle axis.

Figure A3.7.1 shows the general view of the 1666S adapter.
Figure A3.7.2 to A3.7.4 show details of the 1666S adapter.
Figure A3.7.5 shows the stay-out zone around the 1666S adapter.
Figure A3.7.6 shows the limit load of the 1666S adapter.
Figure A3.7.7 shows the spacecraft separation shock spectrum of the 1666S adapter.
Figure A3.7.8 shows the limit load at separation plane of the 1666S adapter.

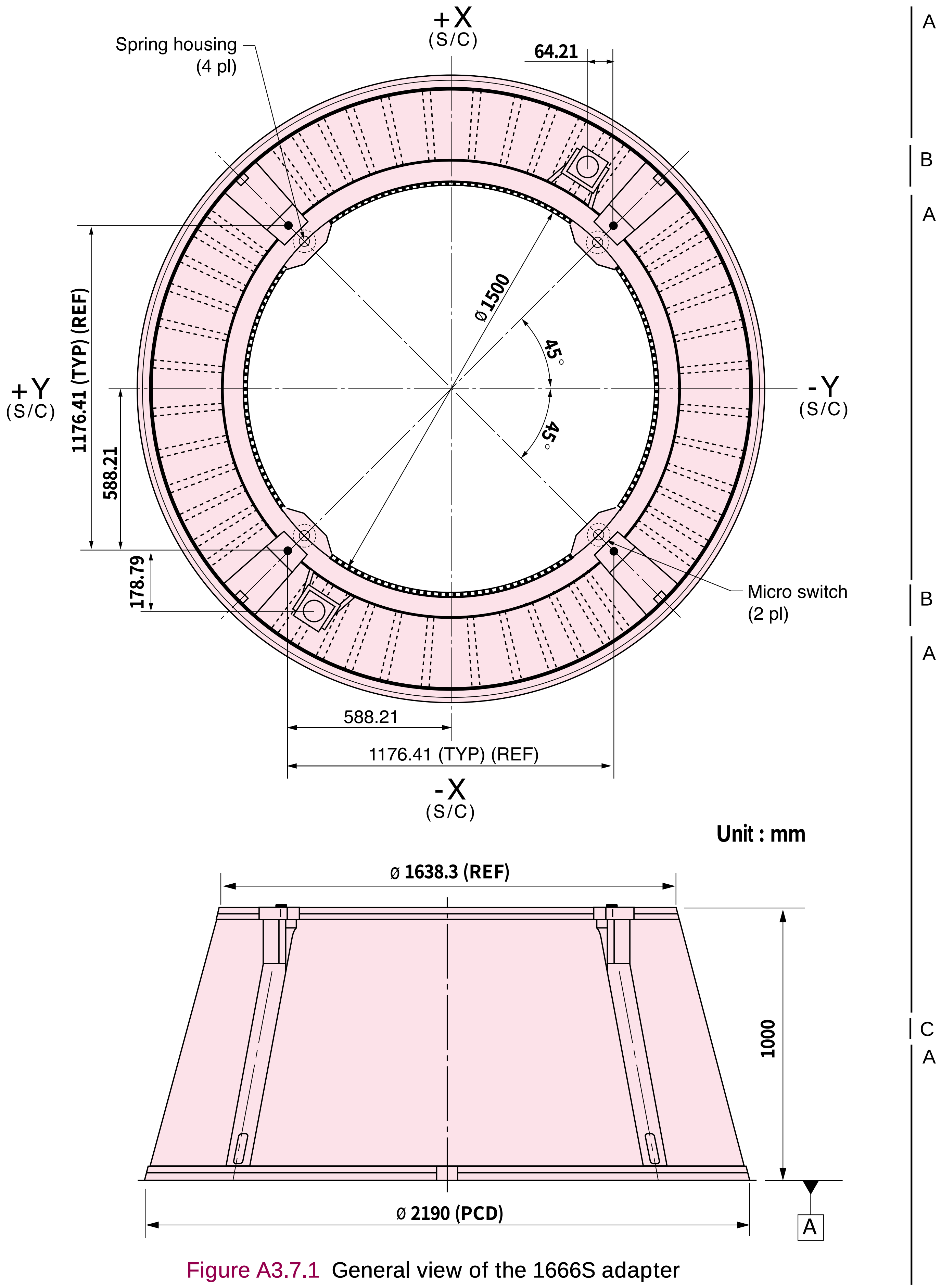


Figure A3.7.1 General view of the 1666S adapter

A

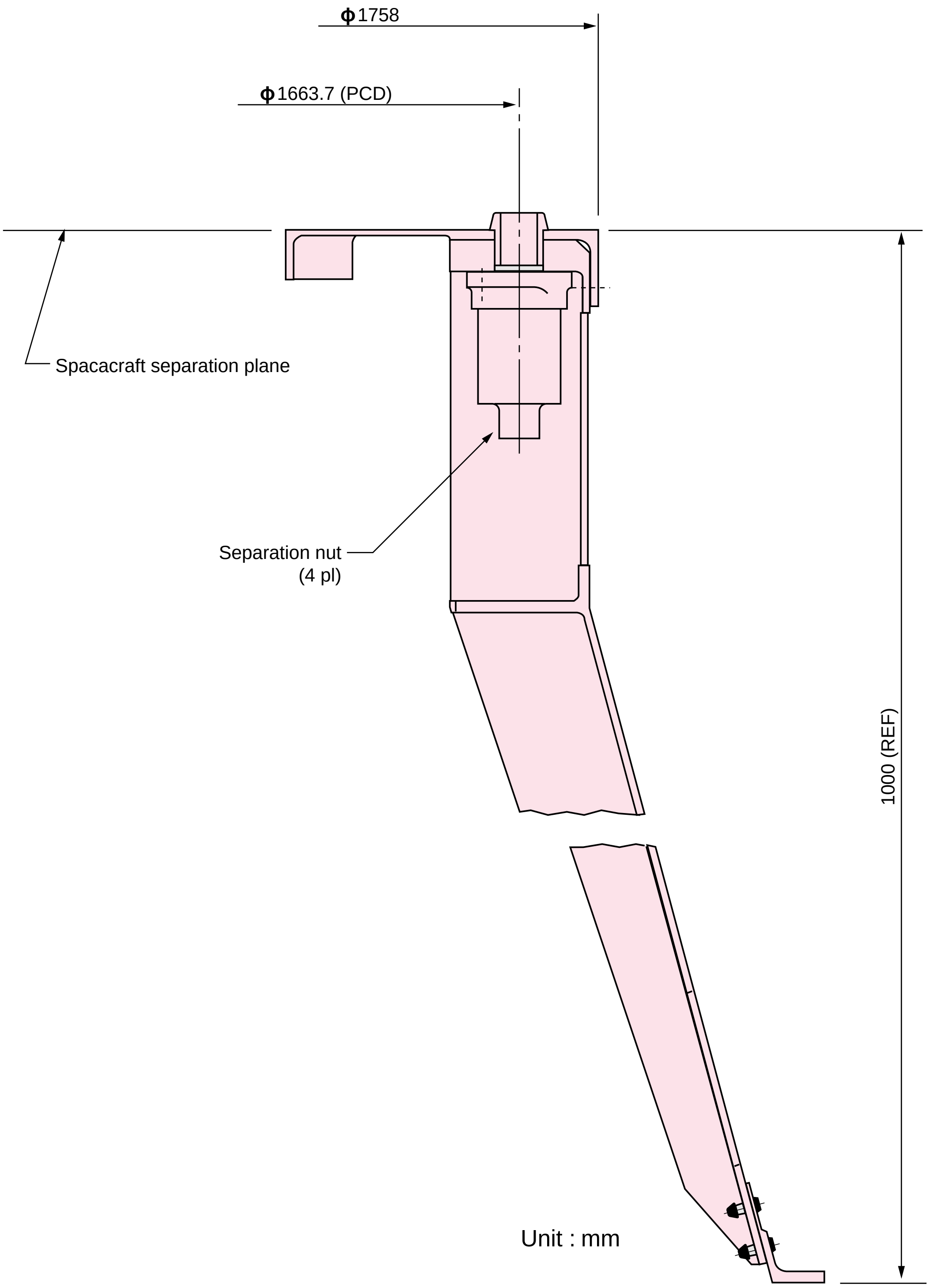


Figure A3.7.2 Details of the 1666S adapter #1

A

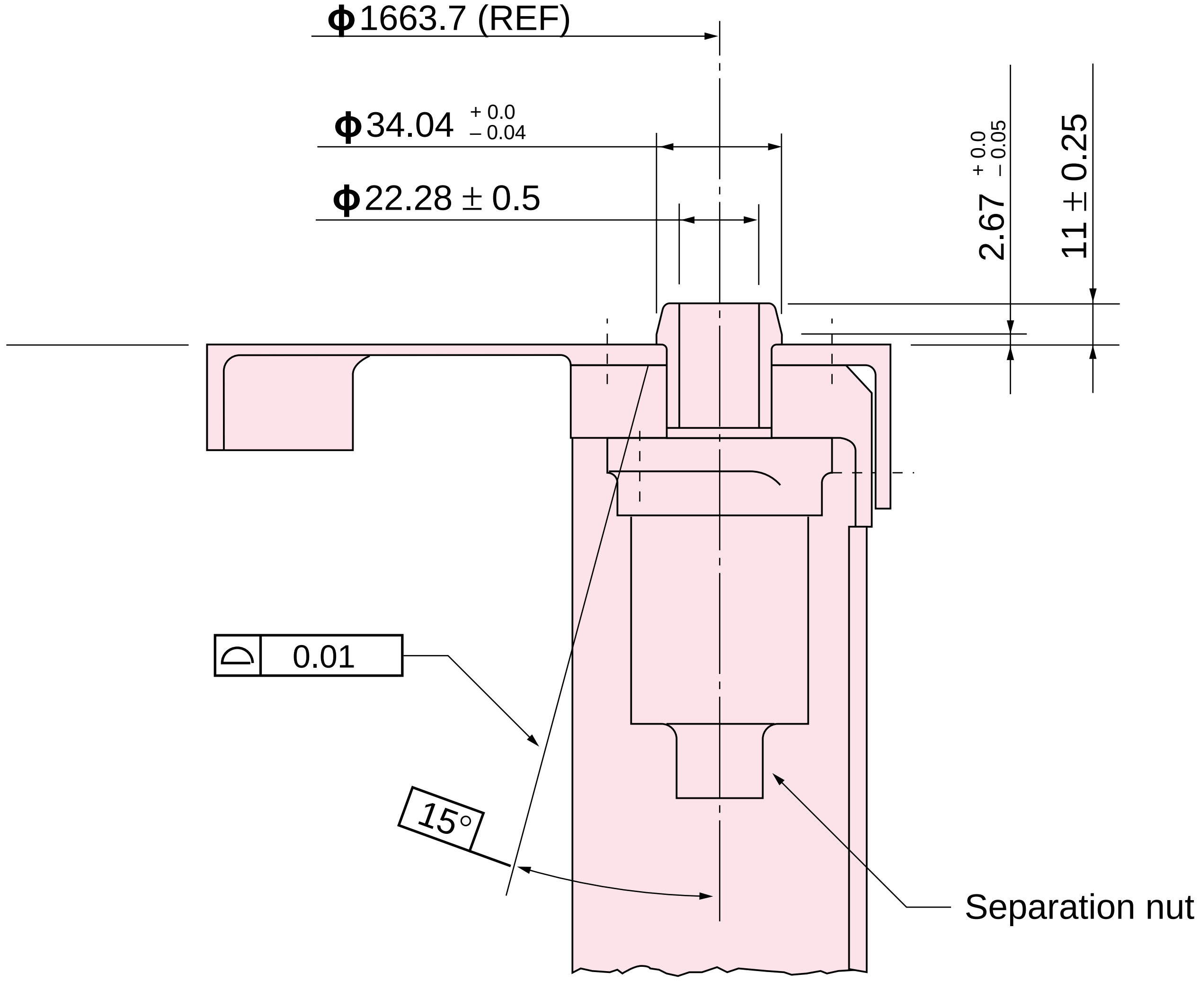


Figure A3.7.3 Details of the 1666S adapter #2

B

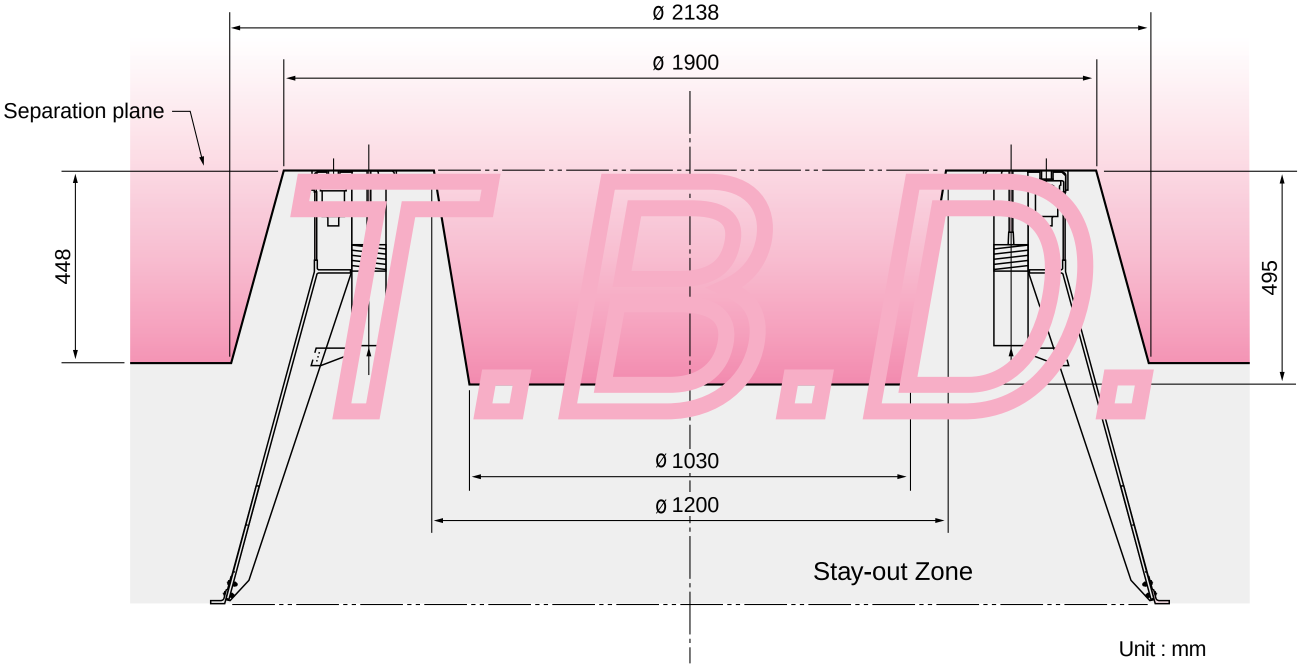


Figure A3.7.4 Stay-out zone around the 1666S adapter #1

B

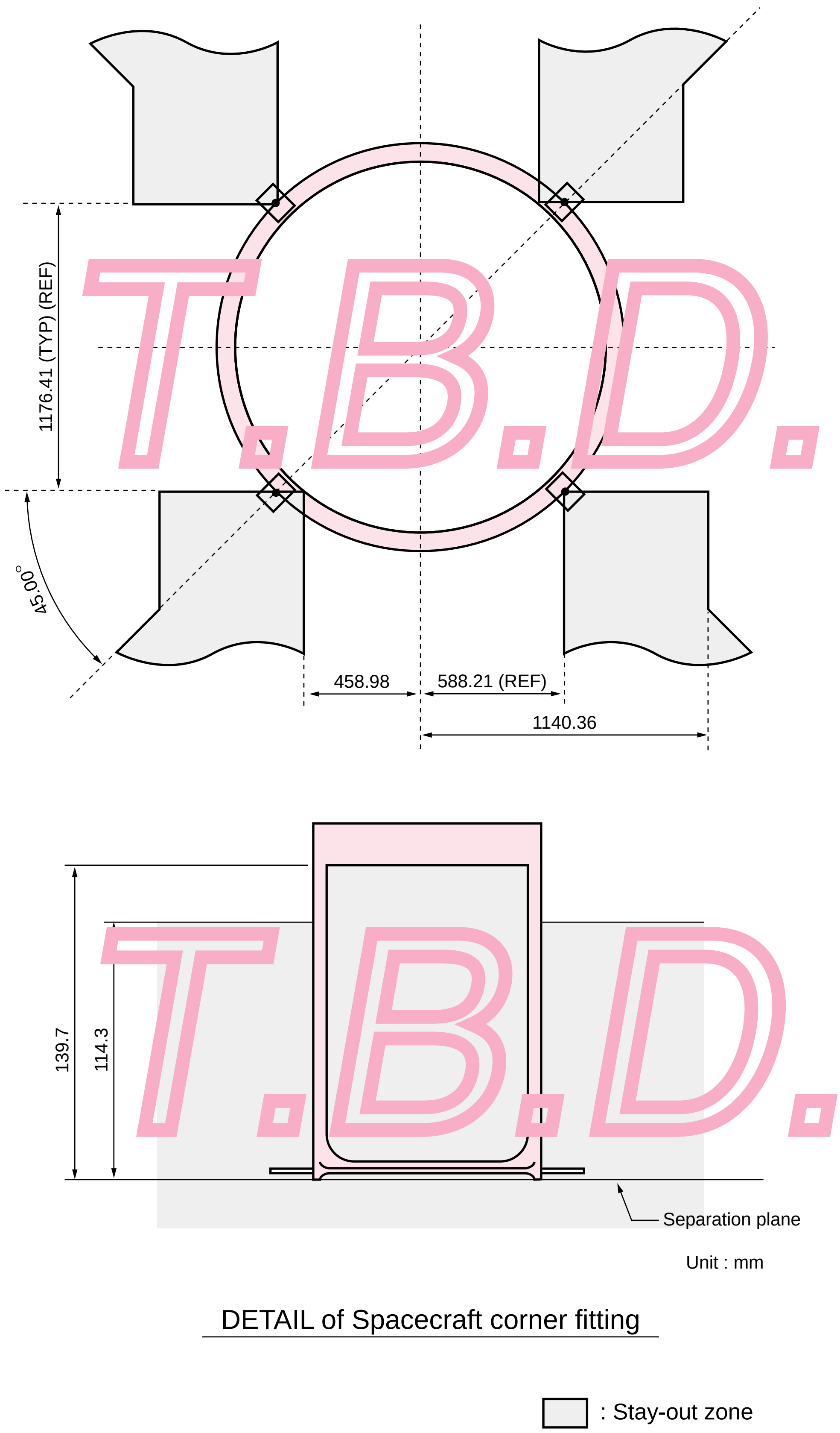
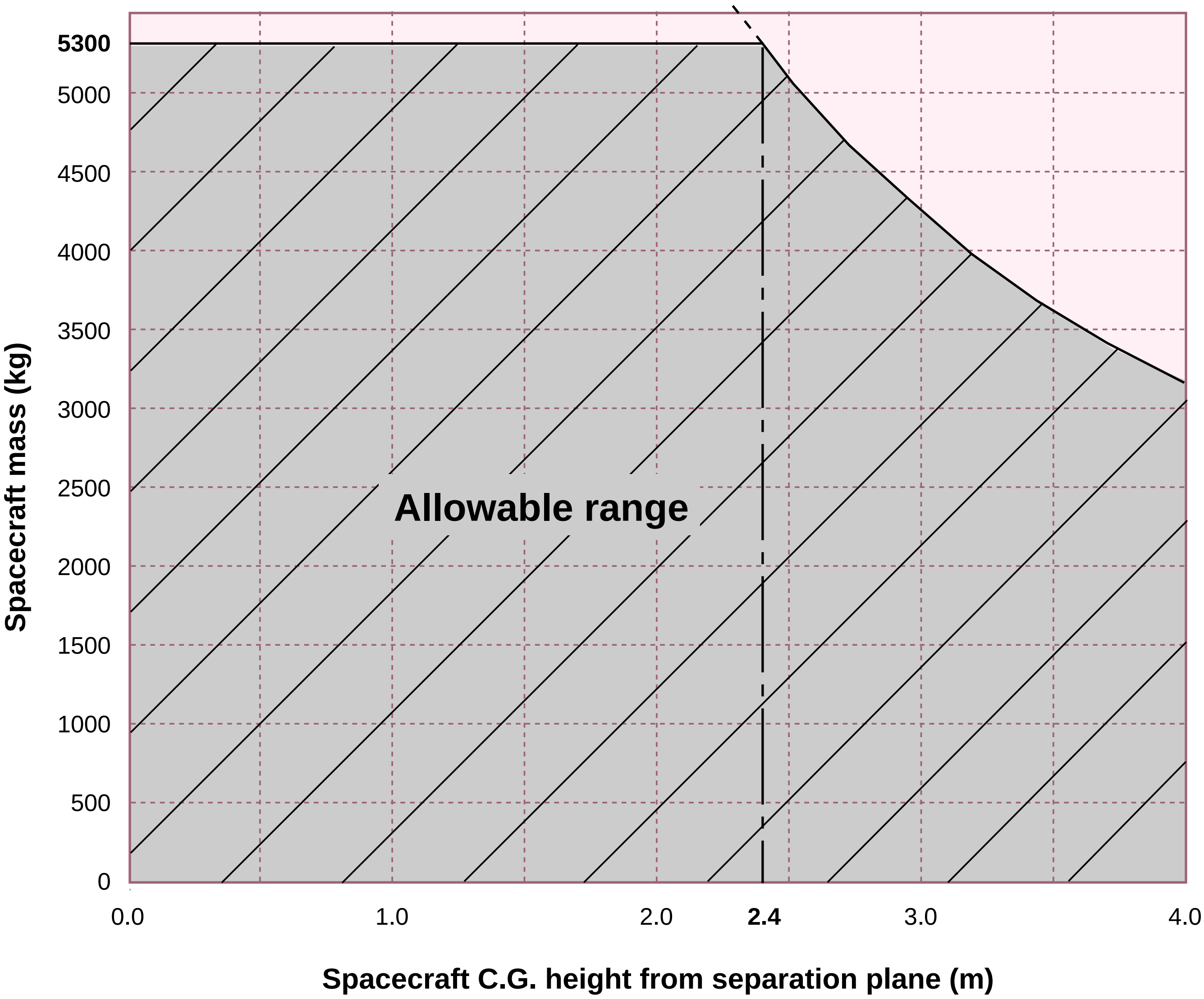


Figure A3.7.5 Stay-out zone around the 1666S adapter #2



A

Figure A3.7.6 Limit load of the 1666S adapter

A

T.B.D.

Figure A3.7.7 Spacecraft separation shock spectrum of the 1666S adapter

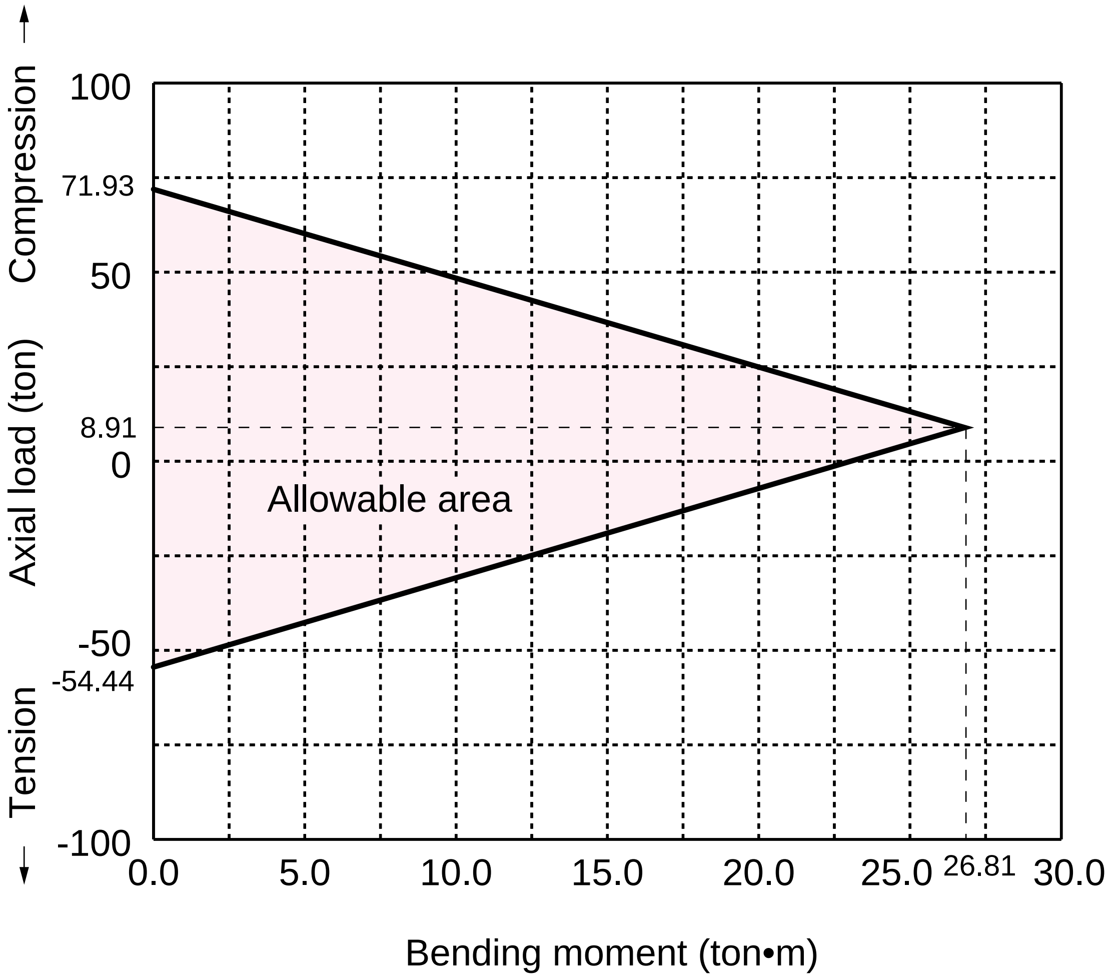


Figure A3.7.8 Limit loads at separation plane of the 1666S adapter

B

APPENDIX 3.8 2360S ADAPTER A

The main characteristics are as follows. C

(1)	Interface diameter	: 2,360 mm (PCD)
(2)	Height	: 300 mm
(3)	Material	: co-cured graphite epoxy
(4)	Attached system	: 8 places 15.9 mm (5/8") Bolts and Separation nuts
(5)	Separation system	: None
(6)	Separation nuts	
	Maximum tension	: 118 kN / Bolt
(7)	Maximum load per spring	: None
(8)	Adapter mass	: 40 kg

Figure A3.8.1 shows the photograph of the 2360S adapter. A
Figure A3.8.2 shows a general view of the 2360S adapter.
Figures A3.8.3 to A3.8.5 show details of the 2360S adapter.

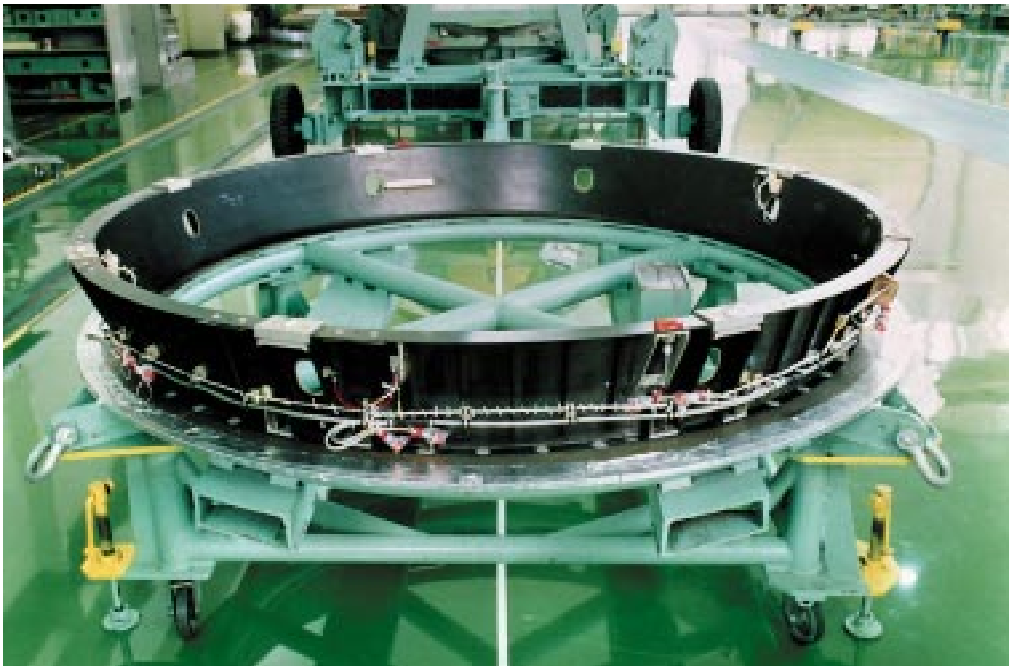
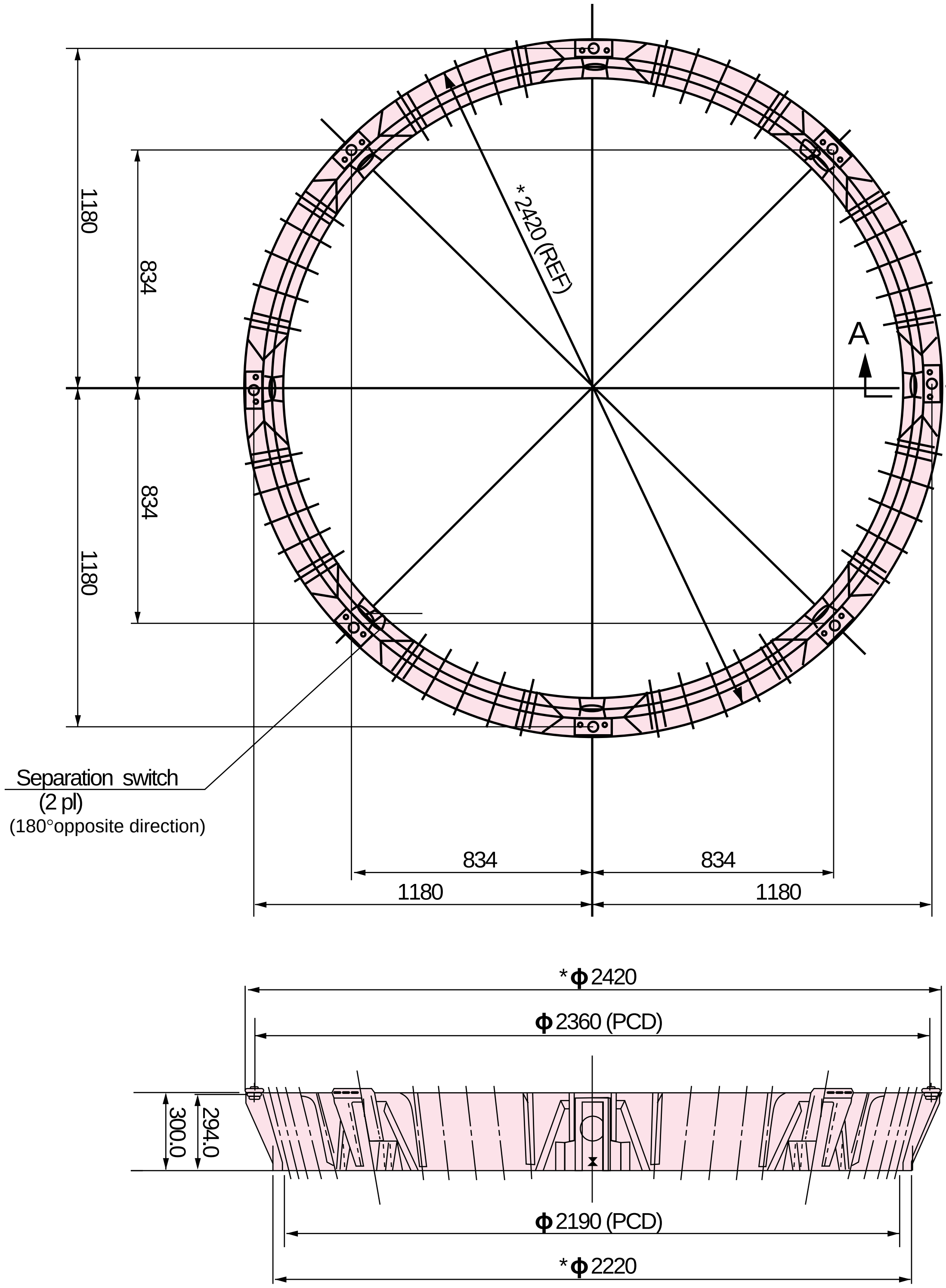


Figure A3.8.1 Photograph of the 2360S adapter A



* Outside diameter

Unit : mm

Figure A3.8.2 General view of 2360S adapter

| A

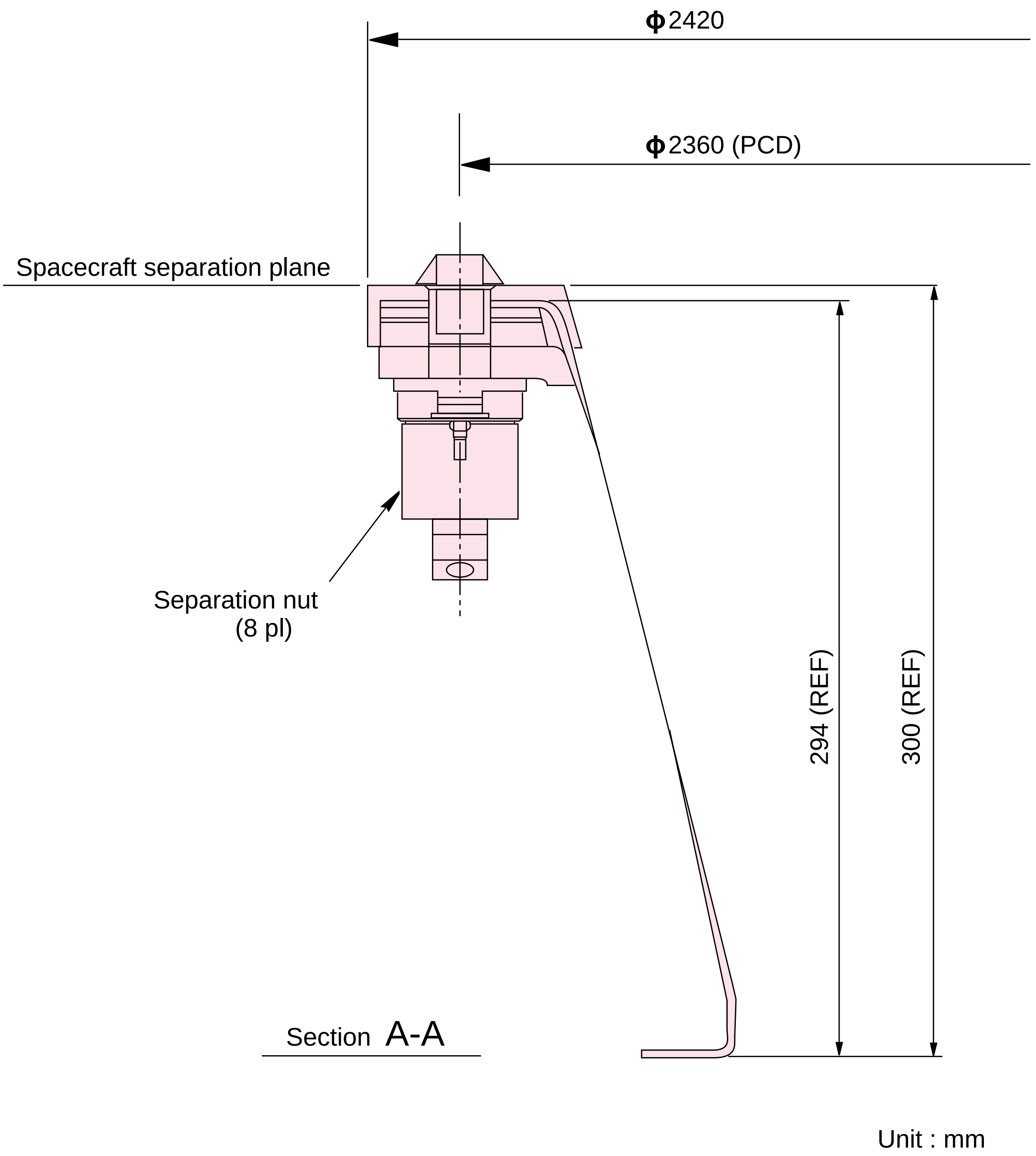
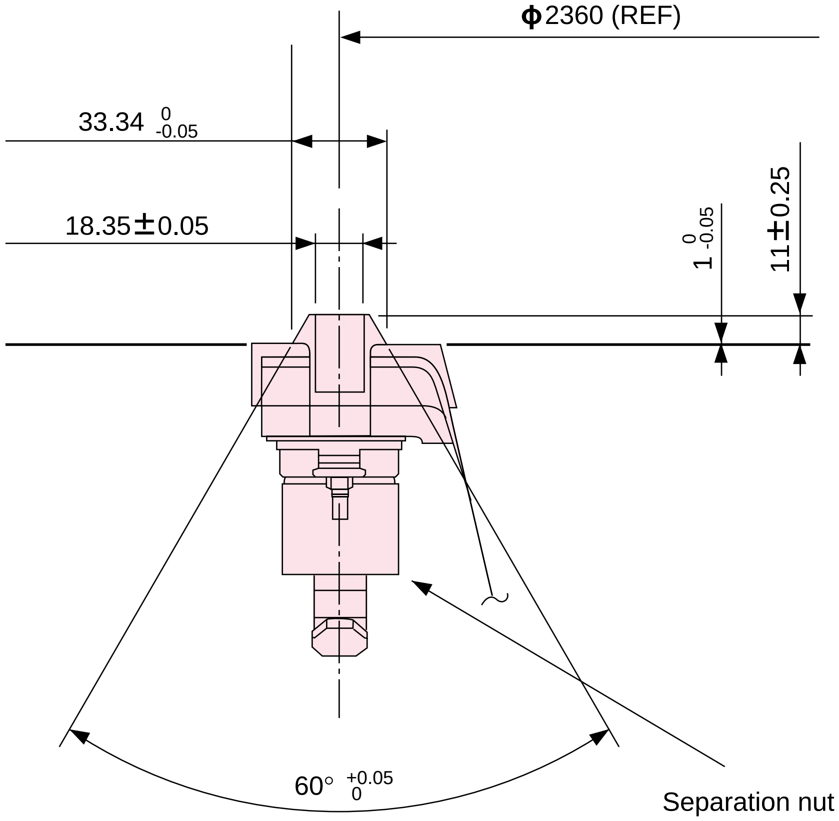


Figure A3.8.3 Details of 2360S adapter #1

A



Details of separation nut
(Adapter side)

Unit: mm

Figure A3.8.4 Details of 2360S adapter #2

| A

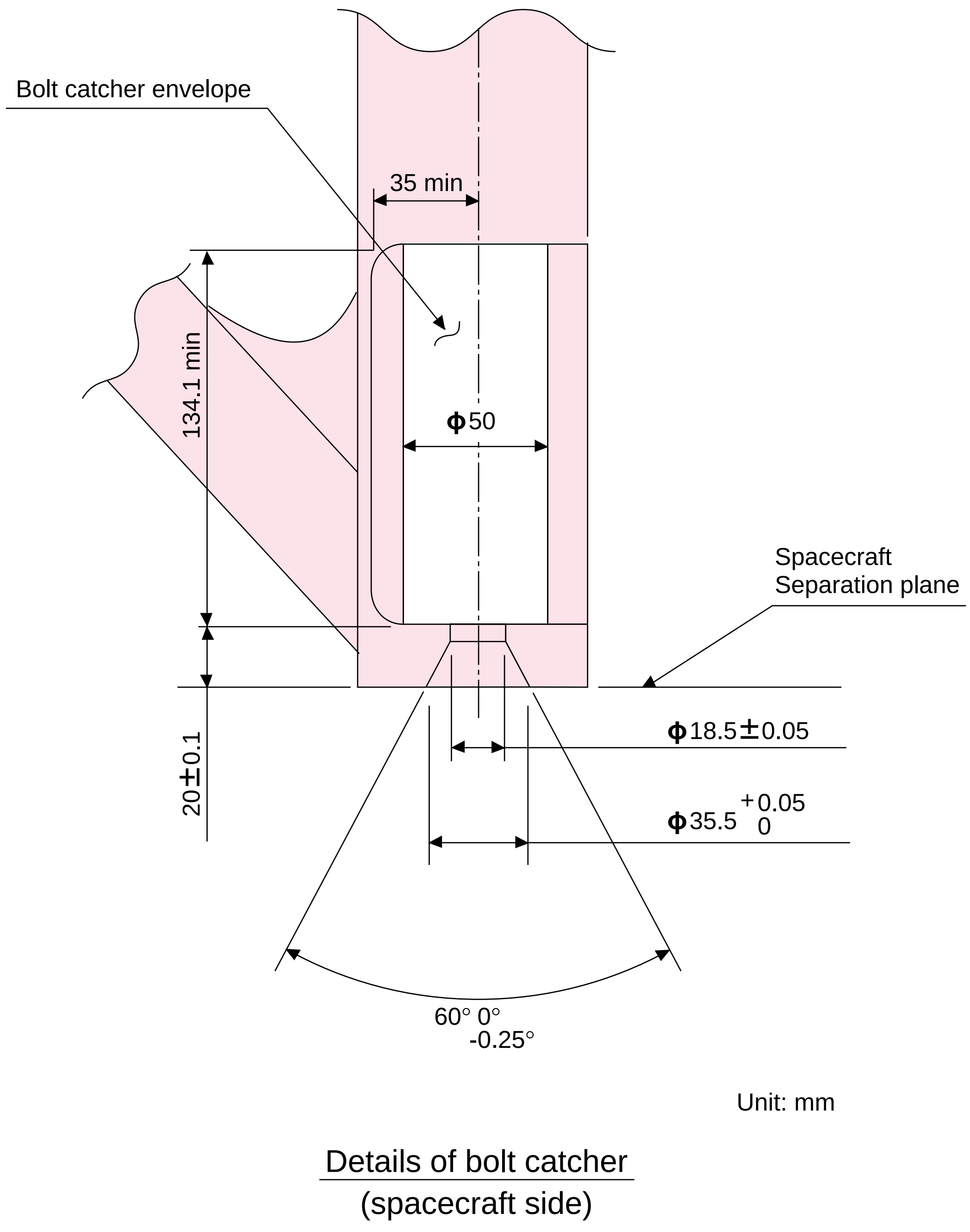


Figure A3.8.5 Details of 2360S adapter #3 | A

APPENDIX 3.9 3470S ADAPTER A

The main characteristics are as follows. C

(1) Interface diameter : 3,472 mm (PCD)
(2) Height : 1,089 mm
(3) Material : Aluminum semi-monocoque
(4) Attached system : 4 places 15.9 mm (5/8") Bolts and Separation nuts
(5) Separation system : None
(6) Separation nuts
    Maximum tension : 147 kN / Bolt
(7) Maximum load per spring : None
(8) Adapter mass : 350 kg

Figure A3.9.1 shows a general view of the 3470S adapter. A
Figures A3.9.2 to A3.9.4 show details of the 3470S adapter.

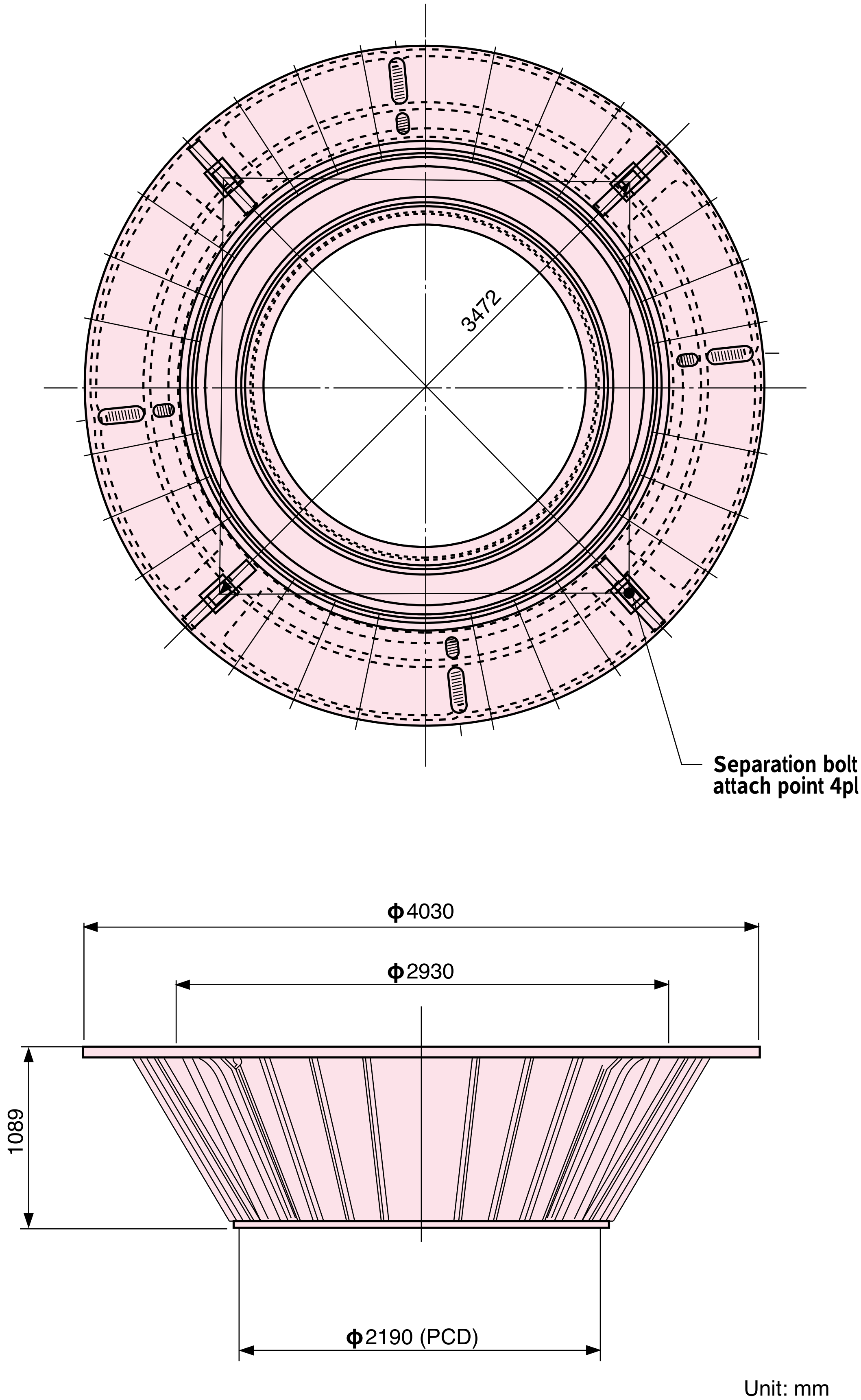


Figure A3.9.1 General view of 3470S adapter

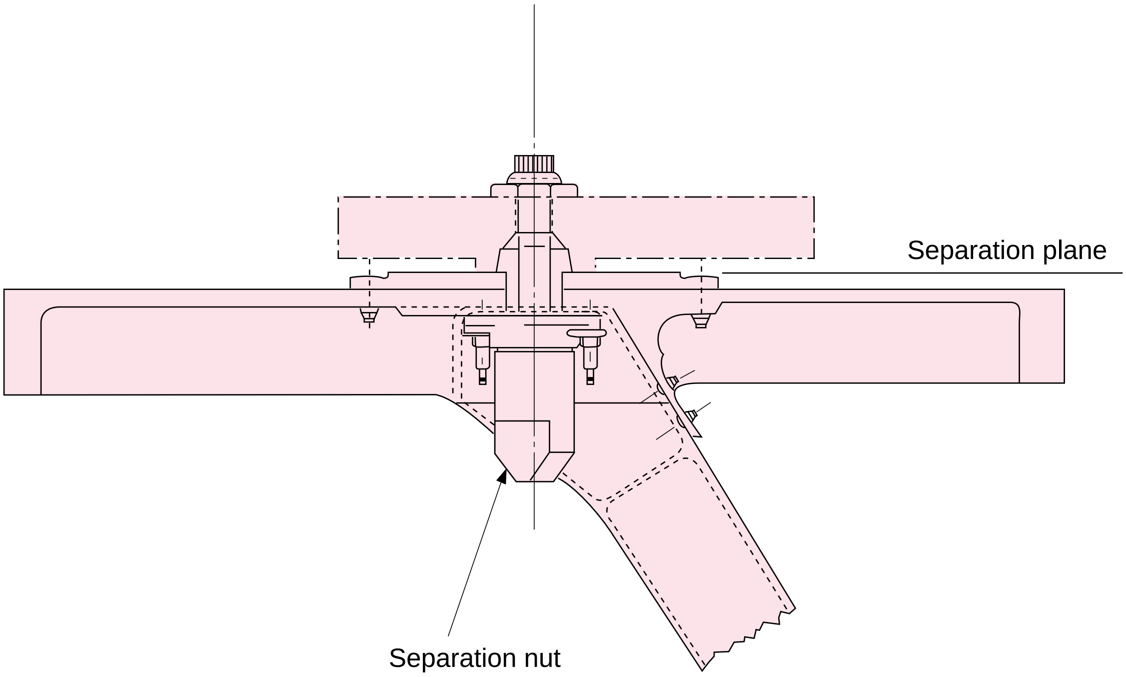
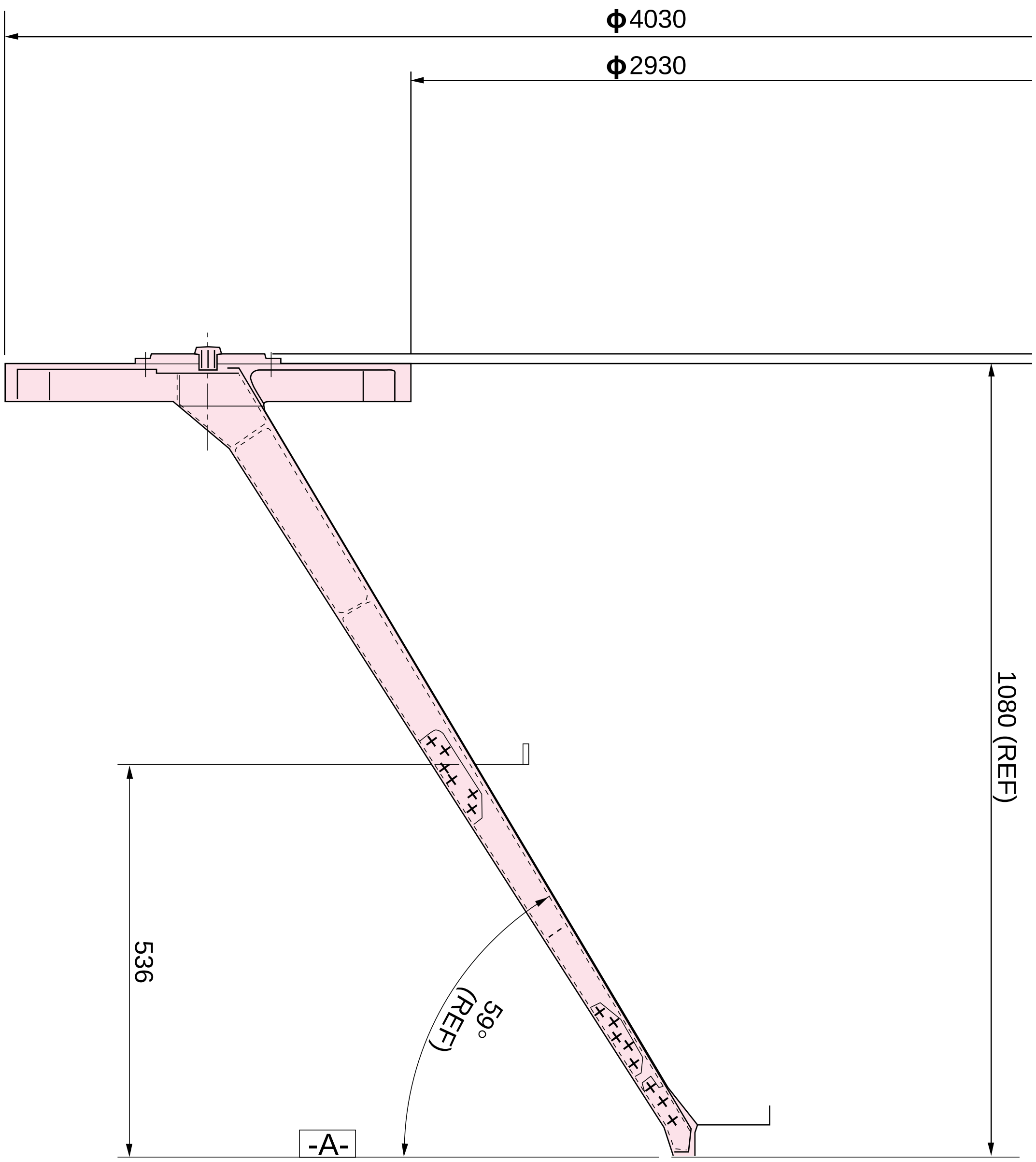


Figure A3.9.2 Details of 3470S adapter #1

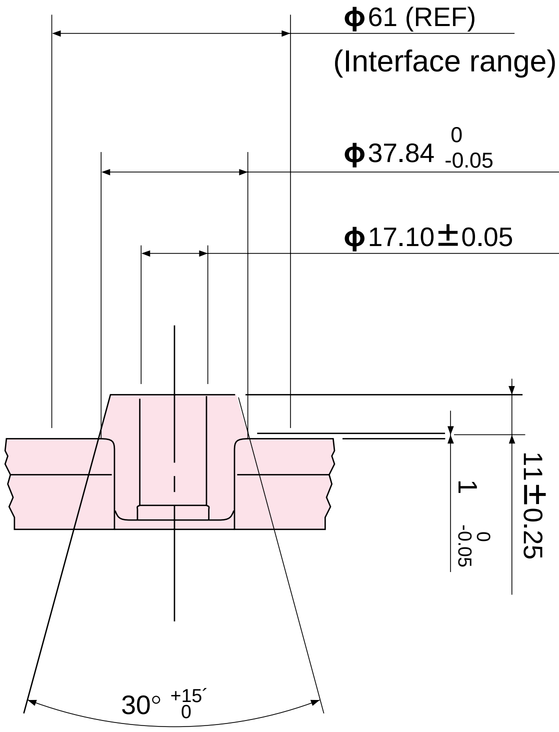
A



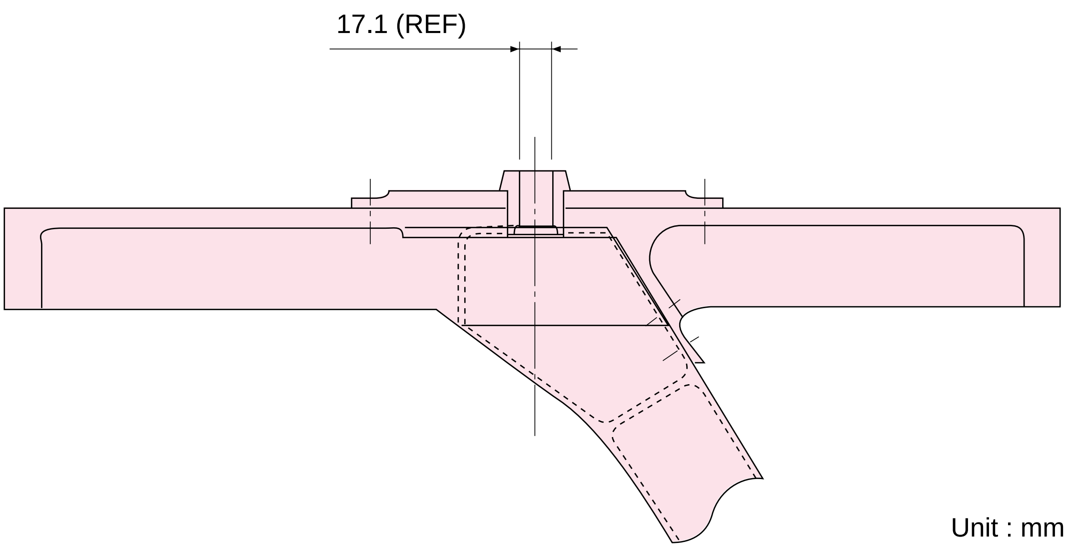
Unit: mm

Figure A3.9.3 Details of 3470S adapter #2

A



Spacecraft connecting point details



Unit : mm

Figure A3.9.4 Details of 3470S adapter #3

A

APPENDIX 3.10 1666MA ADAPTER

C

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 system	: 4 or 8 springs
(6)	Clamp bands	
	Maximum tension	: 38.9 kN
(7)	Maximum load per spring	: 1670 N
(8)	Adapter mass	: 100 kg

This adapter has 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 A3.10.1 shows the photograph of the 1666MA adapter.

Figure A3.10.2 shows the general view of the 1666MA adapter.

Figures A3.10.3 to A3.10.5 show the details of the 1666MA adapter.

Figures A3.10.6 to A3.10.7 show the stay-out zone around the 1666MA adapter.

Figure A3.10.8 shows the limit loads of the 1666MA adapter.

Figure A3.10.9 shows the spacecraft separation shock spectrum of the 1666MA adapter.

Figure A3.10.10 shows the limit loads at separation plane of the 1666MA adapter.

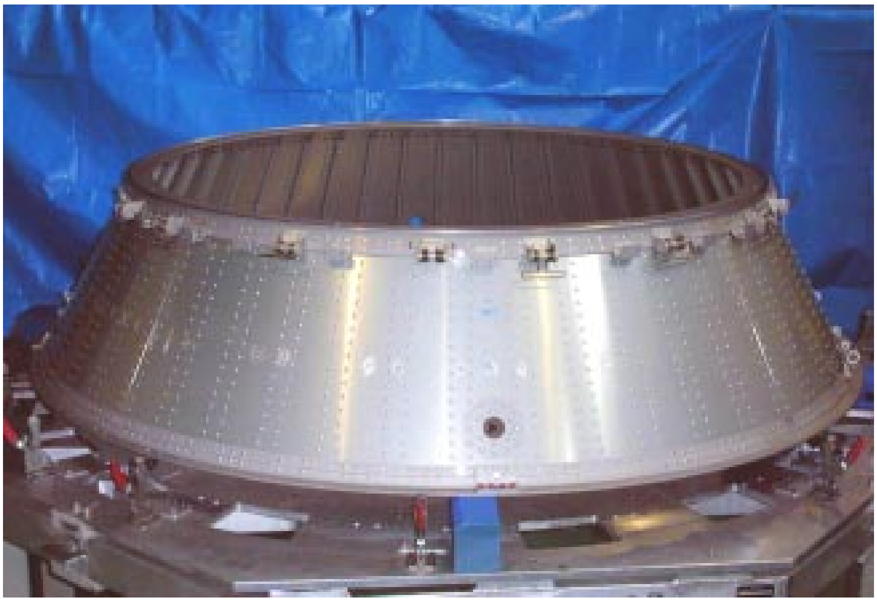


Figure A3.10.1 Photograph of the 1666MA adapter

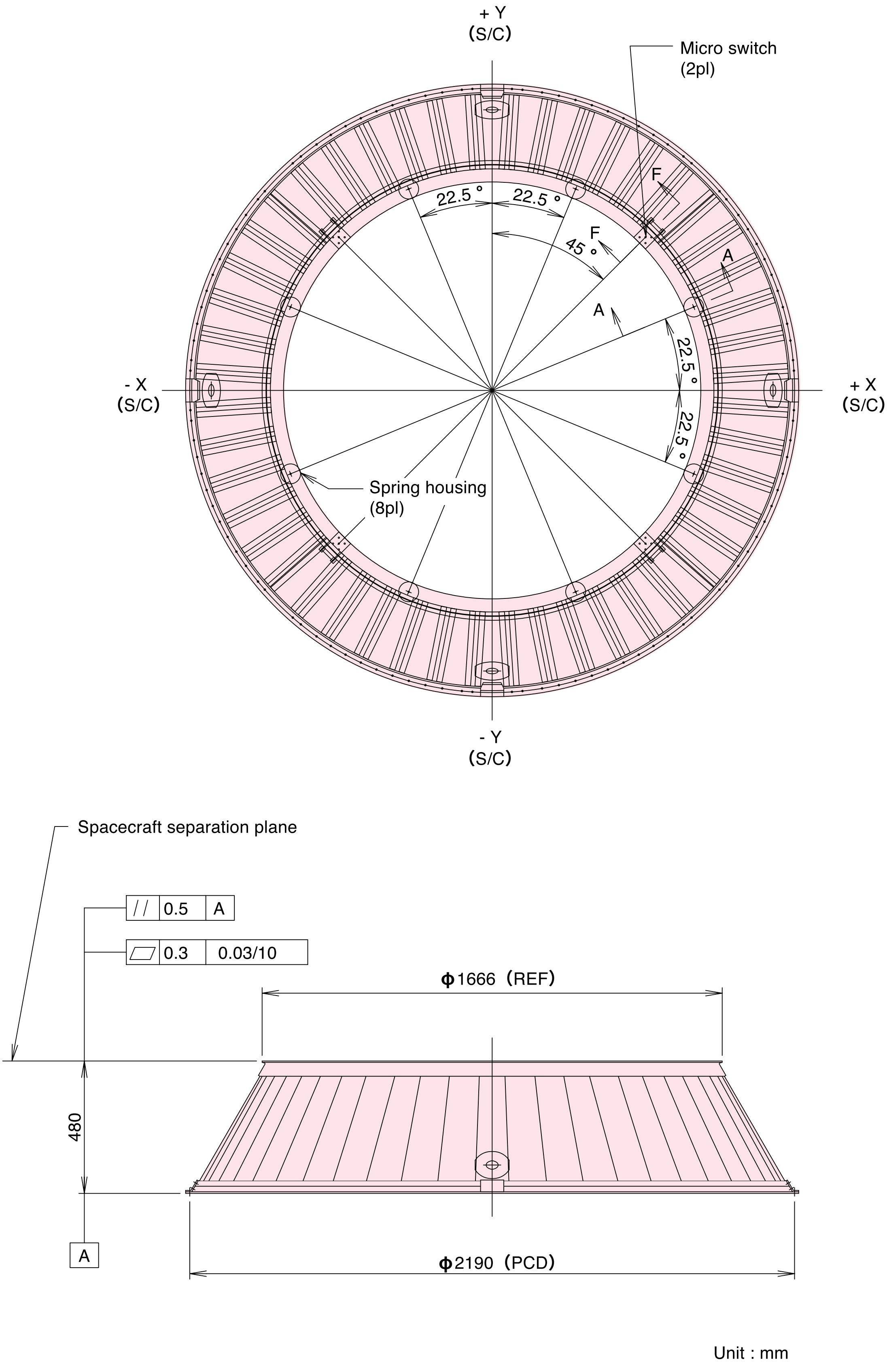
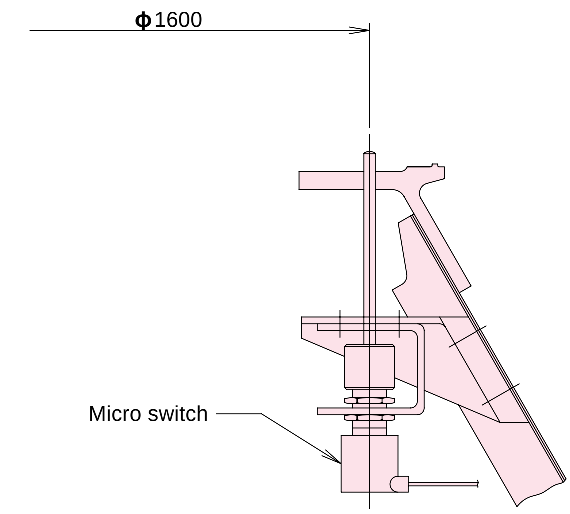


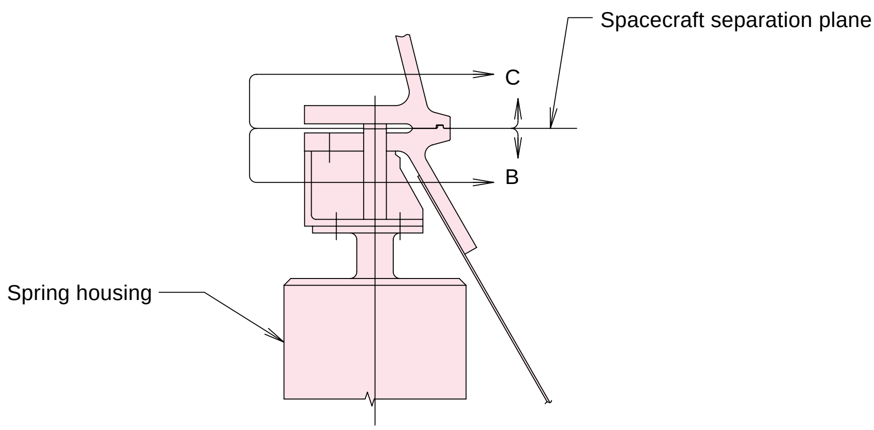
Figure A3.10.2 General view of the 1666MA adapter

C

C



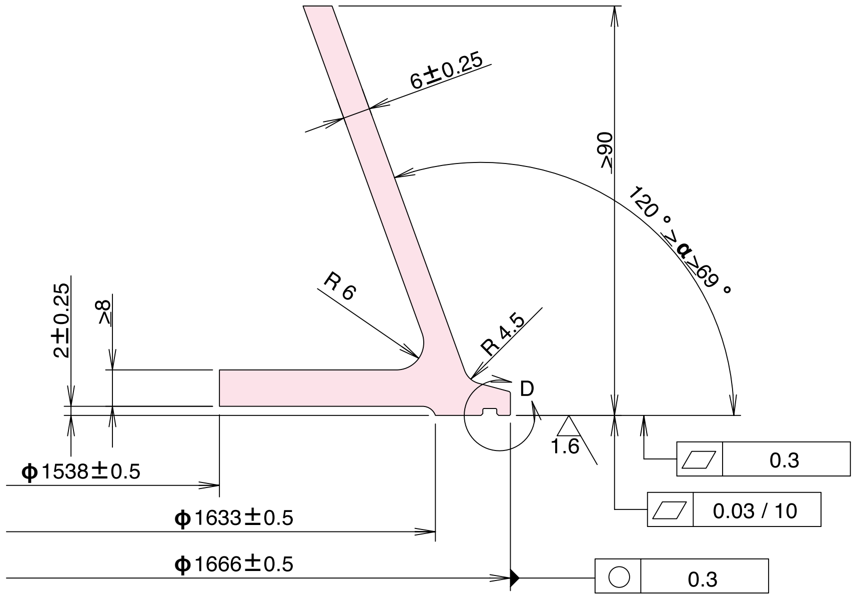
Sect F-F



Sect A-A

Unit : mm

Figure A3.10.3 Details of the 1666MA adapter #1

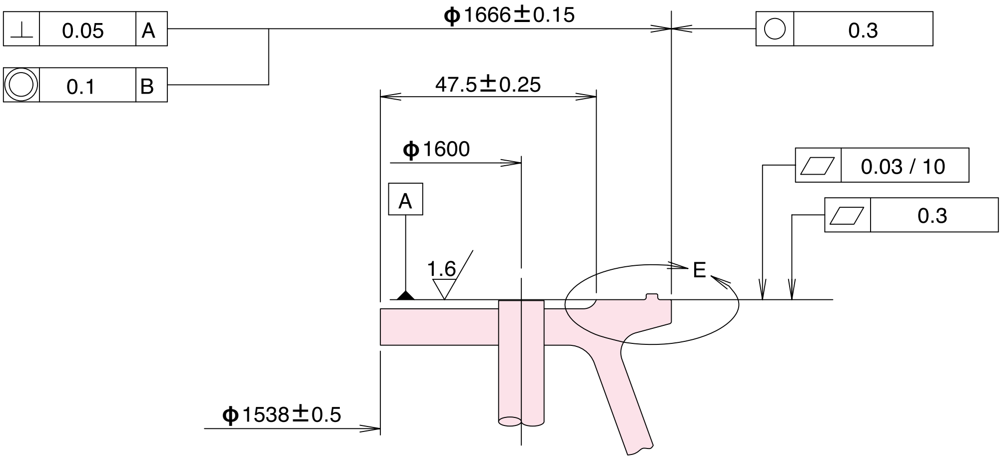


<Note>

Material	Aluminium Alloy
Area	S ≧ 804 mm²
Inertia	Ixx ≧ 652590 mm⁴
	Iyy ≧ 64570 mm⁴
Applicable length	L ≧ 90 mm

Spacecraft frame

Detail C



<Note>

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

Adapter

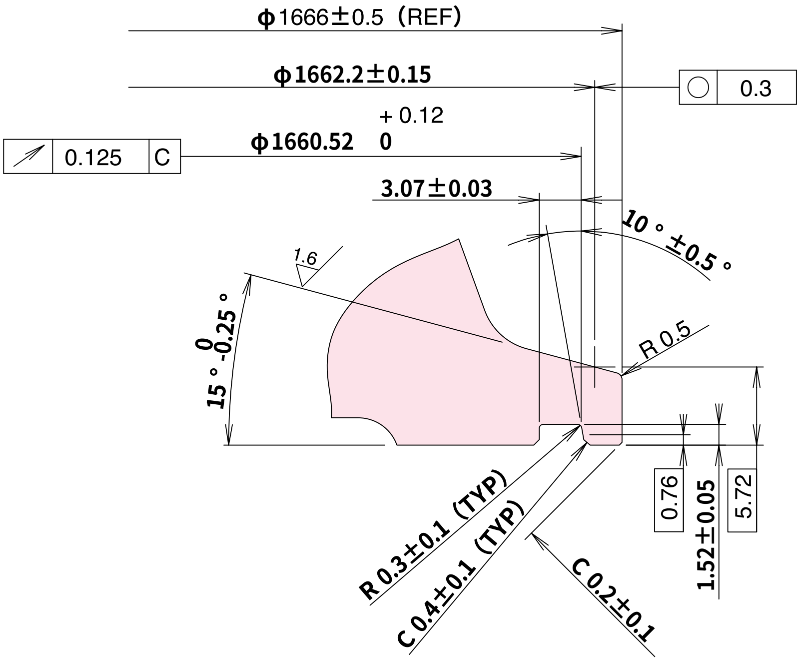
Detail B

Unit : mm

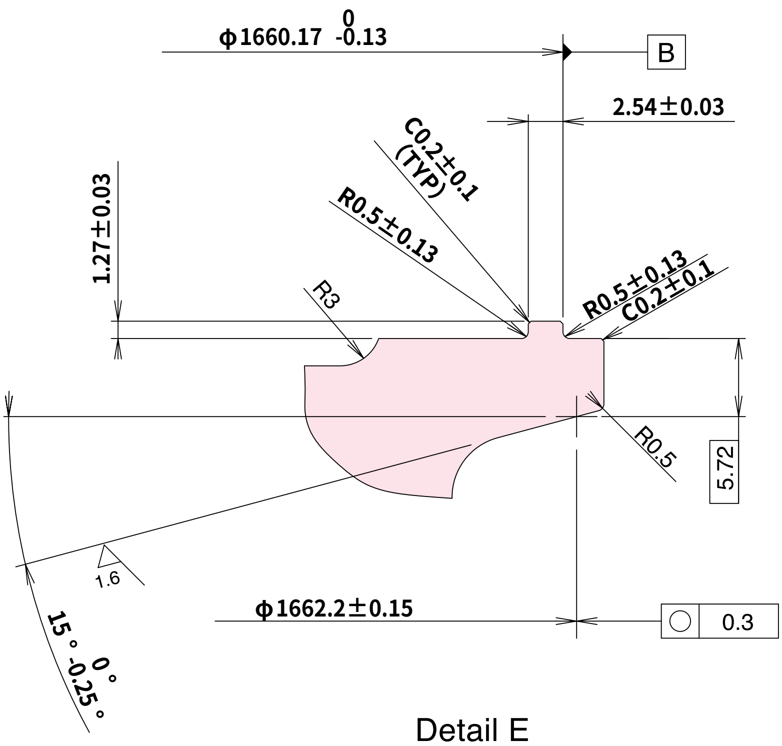
Figure A3.10.4 Details of the 1666MA adapter #2

C

C



Detail D

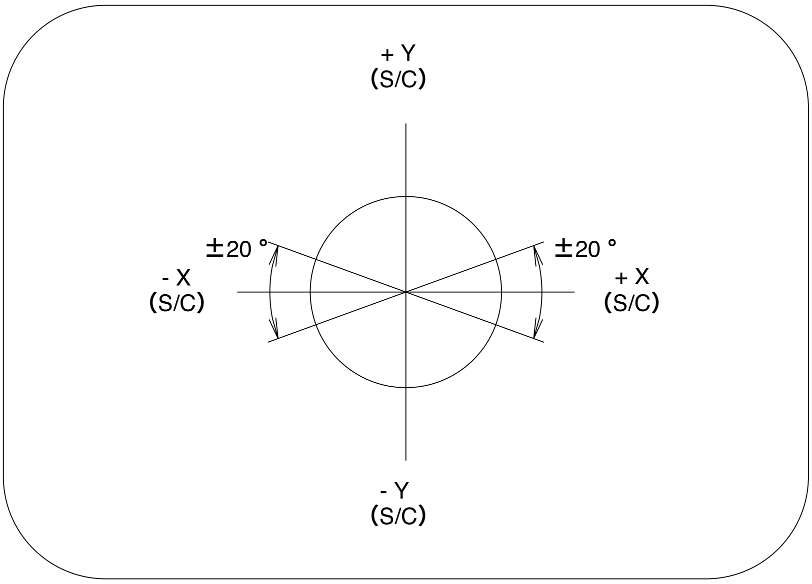


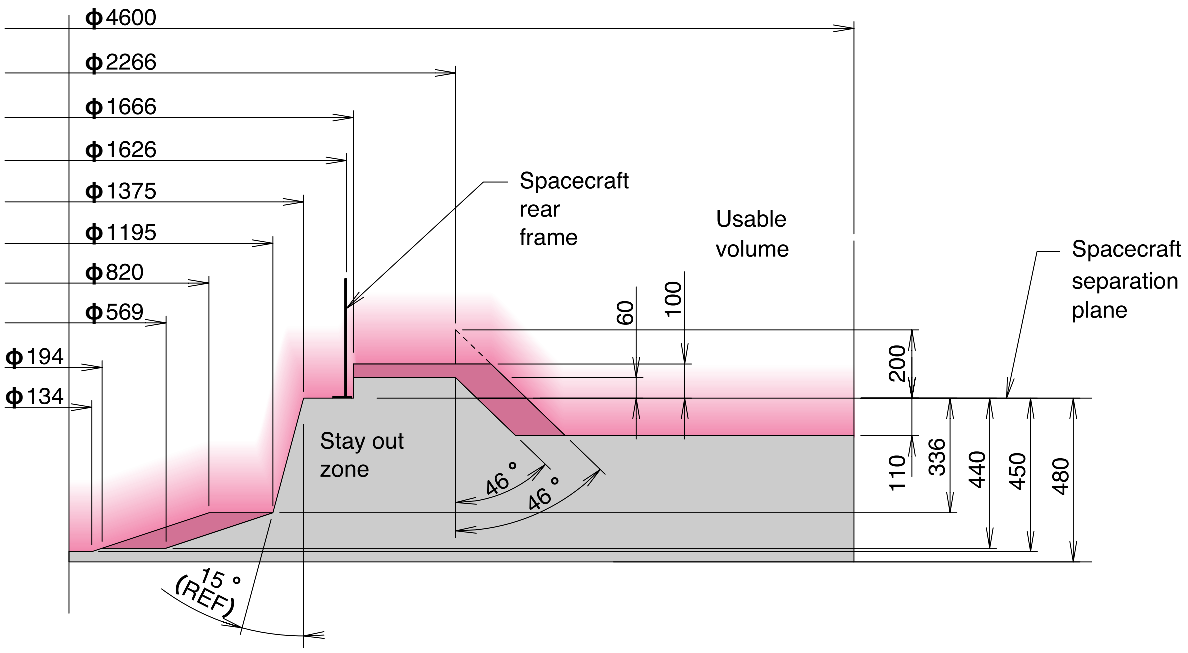
Detail E

Unit : mm

Figure A3.10.5 Details of the 1666MA adapter #3

C





: This area must be coordinated with NASDA.

Unit : mm

Figure A3.10.6 Stay-out zone around the 1666MA adapter (+X / -X)

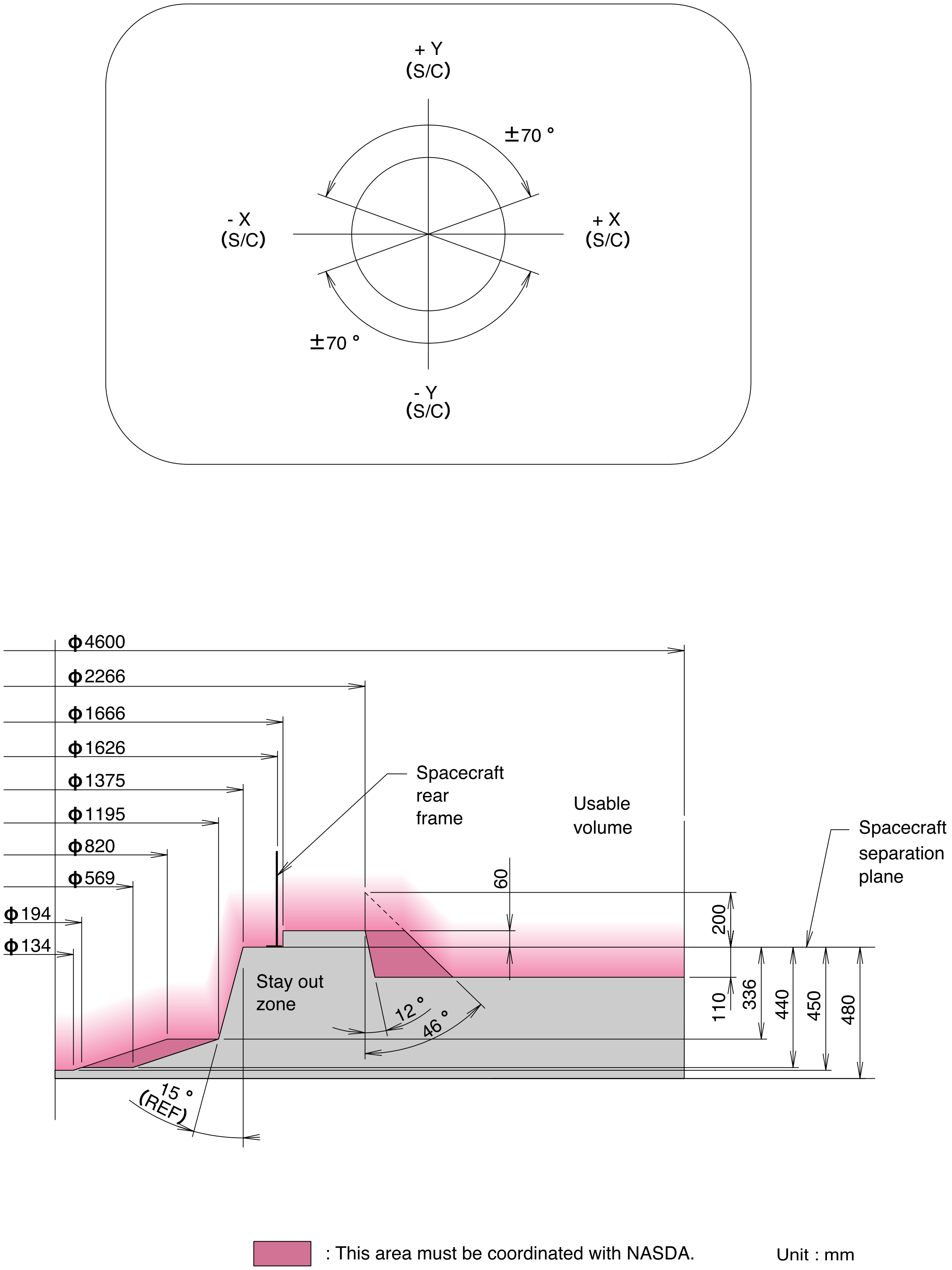
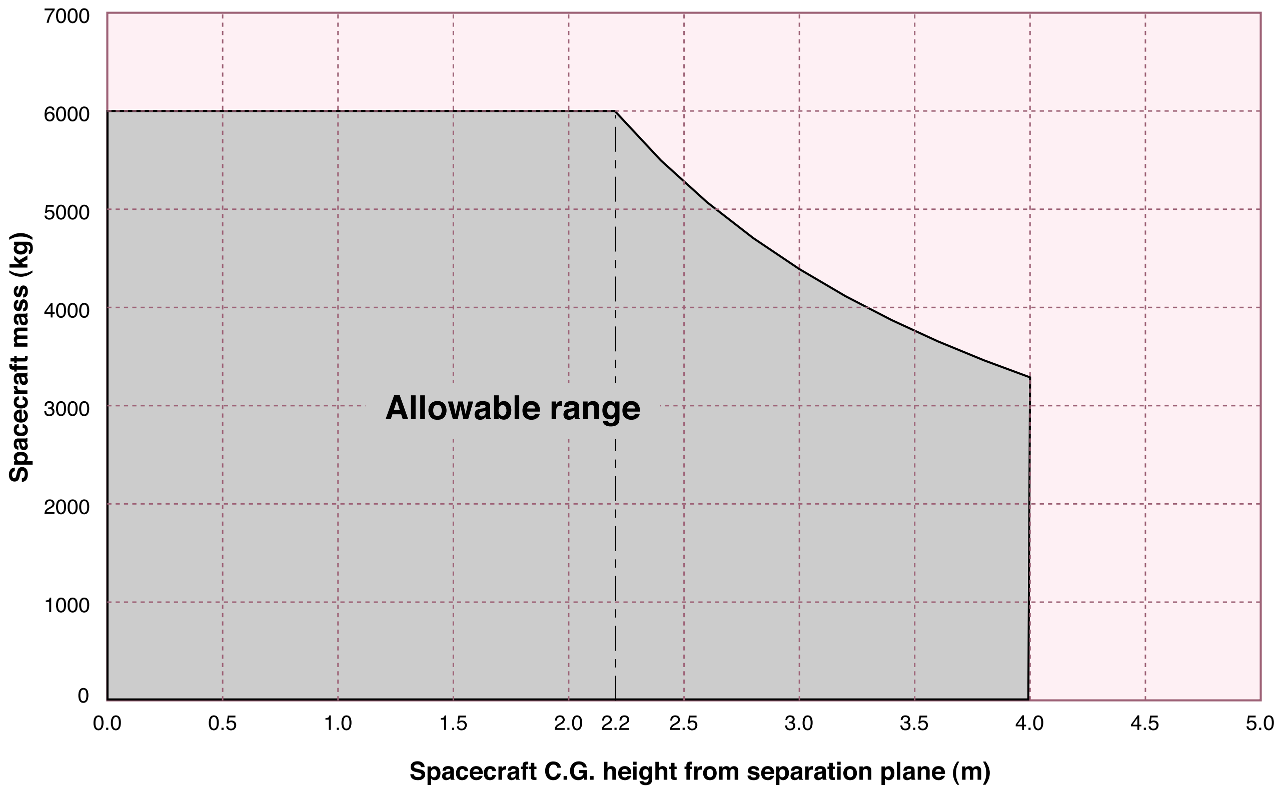
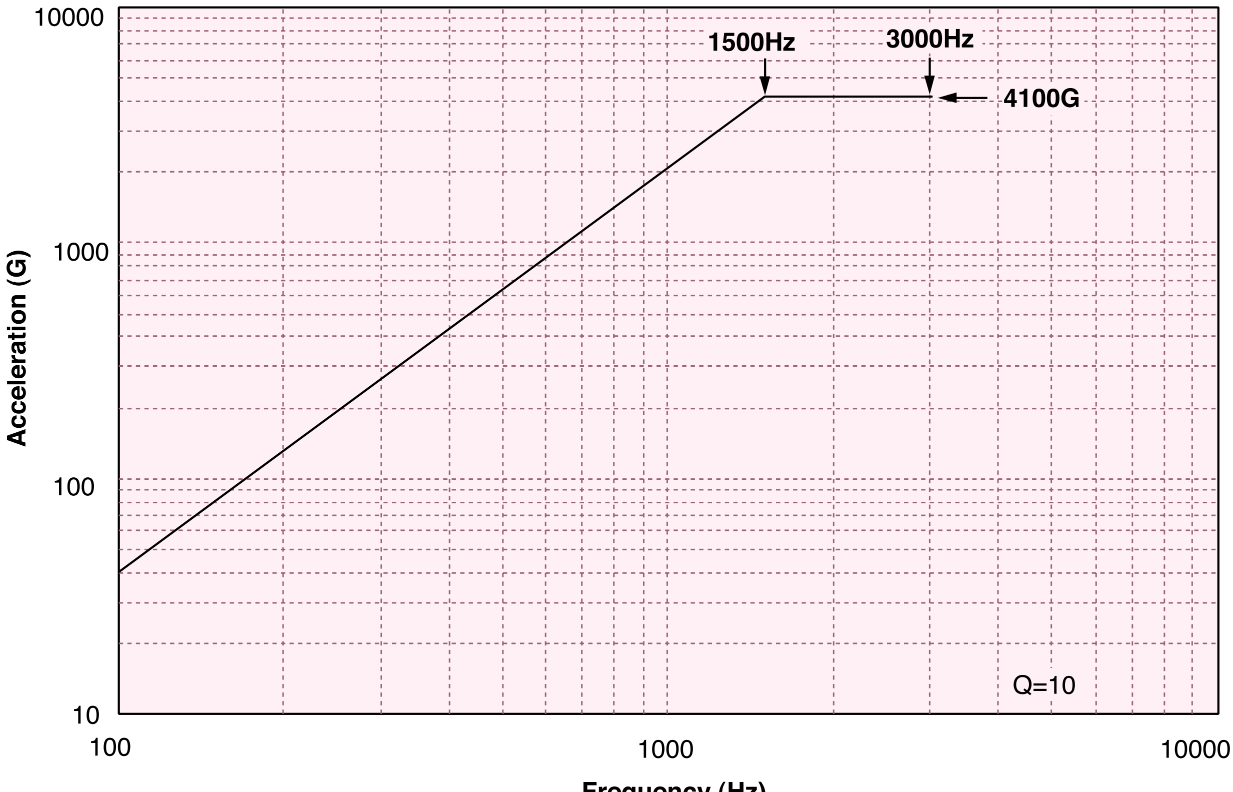


Figure A3.10.7 Stay-out zone around the 1666MA adapter (+Y / -Y)



C

Figure A3.10.8 Limit loads of the 1666MA adapter



G = 9.80665 m/s^2

Figure A3.10.9 Spacecraft separation shock spectrum of the 1666MA adapter

C

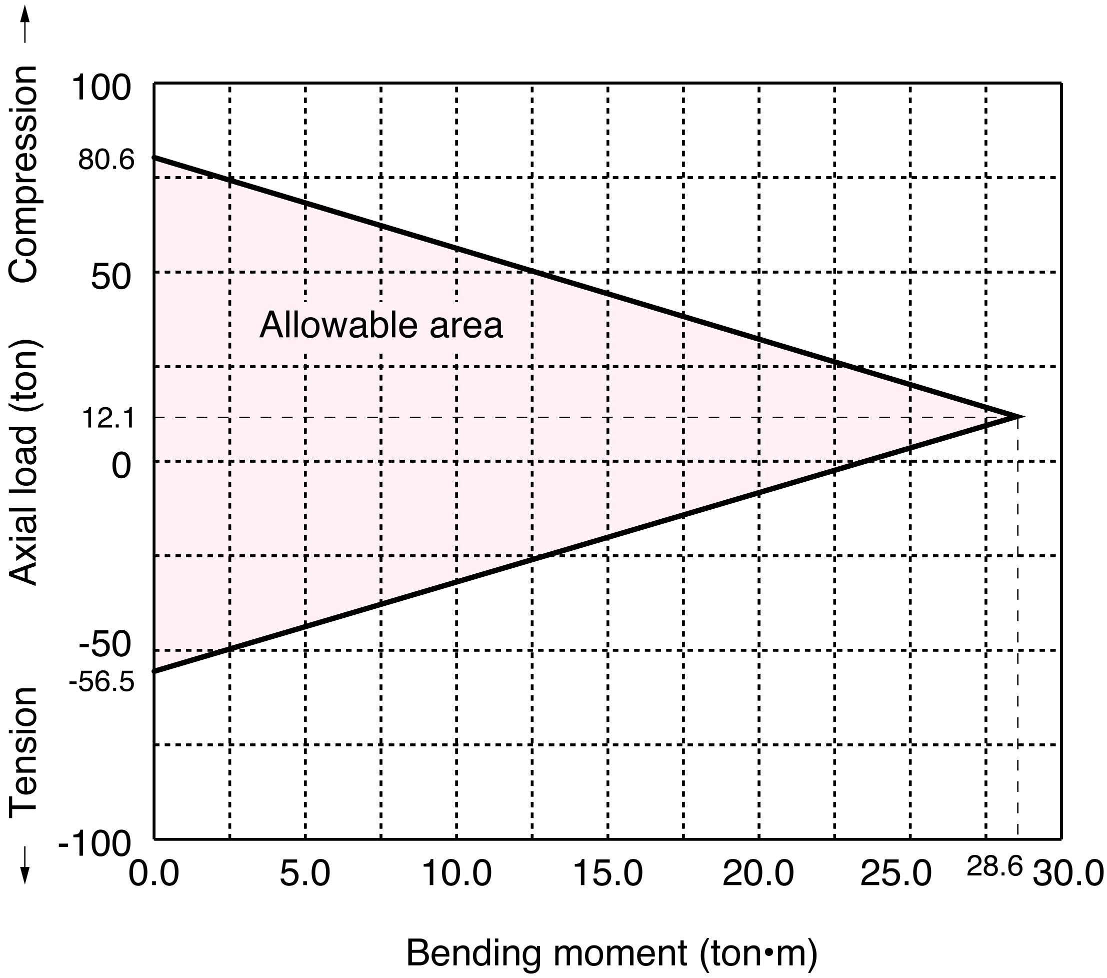


Figure A3.10.10 Limit loads at separation plane of the 1666MA adapter

C

APPENDIX 3.11 239M ADAPTER

C

The main characteristics are as follows.

(1) Interface diameter : 239 mm
(2) Height : 100 mm
(3) Material : Aluminum semi-monocoque
(4) Attached system : Clamp bands
(5) Separation system : 3 springs
(6) Clamp band
    Maximum tension : 7.00 kN
(7) Maximum load per spring : 500 kN
(8) Adapter mass : 5.0 kg

This adapter is for the piggyback satellite, and is mounted on the piggyback satellite support structure which is installed at the side of the PSS.

The mass of the spacecraft shall be 50 to 100 kg.

When interface connector is installed on the separation place, the connector shall be located at the center of adapter axis.

Figure A3.11.1 shows the general view of the 239M adapter .

Figure A3.11.2 shows the details of the 239M adapter.

Figure A3.11.3 shows the spacecraft envelope of the piggyback satellite for the 239M adapter.

C

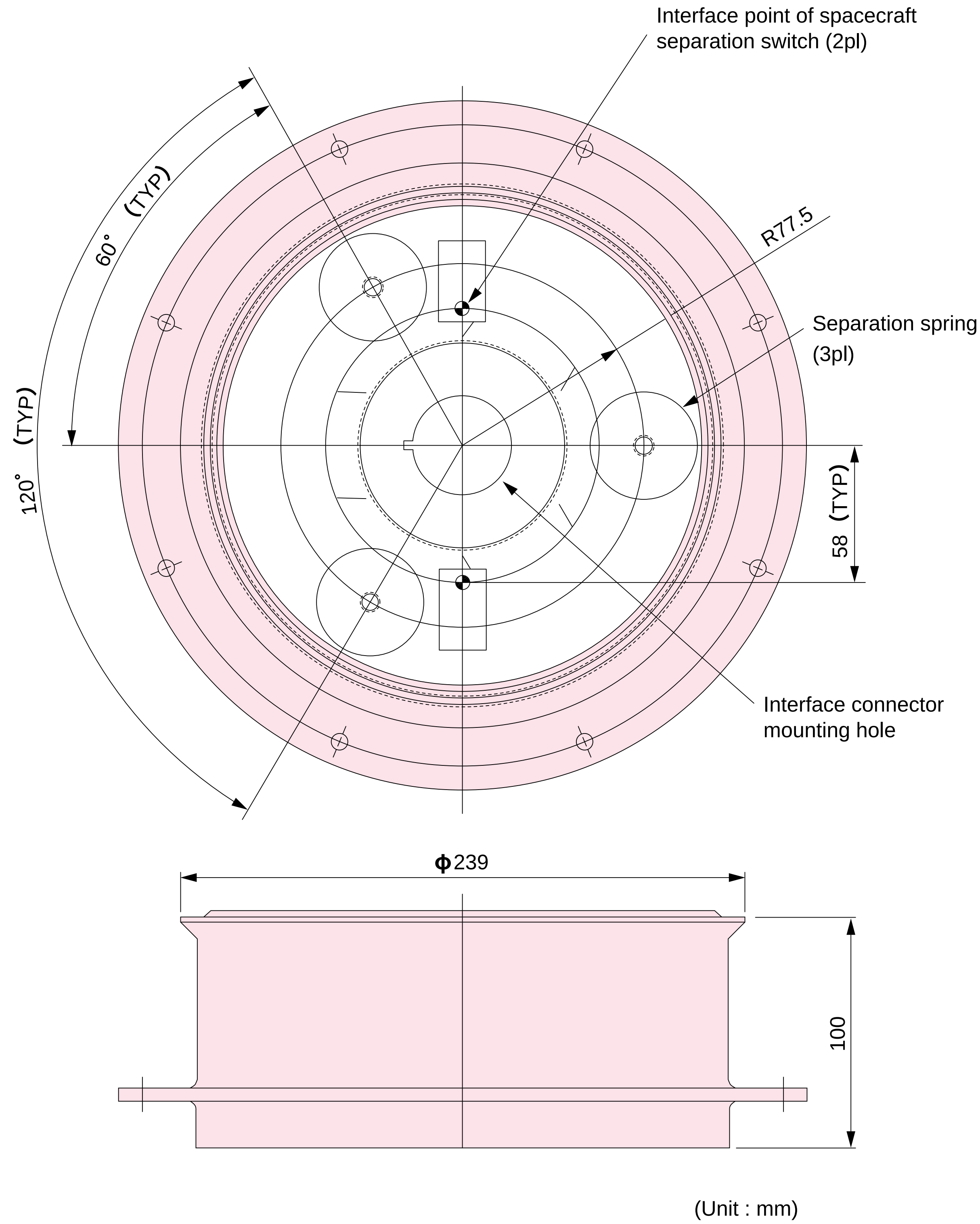


Figure A3.11.1 General view of the 239M adapter

C

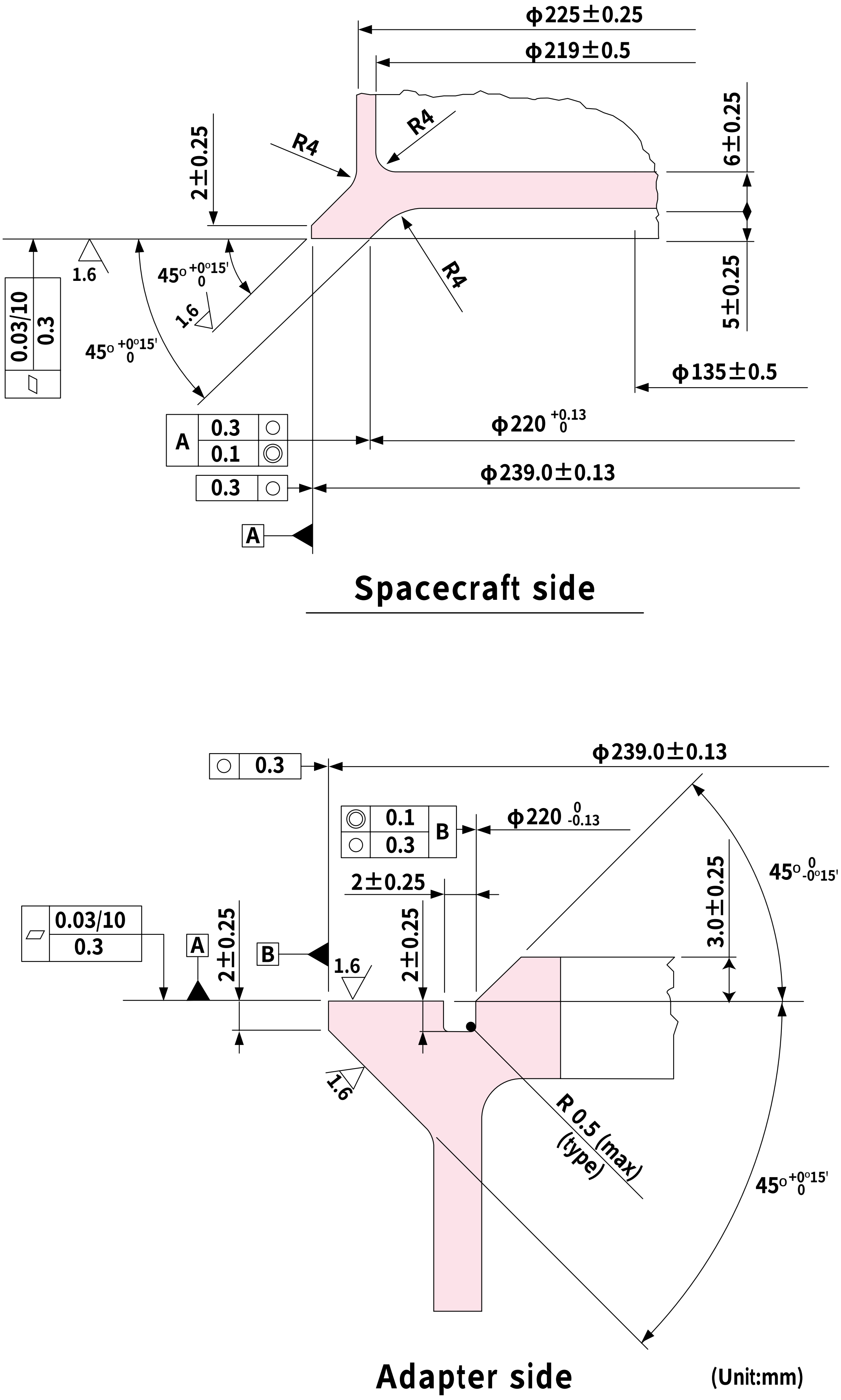


Figure A3.11.2 Details of the 239M adapter

C

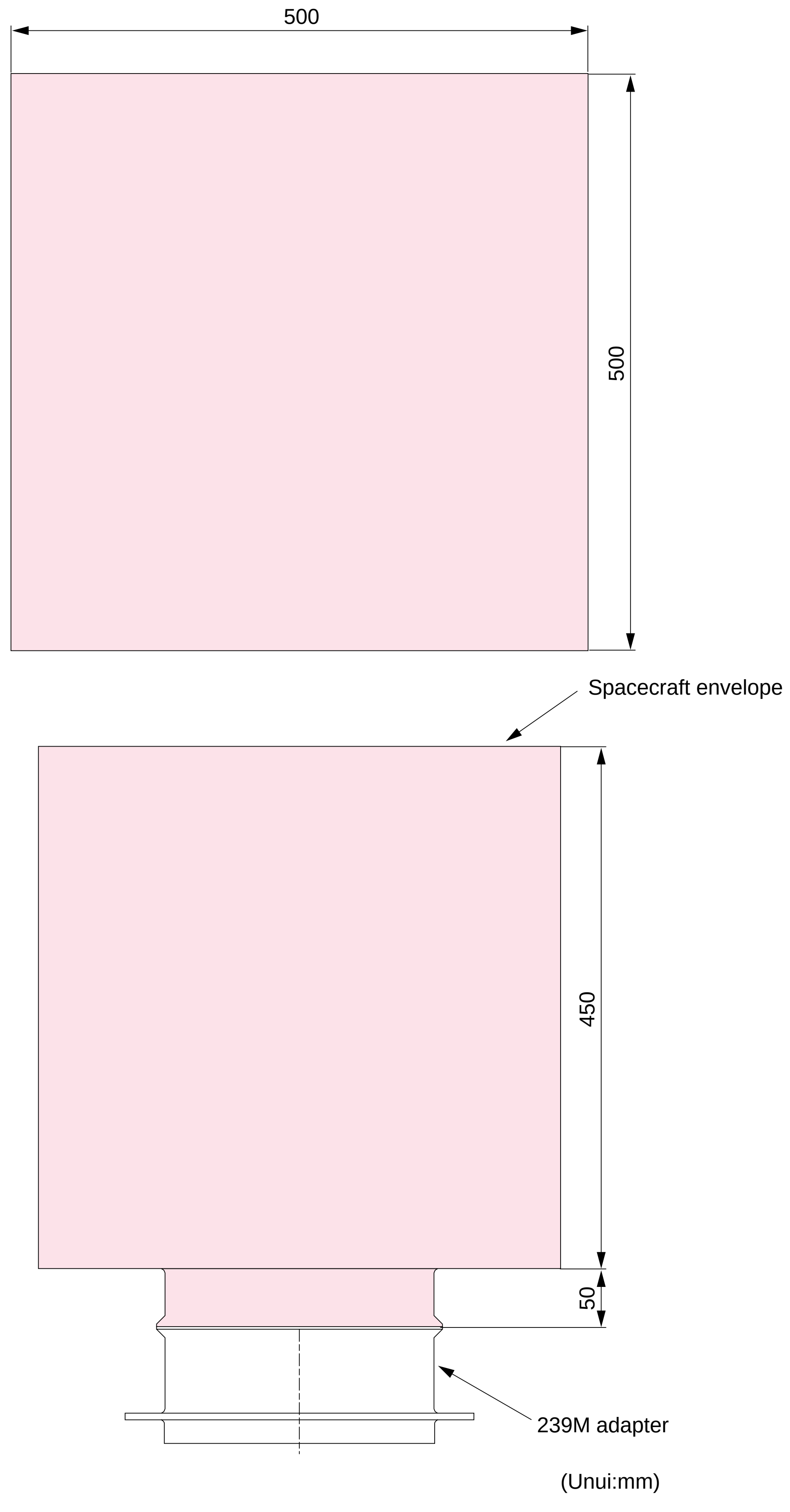


Figure A3.11.3 Spacecraft envelope of the piggyback satellite for the 239M adapter