GSAT-11 SPACECRAFT
Functional Specification

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SCOPE

This document details the functional specifications of the communication payloads of GSAT-11 satellite. Also, the broad functional requirements of the platform systems to support the payload operations are described.

CONTENTS

CHAPTER-1

1 GSAT-11 Configuration

GSAT-11 is a Geo-stationary satellite to be located at orbital slot of 74°E longitude carrying multi-beam communication payload as described in Chapters #3 and #4.

GSAT-11 Spacecraft is configured on ISRO's highly modular I-6K bus. Most of the functional requirements of the communication payloads and the bus platform systems have been derived from the earlier ISROs geostationary satellites INSATs / GSATs.

1.1 GSAT-11 Functional Requirements

1.1.1 All the transponders shall be capable of operating within its specified characteristics and performance both in sunlit and eclipse conditions without any operational restrictions.

1.1.2 The spacecraft having Ka x Ku band and Ku x Ka band transponders in bent-pipe configuration, which provide fixed satellite services to multiple users through star-based configurations.

1.1.3 GSAT-11 dry mass and lift off mass (with propellant) shall be compatible with commercially available launcher Ariane 5 vehicle for an operational mission life of minimum 15 years.

1.1.4 The spacecraft shall be primarily operated from 74°E longitude location.

1.2 GSAT-11 Payload Functional Requirement

GSAT-11 multi-beam communication payload consists of:

- Thirty-two forward link (Ka x Ku) transponders, operating in the 30 GHz uplink and 11 GHz downlink.
- Eight return link transponders, operating in 13 GHz uplink and 20 GHz downlink.
- The payload coverage polygon shall encompass India mainland and Andaman & Nicobar Islands.
- Payload is configured with a Tracking system which will be shared among the four Ku-band reflector systems for maintaining the desired pointing accuracy. The signal for tracking system will be uplinked from four identified stations from ground.
- A switchable Ka x Ka-band direct link is provided between Western and Southern hub beam.
- Four of the Ka x Ku transponders are configured in Multiport Amplifier configuration for dynamic sharing of power among four of the user beams.
- The payload is equipped with two Ku-band Beacons (in orthogonal polarization). Ku band beacon transmitter shall work at 10701 MHz Linear vertical and linear horizontal polarization.
- The payload is equipped with two Ka-band Beacons (in orthogonal polarization). The Ka Beacon transmitter shall work at 19701.5 MHz Linear vertical and linear horizontal polarization.

To meet these requirements, the payload performance specifications are defined in the Chapters #3 and #4 of this document.

From the payload usage point of view, the satellite shall be operated primarily from 74^{0}E geostationary longitudinal location and shall provide communication services for a minimum of 15 years.

CHAPTER-2

2 GSAT-11 Spacecraft System Requirement

2.1 Launcher Capability

GSAT-11 spacecraft shall be compatible with commercially available launch vehicles and from the point of view of:

a. Lift off Mass
b. Interfaces
c. Envelope & load capacity
d. Launch environment
e. Safety requirements

Launch system / Spacecraft INTERFACE CONTROL document and further revisions as and when done with the launcher agency shall be applicable.

2.2 Functional Redundancy

The satellite shall incorporate parts, components, subassembly, assembly, and / or subsystem functional redundancy wherever necessary. The satellite subsystems shall include fail-safe features wherever feasible i.e., a failure within a subsystem shall not disable or degrade the remainder of the satellite. Failure rates for parts shall be assessed as per MIL-STD-217F wherever applicable and or other relevant data from ISRO documents shall be applicable.

The following shall not be included in computing reliability of the spacecraft:

a. Successful injection into parking orbit by the launch vehicle.
b. Apogee engine performance.
c. Structure and Mechanism elements.

2.3 Design Life

The minimum in orbit mission lifetime of the GSAT-11 spacecraft shall be 15 years (with a design goal of 18 years). Unless otherwise stated, the satellite shall be designed to meet all specifications at the End Of Life (EOL). Spacecraft parts, materials and processes subject to wear / tear out or deterioration due to environment including radiation, application stresses or inherent physical processes, shall be designed, fabricated, selected and used to attain performance life requirements.

2.4 Operating Environments

The spacecraft shall be capable of satisfactory operation in standard laboratory environments, vacuum (pressure less than or equal to 10^{-6} torr) and operational environments during the entire mission. In addition, all conductors shall be insulated such that they shall not be subject to electrical break down due to any of the test or operational environments. All subsystems, which are operating during launch, shall be able to operate in the electromagnetic environments of the chosen launcher.

The spacecraft design shall take into consideration space charging phenomena. The spacecraft design shall provide for the electrical connection of all electrically conductive structural elements of the spacecraft to form a common spacecraft ground. Provisions shall be made to electrically ground all conductive external surfaces of the spacecraft. All interface circuits shall be designed to minimize electrical interference on signal lines/return lines, power bus lines/ return lines and spacecraft structure.

2.5 Storage

The spacecraft shall be capable of satisfactory operation following storage up to 5 years in its shipping and storage container. Spacecraft batteries shall be excluded from this storage life time requirements.

2.6 Spacecraft Design / Operation from AIT Point of View

The spacecraft shall be designed:

2.6.1 To provide ease of access to subassemblies for electrical connections, mechanical operations, inspection and alignments, taking into account provisions for any changes or replacements that may be needed on any test sequence. The power distribution harness is made with the use of power bus bars, entire generation and distribution chain is double insulated till fuse end.

2.6.2 To allow tele-metered data to be used during ground testing to the maximum extent in order to minimize use of test connectors.

2.6.3 To preclude incorrect electrical connections.

2.6.4 To reduce the hazards of incorrect assembly.

2.6.5 To allow selection of all stand-by / redundant units with minimum service interruption.

Materials used in the spacecraft design and fabrication should be chosen carefully to protect against the contamination of the spacecraft equipment, especially solar cells, thermal control surfaces and optical surfaces. Specific attention to the magnetic cleanliness shall be given since electromagnetic devices are used for the control purposes of the spacecraft. Spacecraft testing shall be accomplished with minimum disturbance to flight connectors. The spacecraft design shall allow for proper stimulation to various attitude sensors and the communication subsystem for functional tests at the launch site.

2.7 Launch Window

Launch window for the selected launcher shall be determined by Mission Analysis and shall be at least 45 minutes each day of the year for the GTO injection orbit.

2.8 Station Keeping

The satellite shall be capable of being maintained during its entire mission life at a nominal position of 74°E longitude to control orbit inclination to within ±0.1deg of the equatorial plane and longitude within ±0.1deg of the selected nominal location along with co-located satellites. The spacecraft shall be designed to accommodate the resulting magnitude and frequency of inclination-control maneuvers required over the nominal orbital location. During the life time of the satellite, provisions shall be made to accomplish satellite re-positioning at least once.

2.9 Platform and Antenna Pointing Errors

The design of the spacecraft shall provide for controlling the maximum (3σ) antenna pointing errors to within the limits shown in the table below:

Platform Pointing error limits (Normal Mode / Station keeping Mode)	
Pitch	± 0.15 deg.
Roll	±0.15 deg.
Yaw	±0.20 deg

Note: The antenna pointing specifications are ±0.05° about all 3 axes

The above indicated values shall be used in conjunction with the coverage requirements defined in the payload performance requirements chapters of this document to establish the necessary antenna coverage.

During Sun interference periods which last for less than 5 minute, the antenna pointing in the roll & yaw axis shall be allowed to have an additional pointing error of less than 0.1 deg.

2.10 Eclipse Operations

All of the transponder capacity shall be capable of operating as per their specifications during the full eclipse duration. There shall not be any operational restrictions for operating the payloads during eclipse.

2.11 Particle Radiation Protection on Orbit

The spacecraft design shall, to the maximum extent possible within the other design requirements contained in this specification, configure the satellite so as to provide maximum inherent protection to radiation sensitive parts/materials. Specifically it shall be able to withstand without degradation of performance two solar flare events with a magnitude similar to the August 1972 solar flare.

2.12 Spacecraft Axes Definition

The satellite axes are defined as fixed in the satellite in accordance with Figure 2.1. When the satellite is in the operational configuration in orbit:

a) The positive yaw axis nominally in the orbital plane and directed towards the center of the Earth (+X);
b) The positive roll axis (+Y) is nominally in the orbital plane in the direction of orbital motion;
c) The positive pitch axis in nominally normal to the orbital plane and directed towards the south (+Z).

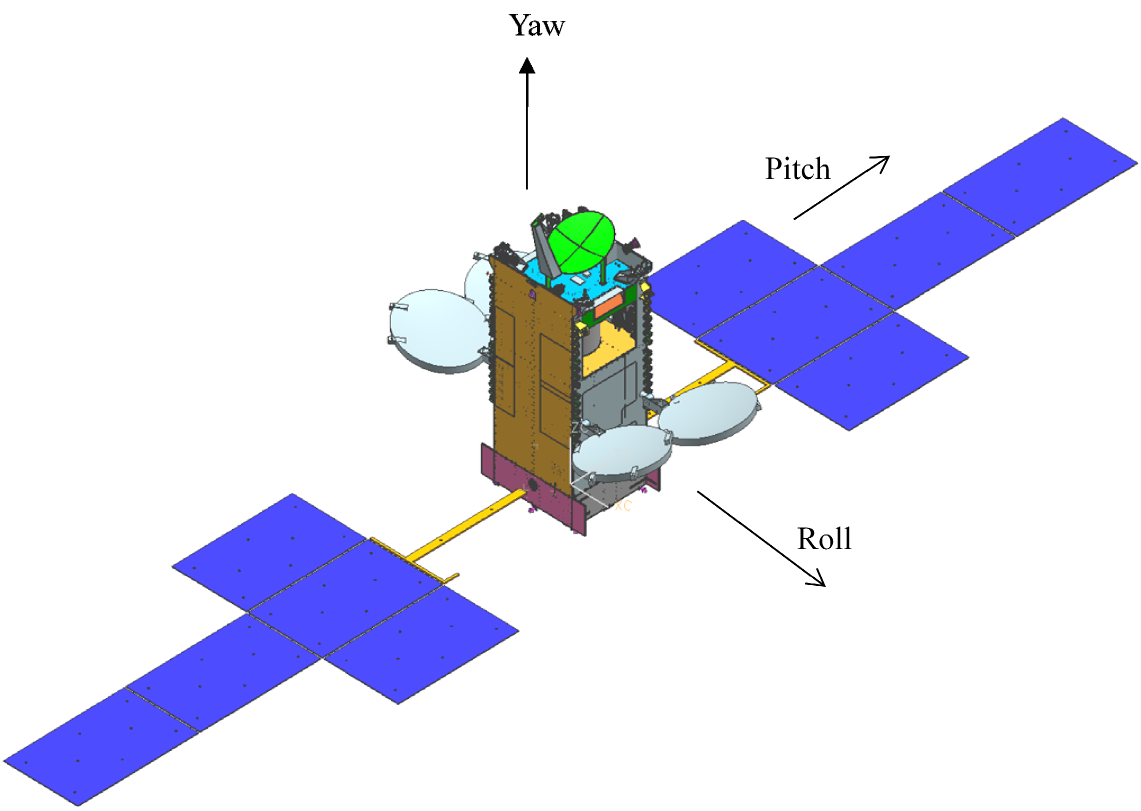


Figure 2.1 GSAT 11 Spacecraft

CHAPTER 3

3 PAYLOAD PERFORMANCE REQUIREMENTS

GSAT-11 to be located at 74 deg East is a multi-beam communication payload configured to cater to the Fixed Satellite Services (FSS) over Indian mainland and Islands. The satellite is having Ka x Ku band and Ku x Ka band transponders in bent-pipe configuration, which provide fixed satellite services to multiple users through star-based configurations.

The forward link is defined from gateway to user terminal via Ka x Ku channels. The return link is defined from user terminal to hub via Ku x Ka channels.

The payload shall provide a total of 40 transponders channels out of which:

a. Thirty-two forward link (Ka x Ku) transponders, operating in the 30 GHz uplink and 11 GHz downlink.

b. Eight return link transponders, operating in 13 GHz uplink and 20 GHz downlink.

The Transponder configuration diagram with Ka x Ku & Ku x Ka payload chains is shown in Figure 3-2 to Figure 3-7.

Payload is configured with a Tracking system which will be shared among the four Ku-band reflector systems for maintaining the desired pointing accuracy. Figure 3-6 provides the block schematic of tracking system. The signal for tracking system will be uplinked from four identified stations from ground.

Payload is also configured with a switchable Ka x Ka-band direct link between Western and Southern hub beam (connectivity shown in Figure 3-2 and Figure 3-4).

Four of the Ka x Ku transponders are configured in Multiport Amplifier configuration(shown in Figure 3-2) for dynamic sharing of power among four of the user beams.

Figure 3-7 shows the antenna Interface diagram with the payload.

3.1 Frequency Plan

The payload shall operate in the frequency bands given below in Table 3-1.

Table 3-1: Spectrum Utilization

Transponder	Uplink (MHz)	Downlink (MHz)
Ka x Ku (Forward Link)	29500-30000 In both LH & LV	10700 – 10950, 11200 - 11450 In LH & LV
Ku x Ka (Return Link)	12750 - 13250 In LH & LV	19700 – 20200 In both LH & LV
Ka x Ka (Switchable Link)	29500 – 29750 in LV	19700 – 19950 in LH

The individual Beam plan with frequency allocation shall be as per Table 3-4 and Table 3-5.

An illustration of the frequency bands exploited in GSAT-11 system is shown in Figure 3-1 . The arrows represent the nominal uplink-to-downlink frequency band mappings.

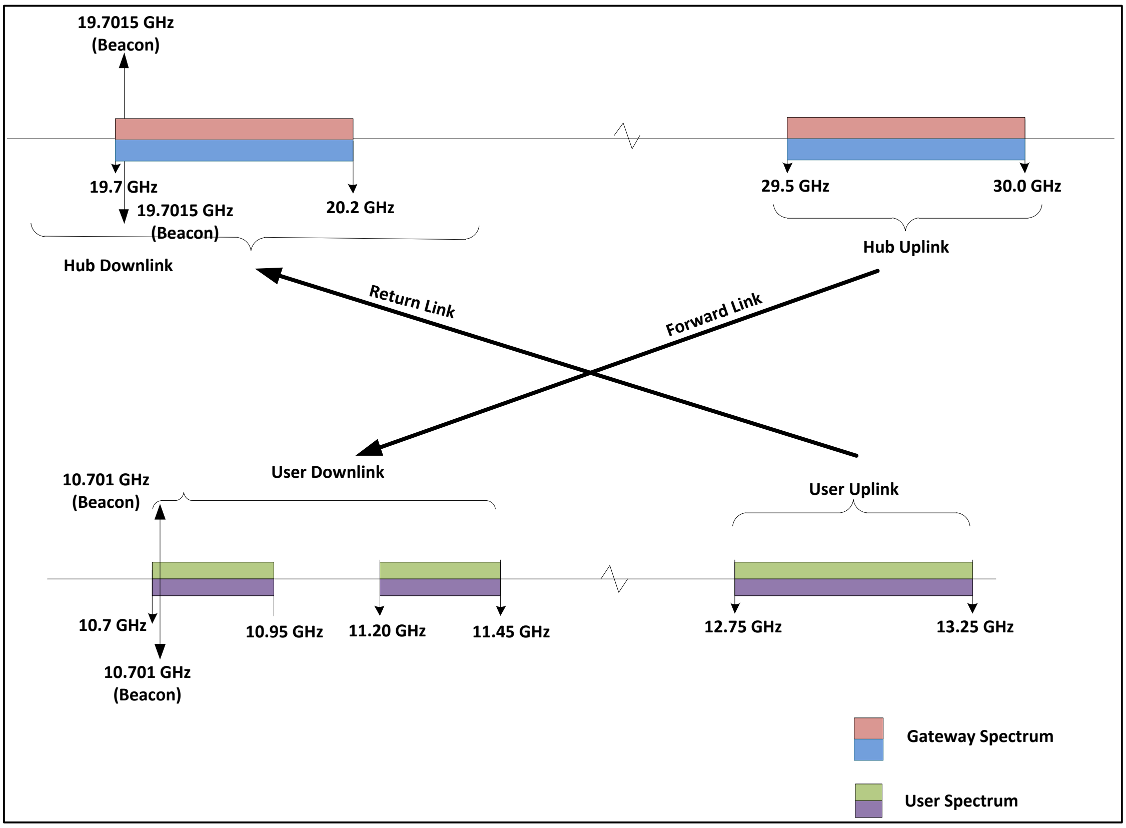


Figure 3-1 System Frequency Plan

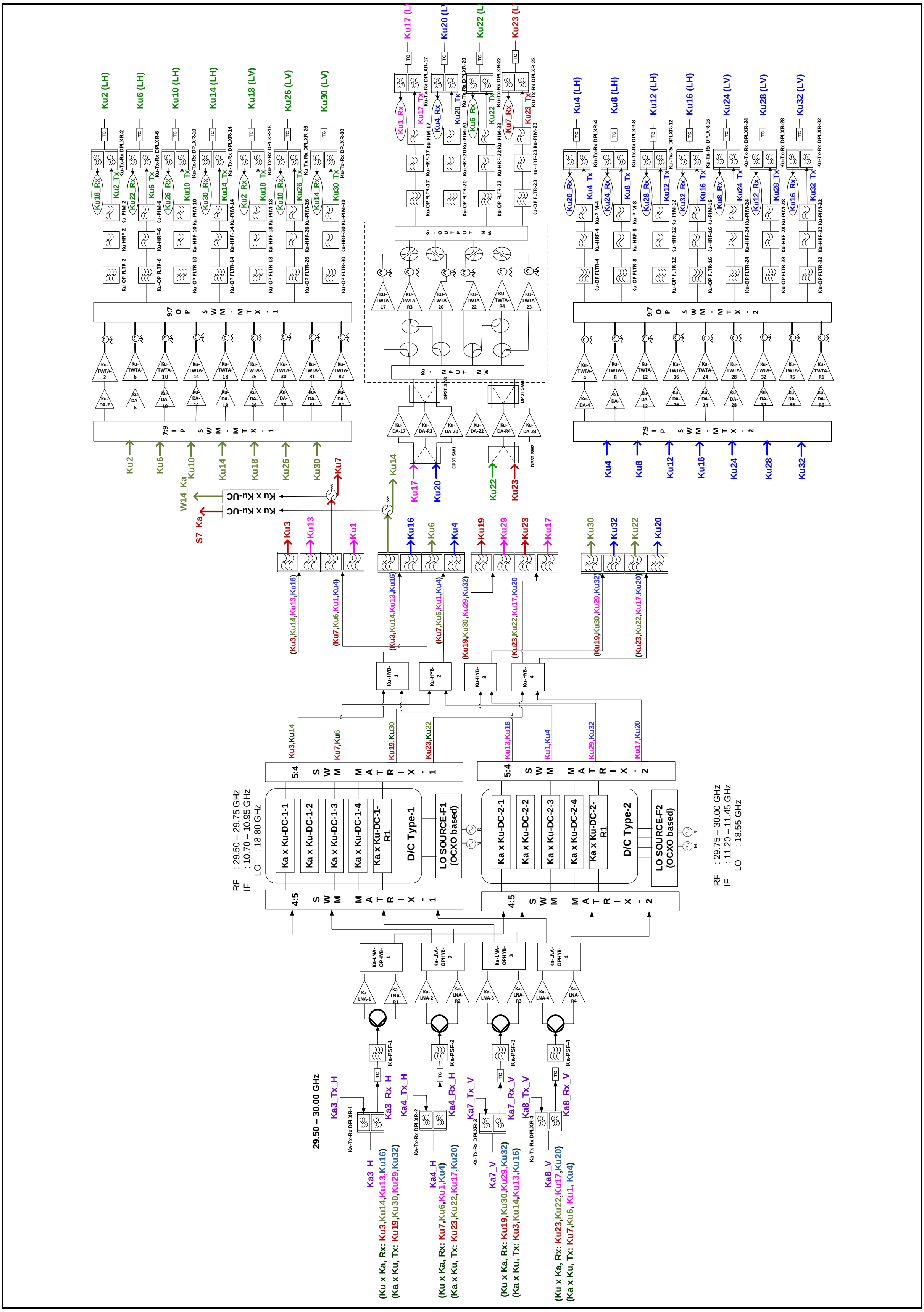


Figure 3-2 Ka x Ku Forward Link-1

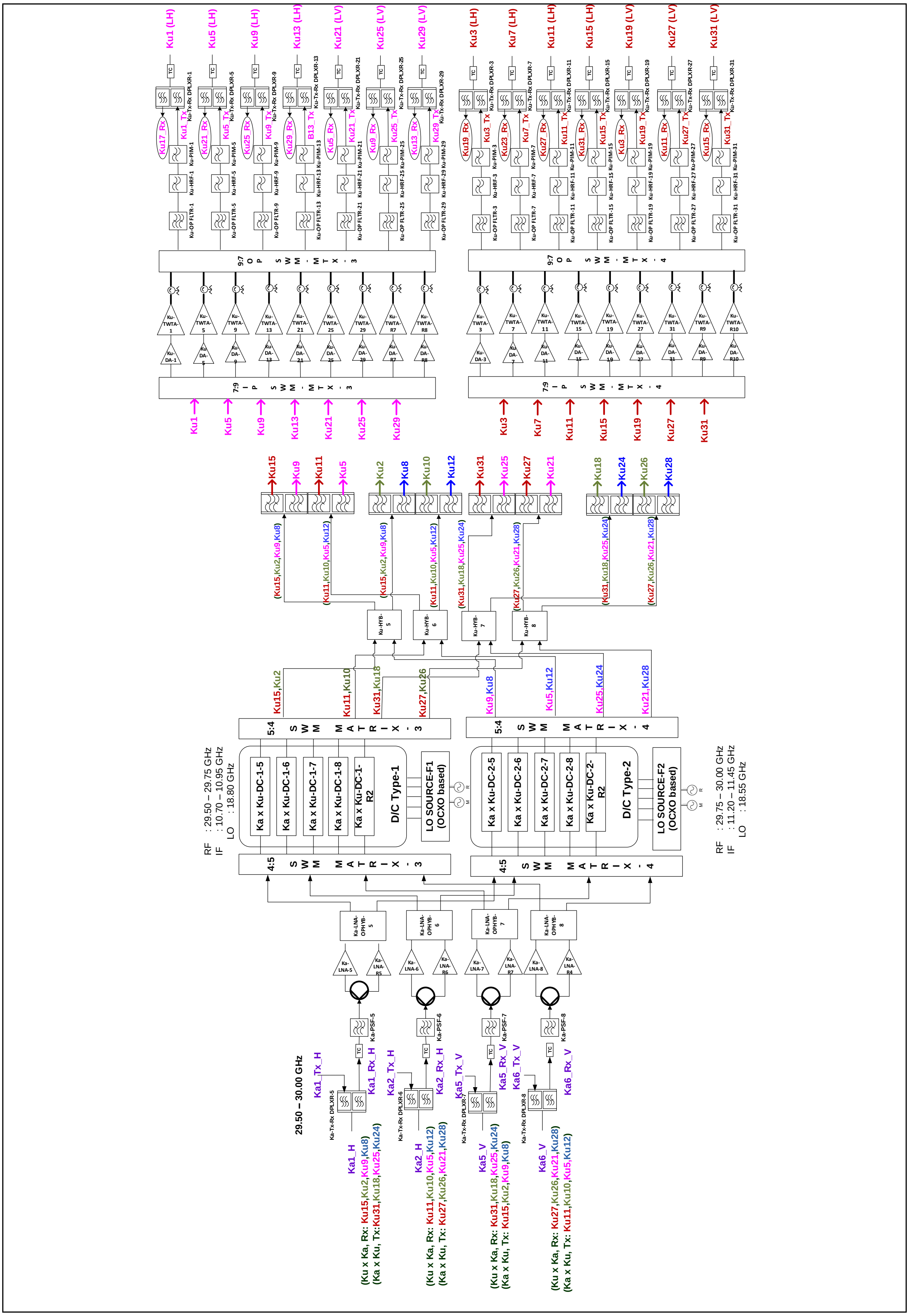


Figure 3-3 Ka x Ku Forward Link-2

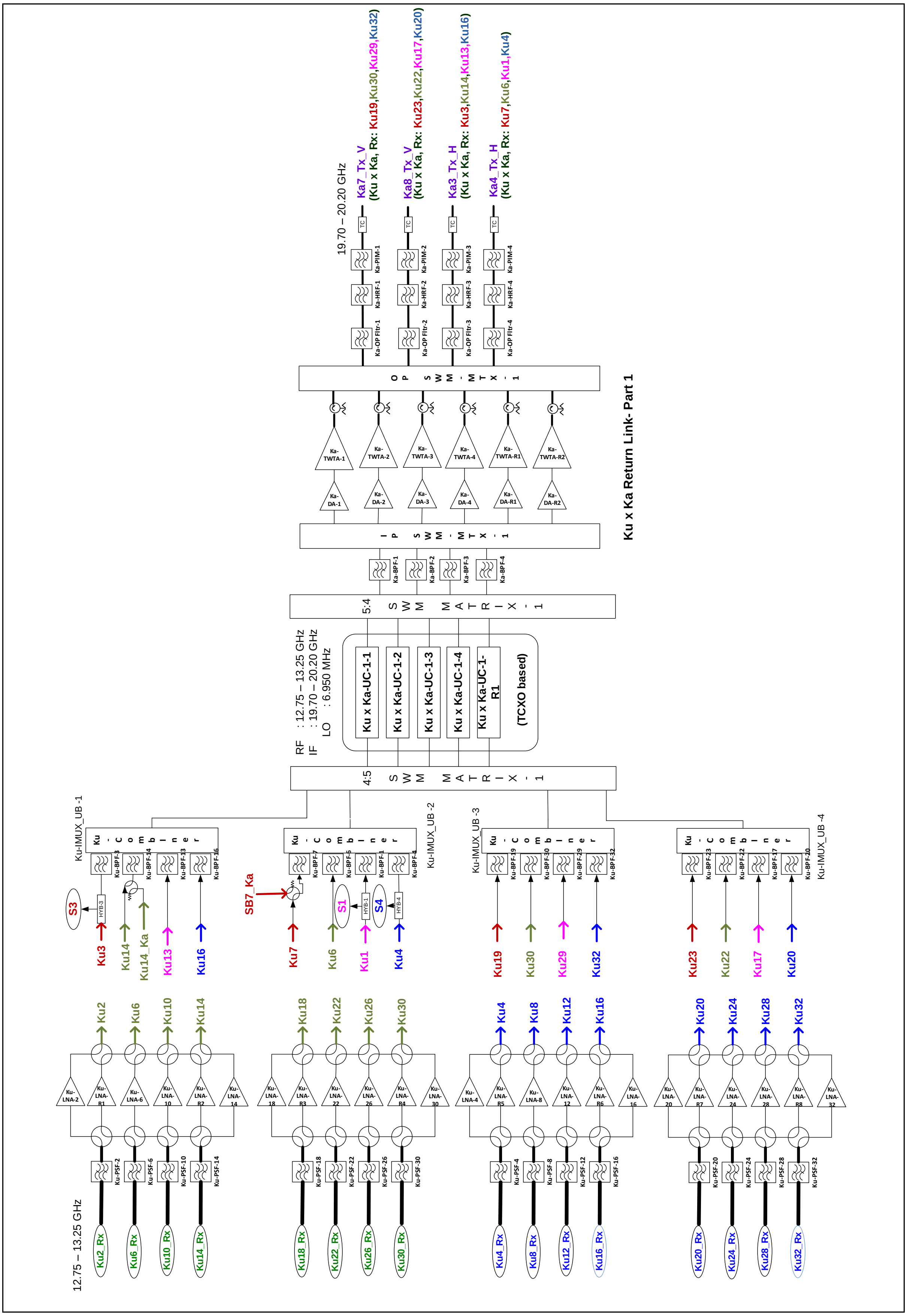


Figure 3-4 Ku x Ka Return Link-1

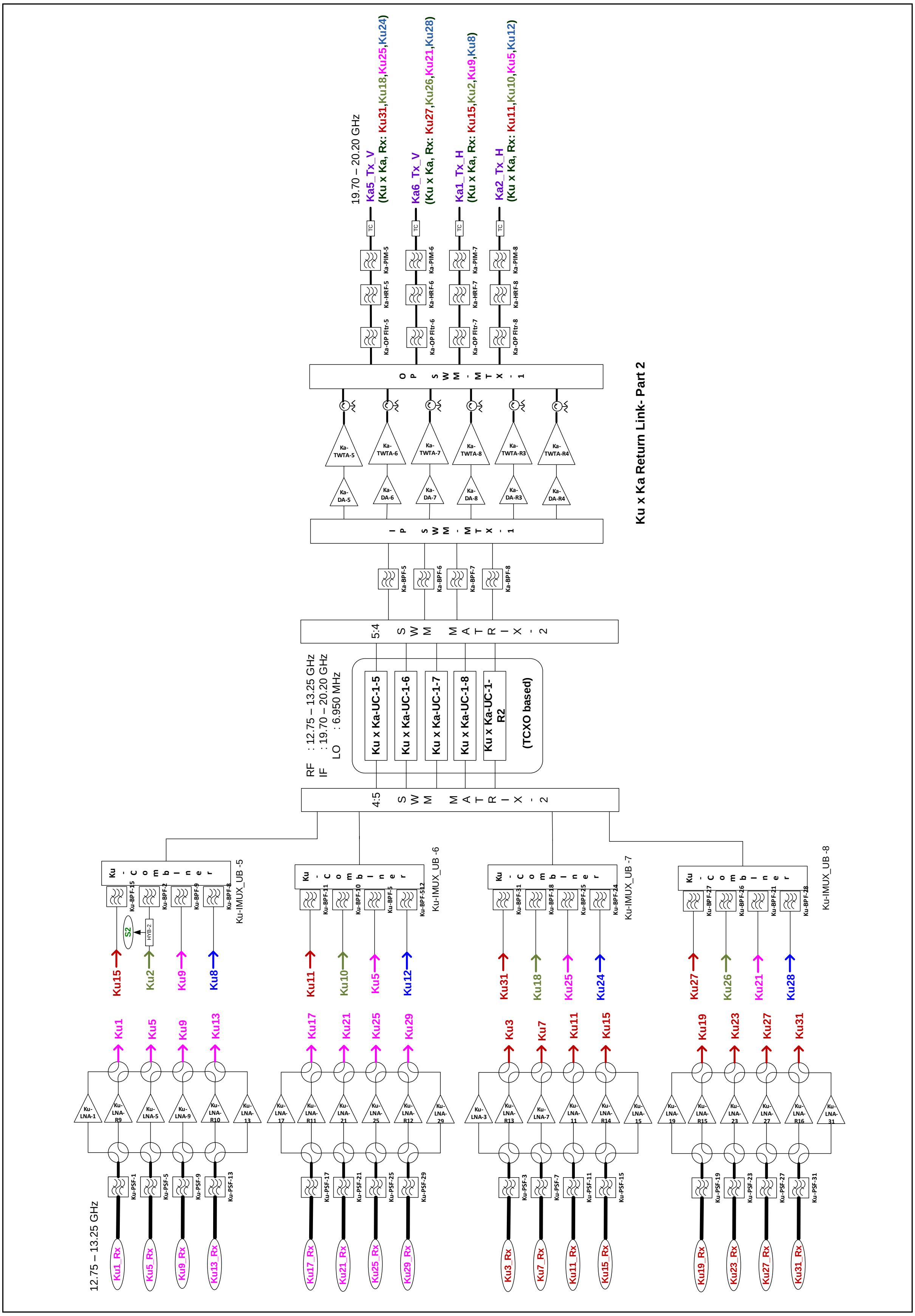


Figure 3-5 Ku x Ka Return Link-2

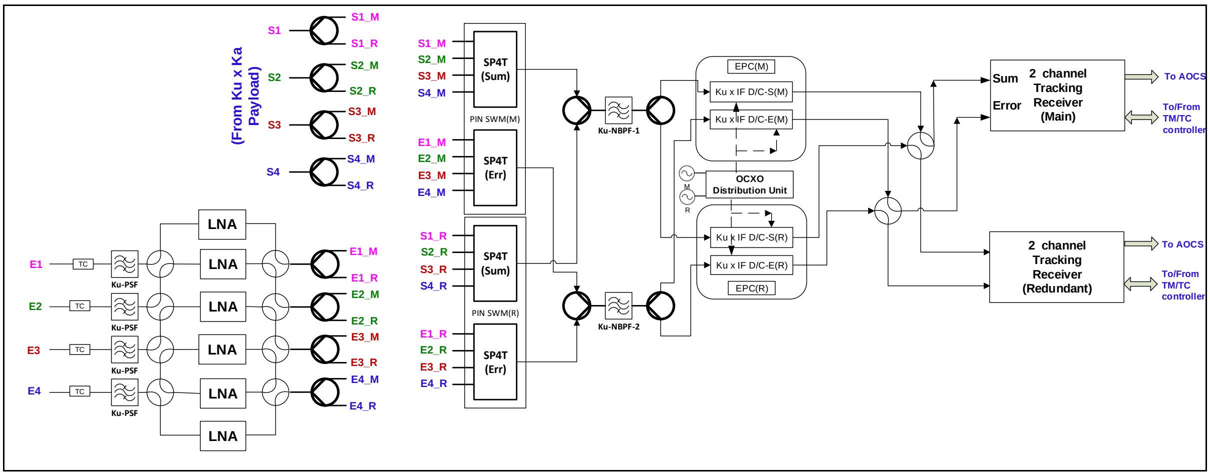


Figure 3-6 Tracking System Configuration

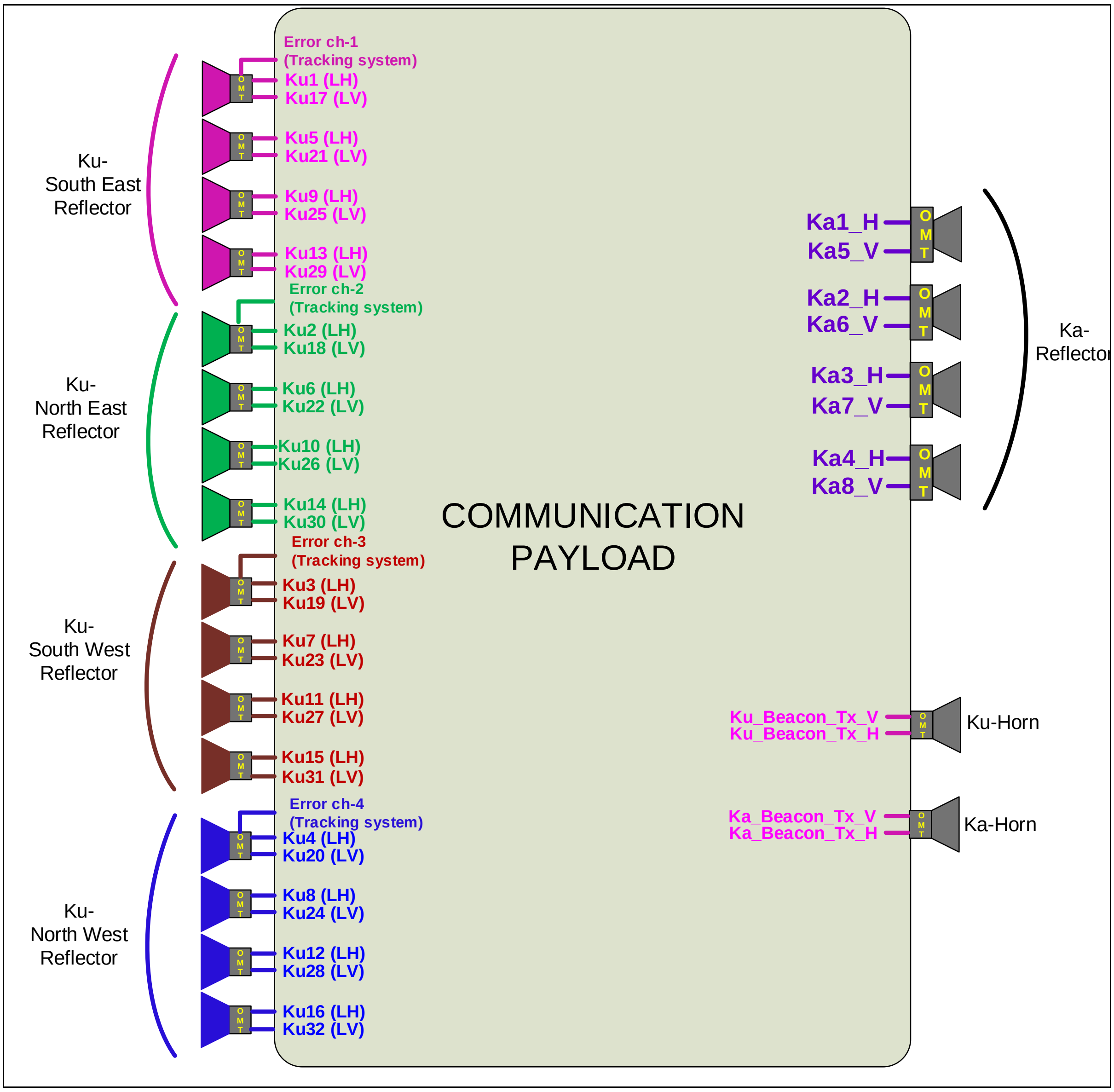


Figure 3-7 Antenna Interface Configuration

3.1.1 User Spectrum

The user uplink and downlink frequencies are (as shown in Figure 3-1) in Ku-band, uplink being 12750 to 13250 MHz and downlink 10700 to 10950 & 11200 to 11450 MHz, utilizing a total band of 500 MHz in uplink as well as in downlink. Effective spectrum of 4 GHz is generated in both Uplink and Downlink frequency bands through 4 color frequency reuse and 2 times polarization reuse.

3.1.2 Hub Spectrum

Each hub is connected with eight user beams (four beams in one polarization + overlying four beams in orthogonal polarization). As shown in Figure 3-1, the uplink and downlink frequencies for hub operations are in Ka-band, uplink being 29500 -30000 MHz and downlink of 19700 – 20200 MHz utilizing a total band of 500 MHz in uplink as well as in downlink. Through the frequency reuse of 4 times and also polarization reuse, the effective BW becomes 4 GHz each in Uplink and Downlink frequency bands.

3.1.3 Uplink Beacon Frequency (Tracking)

Uplink Beacon frequency identified for On-board Antenna tracking system is 12750 MHz in Right Hand Circular Polarization (RHCP). This beacon signal will be up-linked from identified beacon uplink stations located in four beams (i.e. Ku1, Ku2, Ku3 and Ku4).

3.2 Service Coverage Area

GSAT-11 spacecraft with 32 spot beams in Ku-band incorporated frequency reuse and polarization reuse scheme to increase frequency spectrum utilization efficiency. The coverage requirement shall be met for all antenna beam-pointing errors as specified in this document.

Gateway beams in Ka-band named as ‘Ka1’ to ‘Ka4’ represent coverage for Uplink/Downlink in Horizontal polarization. For coverage in the orthogonal polarization i.e in Vertical polarization, corresponding beams are ‘Ka5’ to ‘Ka8’. Thus beam ‘Ka5’ will overlay with beam ‘Ka1’ and so on.

Similarly, beams named as 'B1' to 'B16' represent Ku-Band Coverage Uplink Vertical/Downlink Horizontal polarization. For coverage in the orthogonal polarization i.e. Uplink Horizontal/Downlink Vertical, corresponding beams are 'B17' to 'B32'. Thus beam 'Ku17' will overlay with beam 'Ku1' and so on.

The Transmit and Receive antennas shall have radiation patterns such that all the communication performance specifications shall be met over the service area. The coverage boundary for the service area is defined as follows.

- **Ka-Band Coverage:** 4 spot beams over Indian Main land region in each polarization.
- **Ku-band Coverage:** 14 spot beams to cover Indian Mainland and 1 beam each for Andaman Nicobar and Lakshadweep islands in each polarization.

Table 3-2: Forward Link (Ka x Ku) beam Association

Ka-band		**Ku-beams**	
Beam No.	**Uplink Polarization**	**Beam No.**	**Downlink polarization**
Ka5	Linear–V	B15, B2, B9, B8	Linear–H
Ka6	Linear–V	B11, B10, B5, B12	Linear–H
Ka7	Linear–V	B3,B14, B13, B16	Linear–H
Ka8	Linear–V	B7, B6, B1, B4	Linear–H
Ka1	Linear–H	B31, B18, B25, B24	Linear–V
Ka2	Linear–H	B27, B26, B21, B28	Linear–V
Ka3	Linear–H	B19, B30, B29, B32	Linear–V
Ka4	Linear–H	B23, B22, B17, B20	Linear–V

Table 3-3: Return Link (Ku x Ka) beam association

Ku-band		**Ka-band**	
Beam Sharing the same Transponder For Ku x Ka Payload	**Uplink Polarization**	**Beam No.**	**Downlink Polarization**
B15, B2, B9, B8	Linear–V	Ka1	Linear–H
B11, B10, B5,B12	Linear–V	Ka2	Linear–H
B3,B14, B13, B16	Linear–V	Ka3	Linear–H
B7, B6, B1, B4	Linear–V	Ka4	Linear–H
B31, B18, B25, B24	Linear–H	Ka5	Linear–V
B27, B26, B21, B28	Linear–H	Ka6	Linear–V
B19, B30, B29, B32	Linear–H	Ka7	Linear–V
B23, B22, B17, B20	Linear–H	Ka8	Linear–V

3.3 Frequency Channelization Plan

Ku-band frequencies will be repeated 4 times due to 4 colour reuse and Ka-band frequencies will be repeated 4 times due to 4 times reuse. Ku-band and Ka-band will utilize 500 MHz spectrum in each polarization. Frequency Channelization plan for the Ka x Ku and Ku x Ka transponders is shown in Table 3-4 and Table 3-5.

Table 3-4: Forward Link (Ka x Ku) Frequency Channelization Plan

Channel No.	Ku-beam	Channel	Uplink		Downlink		Usable BW (MHz)
		Designation	Centre freq (MHz)	Pol.	Centre freq (MHz)	Pol.	
1	B1	Ka8 x Ku1	29812	L-V	11262	L-H	116
2	B2	Ka5 x Ku2	29687	L-V	10887	L-H	116
3	B3	Ka7 x Ku3	29562	L-V	10762	L-H	116
4	B4	Ka8 x Ku4	29937	L-V	11387	L-H	116
5	B5	Ka6 x Ku5	29812	L-V	11262	L-H	116
6	B6	Ka8 x Ku6	29687	L-V	10887	L-H	116
7	B7	Ka8 x Ku7	29562	L-V	10762	L-H	116
8	B8	Ka5 x Ku8	29937	L-V	11387	L-H	116
9	B9	Ka5 x Ku9	29812	L-V	11262	L-H	116
10	B10	Ka6 x Ku10	29687	L-V	10887	L-H	116
11	B11	Ka6 x Ku11	29562	L-V	10762	L-H	116
12	B12	Ka6 x Ku12	29937	L-V	11387	L-H	116
13	B13	Ka7 x Ku13	29812	L-V	11262	L-H	116
14	B14	Ka7 x Ku14	29687	L-V	10887	L-H	116
15	B15	Ka5 x Ku15	29562	L-V	10762	L-H	116
16	B16	Ka7 x Ku16	29937	L-V	11387	L-H	116
17	B17	Ka4 x Ku17	29812	L-H	11262	L-V	116
18	B18	Ka1 x Ku18	29687	L-H	10887	L-V	116
19	B19	Ka3 x Ku19	29562	L-H	10762	L-V	116
20	B20	Ka4 x Ku20	29937	L-H	11387	L-V	116
21	B21	Ka2 x Ku21	29812	L-H	11262	L-V	116
22	B22	Ka4 x Ku22	29687	L-H	10887	L-V	116
23	B23	Ka4 x Ku23	29562	L-H	10762	L-V	116
24	B24	Ka1 x Ku24	29937	L-H	11387	L-V	116
25	B25	Ka1 x Ku25	29812	L-H	11262	L-V	116
26	B26	Ka2 x Ku26	29687	L-H	10887	L-V	116
27	B27	Ka2 x Ku27	29562	L-H	10762	L-V	116
28	B28	Ka2 x Ku28	29937	L-H	11387	L-V	116
29	B29	Ka3 x Ku29	29812	L-H	11262	L-V	116
30	B30	Ka3 x Ku30	29687	L-H	10887	L-V	116
31	B31	Ka1 x Ku31	29562	L-H	10762	L-V	116
32	B32	Ka3 x Ku32	29937	L-H	11387	L-V	116

Table 3-5: Return Link (Ku x Ka) Frequency Channelization Plan

Channel No.	Ku-beam	Channel Designation	Uplink Centre freq (MHz)	Uplink Pol.	Downlink Centre freq (MHz)	Downlink Pol.	Usable BW (MHz)
33	B1	Ku17 x Ka4	13062	L-V	20012	L-H	116
34	B2	Ku18 x Ka1	12937	L-V	19887	L-H	116
35	B3	Ku19 x Ka3	12812	L-V	19762	L-H	116
36	B4	Ku20 x Ka4	13187	L-V	20137	L-H	116
37	B5	Ku21 x Ka2	13062	L-V	20012	L-H	116
38	B6	Ku22 x Ka4	12937	L-V	19887	L-H	116
39	B7	Ku23 x Ka4	12812	L-V	19762	L-H	116
40	B8	Ku24 x Ka1	13187	L-V	20137	L-H	116
41	B9	Ku25 x Ka1	13062	L-V	20012	L-H	116
42	B10	Ku26 x Ka2	12937	L-V	19887	L-H	116
43	B11	Ku27 x Ka2	12812	L-V	19762	L-H	116
44	B12	Ku28 x Ka2	13187	L-V	20137	L-H	116
45	B13	Ku29 x Ka3	13062	L-V	20012	L-H	116
46	B14	Ku30 x Ka3	12937	L-V	19887	L-H	116
47	B15	Ku31 x Ka1	12812	L-V	19762	L-H	116
48	B16	Ku32 x Ka3	13187	L-V	20137	L-H	116
49	B17	Ku1 x Ka8	13062	L-H	20012	L-V	116
50	B18	Ku2 x Ka5	12937	L-H	19887	L-V	116
51	B19	Ku3 x Ka7	12812	L-H	19762	L-V	116
52	B20	Ku4 x Ka8	13187	L-H	20137	L-V	116
53	B21	Ku5 x Ka6	13062	L-H	20012	L-V	116
54	B22	Ku6 x Ka8	12937	L-H	19887	L-V	116
55	B23	Ku7 x Ka8	12812	L-H	19762	L-V	116
56	B24	Ku8 x Ka5	13187	L-H	20137	L-V	116
57	B25	Ku9 x Ka5	13062	L-H	20012	L-V	116
58	B26	Ku10 x Ka6	12937	L-H	19887	L-V	116
59	B27	Ku11 x Ka6	12812	L-H	19762	L-V	116
60	B28	Ku12 x Ka6	13187	L-H	20137	L-V	116
61	B29	Ku13 x Ka7	13062	L-H	20012	L-V	116
62	B30	Ku14 x Ka7	12937	L-H	19887	L-V	116
63	B31	Ku15 x Ka5	12812	L-H	19762	L-V	116
64	B32	Ku16 x Ka7	13187	L-H	20137	L-V	116

Table 3-6: Ka x Ka (Switchable) Frequency Channelization Plan

Channel No.	Channel Designation	Uplink Centre Freq (MHz)	Uplink Pol.	Downlink Centre Freq (MHz)	Downlink Pol.	Usable BW (MHz)
7/46	Ka8 x Ka3	29562	L-V	19887	L-H	116
14/39	Ka7 x Ka4	29687	L-V	19762	L-H	116

Note *:" Channel Designation" may be read as "Uplink Freq. band & Associated Coverage Beam number x Downlink Freq. Band & Associated Coverage Beam number ". Thus, e.g. Ka1 x Ku15 indicates a channel that operates with uplink in Ka-Band through Beam No.1 and downlink in Ku-Band through Beam No. 15.

Table 3-7:Ground Beacon Transmitter Frequency

Beacon	Transmit Frequency (MHz)	Polarization
Ku-band Ground Beacon (for On-Board Tracking System)	12750	RHCP

3.4 On-board Power Sharing

Four of the User Beams i.e. B17, B20, B22 and B23 in Ka x Ku Forward link are configured in Multiport Amplifier configuration to share the power among them. These beams shall be controlled by Ka4 Hub.

3.5 Polarization

3.5.1 Ka-Band

The uplink and down link signals in Ka-Band shall operate in the linear polarization. The sense of polarization will be as defined below:

- **Receive** : 29500 – 30000 MHz (Linear Vertical & Linear Horizontal)
- **Transmit** : 19700 – 20200 MHz (Linear Vertical & Linear Horizontal)

3.5.2 Ku-Band

The uplink and down link signals in Ku-Band shall operate in the linear polarization. The sense of polarization will be as defined below:

- Receive : 12750 – 13250 GHz (Linear Vertical & Linear Horizontal)
- Transmit : 10700 – 10950 MHz, 11200 – 11450 MHz (Linear Vertical & Linear Horizontal)

Notes:

a) Receive E field vector shall be Vertical, defined as being parallel to the spacecraft's nominal pitch axis.

b) Transmit E field vector shall be Horizontal, defined as being parallel to the spacecraft's nominal roll axis.

3.6 Multibeam Isolation (Composite C/I)

The beam to beam isolation for Ku-band and Ka-band transmit as well as receive beams among same frequency beams shall be as per Table 3-8. This includes contribution from Co-polarization as well Cross polarization components of same frequency beams.

Table 3-8: Multibeam composite C/I

Multi-beam System C/I (dB)	Ka x Ku Forward Link	Ku x Ka Return Link	Ka x Ka Return Link
(over < 60% Beam Area) - User Beams - Hub Beams	 > 18.5 > 18.5	 > 18.5 > 18.5	 > 18.5 > 18.5
(over < 100% Beam Area) - User Beams - Hub Beams	 > 14 > 14	 > 14 > 14	 > 14 > 14

3.7 Gain

3.7.1 Saturation Flux Density and Channel Gain Settings

Ka x Ku Payload

When any of Ka x Ku transponder is set to the maximum gain and the spacecraft is illuminated with a flux density of –96 ± 2 dBW/m^2 (at the centre frequency of the beams as per Table 3-4), by a single carrier of appropriate polarization in the direction of minimum antenna gain (anywhere in the service area), the derived effective isotropic radiated power (EIRP) in that beam shall be equivalent to saturated EIRP of beam.

When the beams configured in multiport amplifier configuration (Refer Section 3.4) are illuminated by four equal amplitude RF carriers at centre frequencies of the beams, with a flux density of −102 ± 2 dBW/m2 per carrier of appropriate polarization in the direction of minimum antenna gain (anywhere in the service area), the derived effective isotropic radiated power (EIRP) in that beam shall be equivalent to 3 dB below the saturated EIRP of beam.

When any one of the beams configured in multiport amplifier configuration is illuminated by a single RF carrier at centre frequency of the respective beam, with a flux density of -96 ± 2 dBW/m2 of appropriate polarization in the direction of minimum antenna gain (anywhere in the service area), the derived effective isotropic radiated power (EIRP) in that beam shall be equivalent to 3 dB above the saturated EIRP of beam.

Provision shall be made to permit setting of gain of each channel by ground command, by inserting attenuation into transmission path. For each channel the variable attenuator shall have variations between 0 to 16 dB in steps of 1 dB. For any setting of the attenuator the tolerance shall be within ±0.5 dB, over its nominal value.

Ku x Ka Payload

When any of Ku x Ka transponder is set to the maximum gain and the spacecraft is illuminated with a flux density of −96 ± 2 dBW/m^2 (at the centre frequency of the beams as per Table 3-5), by a single carrier of appropriate polarization in the direction of minimum antenna gain (anywhere in the service area), the output power amplifier in that beam shall be driven to the rated output power of the transponder.

When any of the Ku x Ka transponder is illuminated by four equal amplitude RF carriers at centre frequencies of the user beams sharing the same transponder (Refer Table 3-3), with a flux density of −108 ± 2 dBW/m2 per carrier of appropriate polarization in the direction of minimum antenna gain (anywhere in the service area), the output power amplifier in that beam shall be driven to 3 dB below the rated output power of the transponder.

Provision shall be made to permit setting of gain of each channel by ground command, by inserting attenuation into transmission path. For each channel the variable attenuator shall have variations between 0 to 16 dB in steps of 1 dB. For any setting of the attenuator the tolerance shall be within ±0.5 dB, over its nominal value.

Ka x Ka Payload

When a channel is set to the maximum gain and the spacecraft is illuminated with a flux density of −96 ± 2 dBW/m^2 (at the centre frequency of the channel), by a single carrier of appropriate polarization in the direction of minimum antenna gain (anywhere in the service area) with none of the other three Ku x Ka channels sharing the same transponder are illuminated, the output power amplifier in that channel shall be driven to the rated output power in the channel. Provision shall be made to permit setting of gain of each channel by ground command, by inserting attenuation into transmission path. For each channel the variable attenuator shall have variations between 0 to 16 dB in steps of 1 dB. For any setting of the attenuator the tolerance shall be within ±0.5 dB, over its nominal value.

When one of the Ku x Ka transponder sharing power between Ka x Ka and Ku x Ka links is illuminated by four equal amplitude RF carriers at center frequencies of the user beams sharing the same transponder (Refer Table 3-3), with a flux density of –108 ± 2 dBW/m2 per carrier of appropriate polarization in the direction of minimum antenna gain (anywhere in the service area), the output power amplifier in that beam shall be driven to 3 dB below the rated output power of the transponder.

3.7.2 Automatic Level Control (ALC)

With the ALC mode ON, the variation in the transponder output power over the specified SFD range of -102± 2 to -80± 2 dBW/m^2 of ALC mode shall be less than 1 dB-pp. ALC function can be switched ON/OFF using the Tele-command signals. Under operational conditions, Transponder to operate in ALC OFF mode.

3.7.3 Small Signal Gain Stability

The change in small signal gain at the centre frequency of each transmission channel due to all causes, excluding antenna pointing stability, shall not exceed 2 dB peak to peak over any operating day and 4 dB peak to peak over the design life.

The in-orbit diurnal temperature variations as being experienced on the existing spacecraft shall be considered for the purpose of establishing the test-results compliance.

3.8 Receive System Gain-to-Noise-Temperature (G/T) Ratio

The receive G/T of the channel, measured at any frequency within the receive band shall not be lower than the values as specified in Table 3-9 over specified coverage area, under all channel loading and gain setting conditions.

Table 3-9: Receive G/T

Receive G/T (dB/K)	Ka x Ku Forward Link	Ku x Ka Return Link	Ka x Ka Return Link
over < 60% Beam Area	> 14.5	> 13	> 14.5
over < 100% Beam Area	> 14.5	> 11	> 14.5

The above value of the G/T ratio is referred to the interface between receive antenna and the pre-amplifiers. The noise temperature involved in this ratio is the total system noise temperature, which includes, noise received by the antenna when properly oriented in orbit which shall be taken (as 300 K), the noise of the pre-amplifier, and noise of all proceeding and succeeding elements in the transmission channel.

3.8.1 G/T Stability

Excluding Antenna Effects: At any fixed point within the receive beams of the channels, the change in receive G/T, at the centre frequency of any channel, shall not exceed 1.0 dB peak to peak over any operating day and 1.5 dB peak to peak over the design life of the spacecraft, due to all causes except antenna gain stability due to thermal and platform pointing variations.

Including Antenna Effects: At any fixed point within the receive beams of the channels, the change in receive G/T, at the centre frequency of any channel, shall not exceed 3.0 dB peak to peak over any operating day and 3.5 dB peak to peak over the design life of the spacecraft, due to all causes including antenna gain stability due to thermal and platform pointing variations.

The in-orbit diurnal temperature variations as being experienced on the existing spacecraft shall be considered for the purpose of establishing the test-results compliance.

3.9 Transmit EIRP

The EIRP level per transponder/beam and per Channel in case of Ka x Ku , over coverage area under single carrier saturation shall be as per Table 3-10.

Table 3-10: Saturated EIRP

Saturated EIRP (dBW)	Ka x Ku Forward Link	Ku x Ka Return Link	Ka x Ka Return Link
over < 60% Beam Area	> 61	> 60.5	> 60.5
over < 100% Beam Area	> 59	> 60.5	> 60.5

For the user beams sharing power with Multiport configuration in Ka x Ku -channels, the derived nominal (input flux density of -102 dBW/m2 /carrier for each of the four beams) and maximum EIRP (input flux density of -96 dBW/m2/carrier for any one beam) will be as per Table 3-11 below:

Table 3-11: EIRP for Beams sharing power with Multiport amplifier

EIRP (dBW)	Nominal	Maximum
over < 60% Beam Area	> 58	> 64
over < 100% Beam Area	> 56	> 62

Further, since each of the Ku x Ka transponders (Ka-beams) will be shared by four channels/ user beams , when any of the Ku x Ka transponders is illuminated by four equal amplitude RF carriers at center frequencies of the channels/user beams given as per Table 3-3, and input flux density of -108 dBW/m2, the Nominal EIRP per channel/carrier shall be achieved as per Table 3-12. The performance shall be met with intermodulation product levels below 16 dBc (Refer Section 3.12.2)

Table 3-12: Nominal EIRP in Ku x Ka & Ka x Ka channels

Nominal EIRP per Channel /Carrier (dBW)	**Value**
over < 60% Beam Area	> 51.5
over < 100% Beam Area	> 51.5

The same applies to Ka x Ka link which will share transponder with Ku x Ka link channels.

The above Transmit EIRP specification shall be met at the end of 15 years mission life, including the antenna beam pointing error of ±0.05°. The above EIRP value does not include variations due to the EIRP instability given in Section 3.8.1.

Table 3.17 gives a summary of SFD and Nominal/Saturated EIRP as described in above sections.

Table 3-13: Summary of Transponder SFD & EIRP

Excitation	Ka x Ku Forward Link		Ka x Ku MPA Forward Link		Ku x Ka Return Link		Ka x Ka Return Link	
	Input Flux Density dBW/m²	Nominal EIRP (100% Beam Area) dBW	Input Flux Density dBW/m²	Nominal EIRP (100% Beam Area) dBW	Input Flux Density dBW/m²	Nominal EIRP (100% Beam Area) dBW	Input Flux Density dBW/m²	Nominal EIRP (100% Beam Area) dBW
Single Carrier under Saturation at Centre Frequency of Beam	-96.0 ± 2.0	59.0	-	-	-	-	-	-
Four Carriers with equal amplitude at centre frequency of each of the beams 17, 20, 22, 23	-	-	-102.0 ± 2.0	56.0	-	-	-	-
Single Carrier under Saturation at Centre Frequency of any one of Beams 17, 20, 22, 23	-	-	-96.0 ± 2.0	62.0	-	-	-	-
Four Carriers of equal amplitude at centre frequencies of the user beams sharing the same transponder (Refer Table 3-3)	-	-	-	-	-108.0 ± 2.0	51.5 (per carrier)	-	-
Single Carrier under Saturation at Centre Frequency of any one of user beams sharing the same transponder (Refer Table 3-3)	-	-	-	-	-96.0 ± 2.0	60.5	-	-
Four Carriers of equal amplitude at centre frequencies of the Ka x Ka link & three user beams sharing the same transponder (Refer Table 3-3)	-	-	-	-	-	-	-108.0 ± 2.0	51.5 (per carrier)
Single Carrier under Saturation at Centre Frequency of any one of Ka x Ka link or any one of the three user beams sharing the same transponder (Refer Table 3-3)	-	-	-	-	-	-	-96.0 ± 2.0	60.5

3.9.1 EIRP Stability

Excluding Antenna Effects: At any fixed point within the transmit beams of the channels, the change in saturated EIRP, at the centre frequency of any channel, shall not exceed 0.5 dB peak to peak over any operating day and 1 dB peak to peak over the design life of the spacecraft, due to all causes except antenna gain stability due to thermal and platform pointing variations.

Including Antenna Effects: At any fixed point within the transmit beams of the channels, the change in saturated EIRP, at the centre frequency of any channel, shall not exceed 2.5 dB peak to peak over any operating day and 3.5 dB peak to peak over the design life of the spacecraft, due to all causes including antenna gain stability due to thermal and platform pointing variations.

The in-orbit diurnal temperature variations as being experienced on the existing spacecraft shall be considered for the purpose of establishing the test-results compliance.

3.10 Band Width and Gain Flatness

3.10.1 Usable Bandwidth

The centre frequency as well as the usable bandwidth of Ka x Ku channels and Ku x Ka channels shall be as given in Table 3-4 and Table 3-5, respectively.

3.10.2 In-Band Frequency Response

The EIRP of any transmission channel at any specific point within the antenna coverage beam shall not vary by more than the value shown in Table 3-14 and Table 3-15 respectively. This specification applies for any gain step attenuator value of the operating ranges. Specifications over 80% Bandwidth shall be met in multipath environment.

Table 3-14 : In-Band Large Signal Frequency Response

In-bands frequency Response (dB-pp)	Ka x Ku Band Forward Link	Ku x Ka Band Return Link	Ka x Ka Band Link
Over 80 % of channel BW	0.8	0.8	0.8
Over 90 % of channel BW	1.2	1.2	1.2
Over 100 % of channel BW	1.5	1.5	1.5

Table 3-15 : In-Band Small Signal Frequency Response

In-bands frequency Response (dB-pp)	Ka x Ku Band Forward Link	Ku x Ka Band Return Link	Ka x Ka Band Link
Over 80 % of channel BW	1.3	1.7	2.2
Over 90 % of channel BW	2.0	2.2	2.8
Over 100 % of channel BW	2.7	2.7	3.5

3.10.3 Gain Slope

The maximum gain slope within the usable bandwidth of any transmission channel shall not exceed the values given in Table 3-16 and Table 3-17 respectively. Input gain slope is measured between the input to the transponder and the input to the final power amplifier and includes the contribution of the receive antenna. The total gain slope is the performance of the complete channel including receive and transmit antennas. Specification over 80% Band Width shall be met in multipath environment.

Table 3-16 : Input Gain Slope

Gain Slope (dB/MHz)	Ka x Ku Band Forward Link	Ku x Ka Band Return Link	Ka x Ka Band Link
Over 80 % of channel BW	0.15	0.15	0.25
Over 90 % of channel BW	0.30	0.30	0.35
Over 100 % of channel BW	0.70	0.70	0.75

Table 3-17 : Total Gain Slope

Gain Slope (dB/MHz)	Ka x Ku Band Forward Link	Ku x Ka Band Return Link	Ka x Ka Band Link
Over 80 % of channel BW	0.5	0.5	0.5
Over 90 % of channel BW	0.9	0.9	0.9
Over 100 % of channel BW	1.3	1.3	1.3

3.11 AM-PM Transfer Coefficient

The transponder channel AM/PM Transfer coefficient shall not exceed the limits shown in Table 3-18 for any illumination level up to saturation. The transfer coefficient is measured with two carriers having an amplitude difference of up to 20 dB, with the larger carrier amplitude being modulated to a depth of 1 dB with a 1 KHz tone. The specification shall be met under the following conditions:

1. At any frequency within the transmission channels.
2. With any frequency separation between carriers within the channel bandwidth down to a minimum of 1.0 MHz.

Table 3-18: AM-PM Transfer Coefficients

Flux density of two carriers, below the Single Carrier SFD	**AM-PM transfer Coefficient, deg/dB**		
	Ka x Ku	**Ku x Ka**	**Ka x Ka**
0 dB	8	8	8
3 dB	6	6	6
6 dB	4	4	4
9 dB	3	3	3

3.12 Linearity

3.12.1 Linearity of Receive Section

The ratio of carrier to each third order inter-modulation product in the Receive section shall be as given in Table 3-19, when the spacecraft is illuminated with two equal amplitude carriers.

Table 3-19: Third Order Inter-modulation Performance of common input section

Flux Density at the Spacecraft for Each of two Carriers, dBW / m^2			**Third Order Intermodulation Level dBc**
Ka x Ku	**Ku x Ka**	**Ka x Ka**	
-80	-80	-80	-44
-85	-85	-85	-54
-90	-90	-90	-64

3.12.2 Linearity of Complete Transmission Channel

When any of the transponder channels is illuminated by two equal amplitude RF carriers, the level of the third order inter-modulation product shall not exceed the values as given in the Table 3-20. The performance shall be met with any separation of the carriers with a minimum of 100 kHz, but within the usable bandwidth of the channels.

Table 3-20: Inter-modulation Levels

Flux density of each of two carriers, below single carrier saturation	**Third order IMP level (dBc)**		
	Ka x Ku Transponders	**Ku x Ka Transponder**	**Ka x Ka Transponder**
3 dB	-11	-11	-11
6 dB	-16	-16	-16
10 dB	-26	-26	-26
17 dB	-32	-32	-32

3.12.3 Noise Power Ratio

When any of the transponder channels is illuminated by multiple equal amplitude RF carriers, the level of the Noise power ratio shall not exceed the values as given in the Table 3-21.

Table 3-21: Noise Power Ratio

Transponder Output Back-off	**Noise Power Ratio (dB)**		
	Ka x Ku Transponders	**Ku x Ka Transponder**	**Ka x Ka Transponder**
3 dB	14	14	14
4 dB	18	18	18
6 dB	23	23	23
9 dB	25	25	25

3.13 Over Drive Capability

All the channels shall be designed to withstand for a maximum of two hours without subsequent degradation of performance over the specified time, single or multi-carrier illumination flux densities up to 20 dB in excess of those required for the transmission channel saturation at the lowest gain setting. Other performance specifications need not be met in the affected channel during the condition of over drive.

3.14 Group Delay

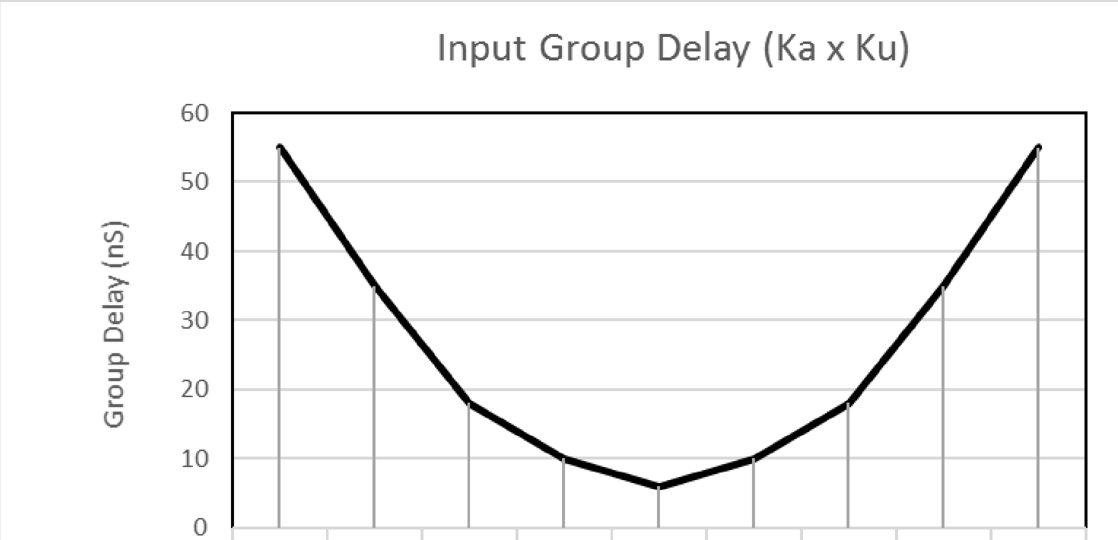
The following Group Delay specifications shall be met over the usable bandwidth when the spacecraft is illuminated with flux densities equal to and less than that required to produce saturation. All the group delay values are relative with respect to the value at the center of the channel.

3.14.1 Input Group Delay

It is measured from the transponder input to the input to the high power amplifier inclusive of receive antenna. The Input Group Delay mask is given in Figure 3-8, Figure 3-9 and Figure 3-10. The Input Group Delay shall be within the specified limits for illumination flux densities from saturation to linear operation of the channels.

3.14.2 Total Group Delay

It is measured through the complete transponder, inclusive of transmit and receive antennas. The Total Group Delay mask is given in Figure 3-11, Figure 3-12 and Figure 3-13. The Total Group Delay shall be within the specified limits for any illumination flux densities from saturation to linear operation of the channels. Specification over 80% Bandwidth shall be met in multipath environment.



	58	52	40	28	0	28	40	52	58
Input GD	55	35	18	10	6	10	18	35	55

Figure 3-8: Ka x Ku Transponder Input Group delay mask

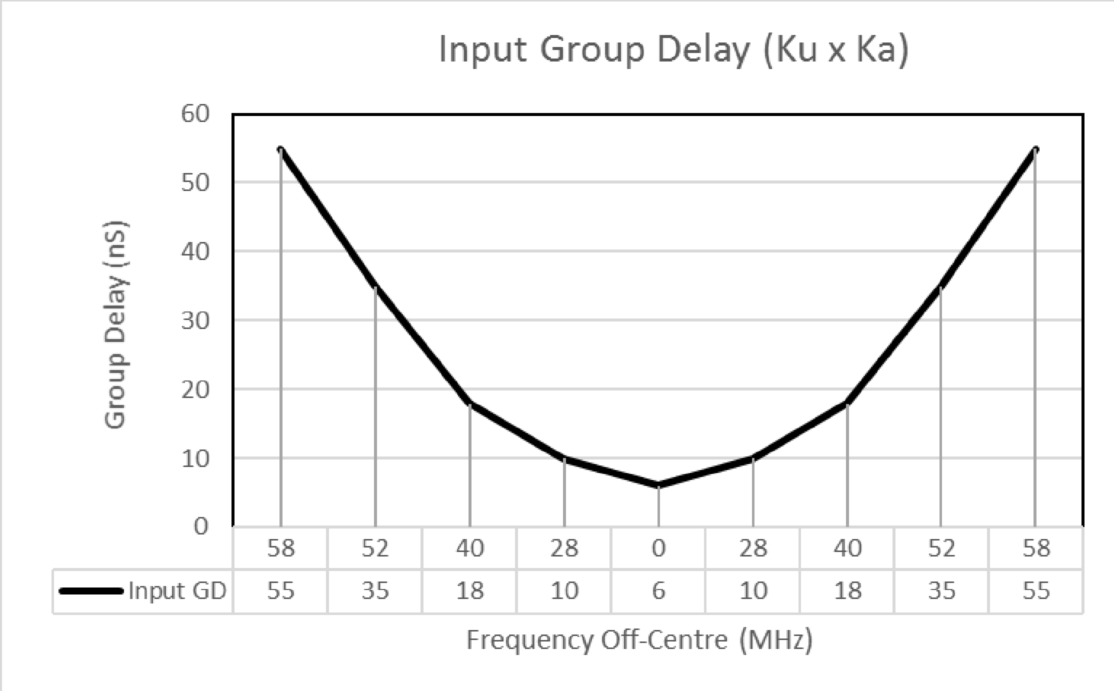


Figure 3-9 Ku x Ka Transponder Input Group delay mask

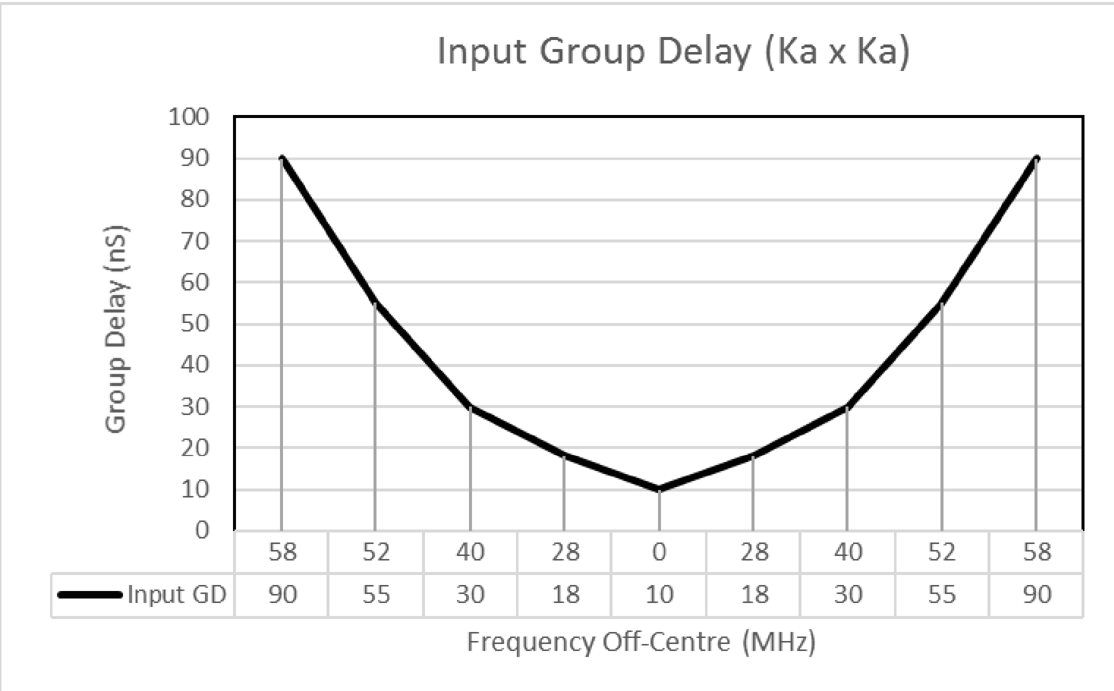
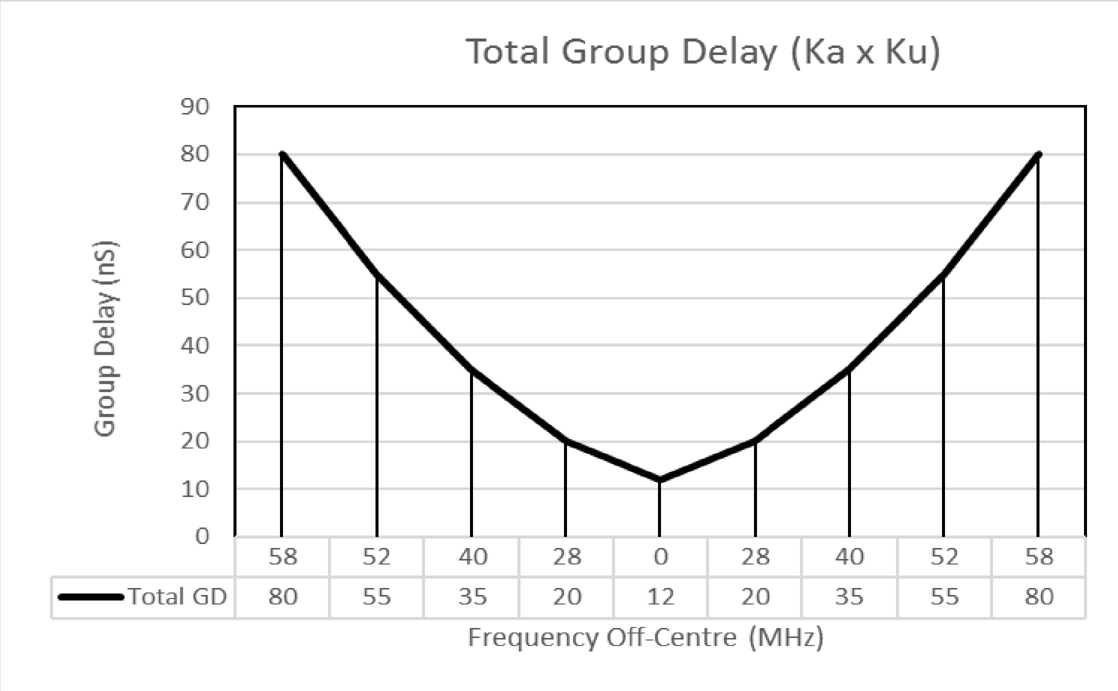
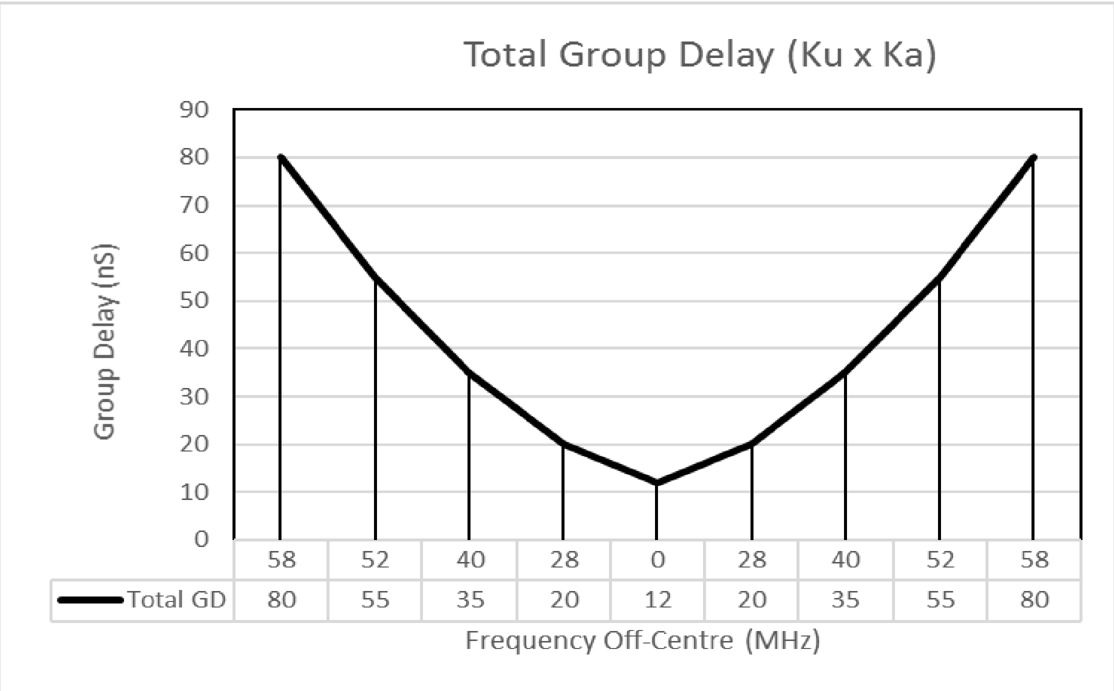


Figure 3-10 Ka x Ka Transponder Input Group delay mask



	58	52	40	28	0	28	40	52	58
Total GD	80	55	35	20	12	20	35	55	80

Figure 3-11 Ka x Ku Transponder Total Group Delay Mask



	58	52	40	28	0	28	40	52	58
Total GD	80	55	35	20	12	20	35	55	80

Figure 3-12 Ku x Ka Transponder Total Group Delay Mask

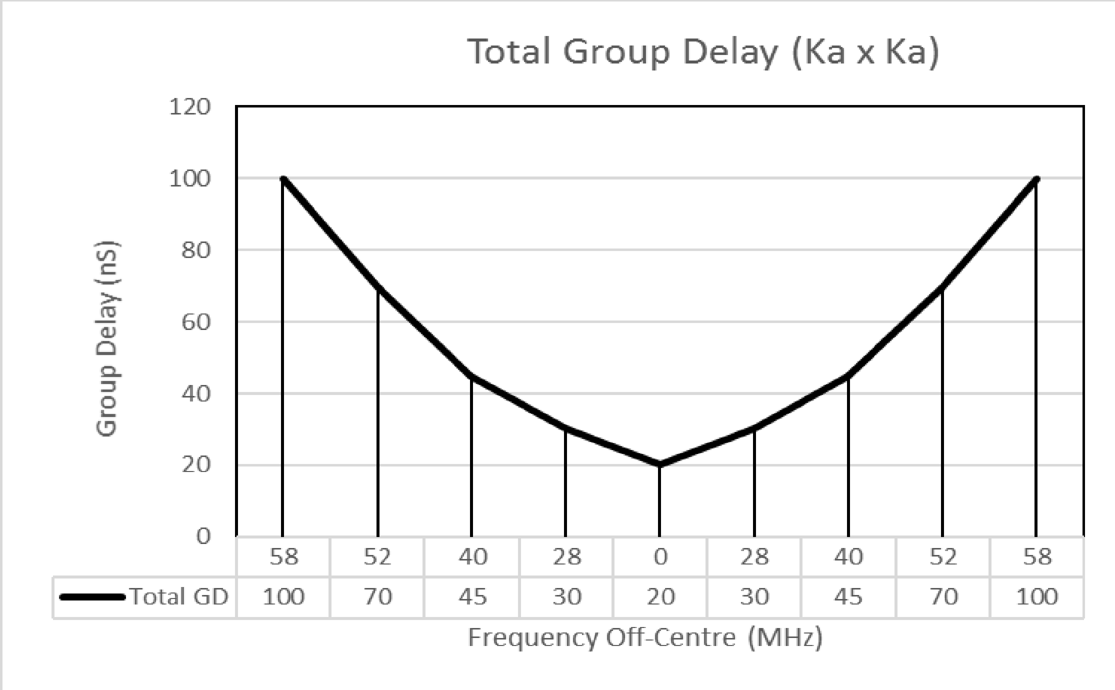


Figure 3-13 Ka x Ka Transponder Total Group Delay Mask

3.14.3 Group Delay Ripple

Total Group Delay ripple over each of the transmission channels including receive and transmit antennas shall not exceed 6 ns peak to peak over the usable bandwidth, under all gain settings. Ripple is defined as having a period of 10 MHz or less.

3.15 Frequency Conversion

Ka x Ku Payload: The Communication Subsystem shall translate receive frequencies in the Ka-band to transmit frequencies in the Ku-band by a net frequency translation of 18800 MHz and 18550 MHz. Frequency inversion from input to output shall not take place.

Ku x Ka Payload: The Communication Subsystem shall translate receive frequencies in the Ku-band to transmit frequencies in the Ka-band by a net frequency translation of 6950 MHz. Frequency inversion from input to output shall not take place.

Ka x Ka Payload: The Communication Subsystem shall translate receive frequencies in the Ku-band to transmit frequencies in the Ka-band by a net frequency translation of 9675 MHz and 9925 MHz. Frequency inversion from input to output shall not take place.

The frequency conversion performance shall have the following limits:

a) The net translation error including initial tolerance shall not exceed the values mentioned in Table 3-22.

Table 3-22 : Frequency Translation Error (ppm)

	Frequency Translation Error (ppm)		
	Ka x Ku	**Ku x Ka**	**Ka x Ka**
Over one-month period, which includes eclipse season	0.9	2.5	2.5
Over the operating lifetime of the satellite	3.5	10	10

b) The spurious phase noise induced on single un-modulated carrier shall not exceed the single sideband noise density specified in Table 3-23.

Table 3-23 : Spurious Phase Noise

Offset from Carrier	**SSB phase noise density, dBc/Hz**
10 Hz	-33
100 Hz	-62
1.0 kHz	-81
2.0 kHz	-84
10.0 kHz	-84
100 kHz	-94
1.0 MHz	-94

3.16 Narrow Band Out of Band Response

3.16.1 Receive Out of Band Response

This parameter is measured between the input to the transmission channel including the receive antenna and the input to the Driver / Power amplifier and shall not exceed the values given in Table 3-24.

Table 3-24: Receive Out of Band Response

Frequency from Channel Centre (MHz)	Receive Out of Band Response (dBc)		
	Ka x Ku Transponders	**Ku x Ka Transponder**	**Ka x Ka Transponder**
± 67.0	> 17	> 17	> 34
± 78.6	> 20	> 20	> 40
± 90.2	> 30	> 30	> 60
± 125 and beyond	> 40	> 40	> 80

3.16.2 Transmit Out of Band Response

The parameter is measured between the input to the Driver / Power amplifier and the output of transmit channel, including the transmit antenna, and shall not exceed the values given in Table 3-25.

Table 3-25: Transmit Out of Band Response

Frequency from Channel/Band# Centre (MHz)	Response (dB)
Ka x Ku Transponder (Channel Centre)	
± 67.0	7
± 78.6	15
± 90.2	20
± 125 and beyond	28
Ku x Ka Transponder (Band Centre)	
± 550.0	30
± 1250.0 and beyond	60
Ka x Ka Transponder (Band Centre)	
± 550.0	30
± 1250.0 and beyond	60

3.17 Wide Band Receive Out of Band Response

A band pass filter shall be provided at the input to the receiver to prevent the reception of frequencies outside the receive frequency band and shall have response as given in Table 3-26.

Table 3-26: Wide band Receive out of Band Response

Frequency, MHz	Ka x Ku	Ku x Ka	Ka x Ka
	Response, dB	Response, dB	Response, dB
At 500 MHz from edges of receive band	> 30	> 30	> 30
At transmit band	> 130	> 130	> 130

3.18 Spurious Outputs

This section applies to spurious outputs including unwanted products resulting from frequency conversion stages, image frequencies, harmonics and leakage.

3.18.1 Spurious Outputs Within Transmit Bands

The total RMS power, resulting from the sum of all spurious signals (excluding the spurious signals resulting from uplink carrier, harmonics of the local oscillator frequency, and discrete spurious due to HPA power supply), in the transmit bands 10.70-10.95 & 11.20 – 11.45 GHz and 19.70-20.20 GHz, measured at the input to the transmit antenna shall be below the noise power in a 4 KHz band.

The RMS power of the spurious signal from the mixer intermodulation products resulting from the one uplink carrier and any harmonic of the local oscillator frequency shall be 60 dB below the single carrier level measured at the input to the transmit antenna.

3.18.2 Spurious Outputs outside Transmit Band

The total RMS power of all spurious signals outside the transmit bands and also output power due to harmonics of the normal output signals measured at input to the transmit antenna shall not exceed -60 dBW in any 4 KHz band.

3.19 Cessation of Emissions

It shall be possible to turn each individual transponder (1 beams in Forward link/4 beams in Return link) ON or OFF by ground command.

CHAPTER 4

4 Beacon Transmitter Performance Requirements

4.1 Ku-Band Beacon Transmitter Performance Requirements

A total of four Beacon Transmitters, two in Ku-band and two in Ka-band are configured in GSAT-11 Payload.

Figure 4-1 shows the Beacon Transmitter configuration. The Payload is equipped with two Ka-band Beacons (in orthogonal polarization) and two Ku-band Beacons (in orthogonal polarization).

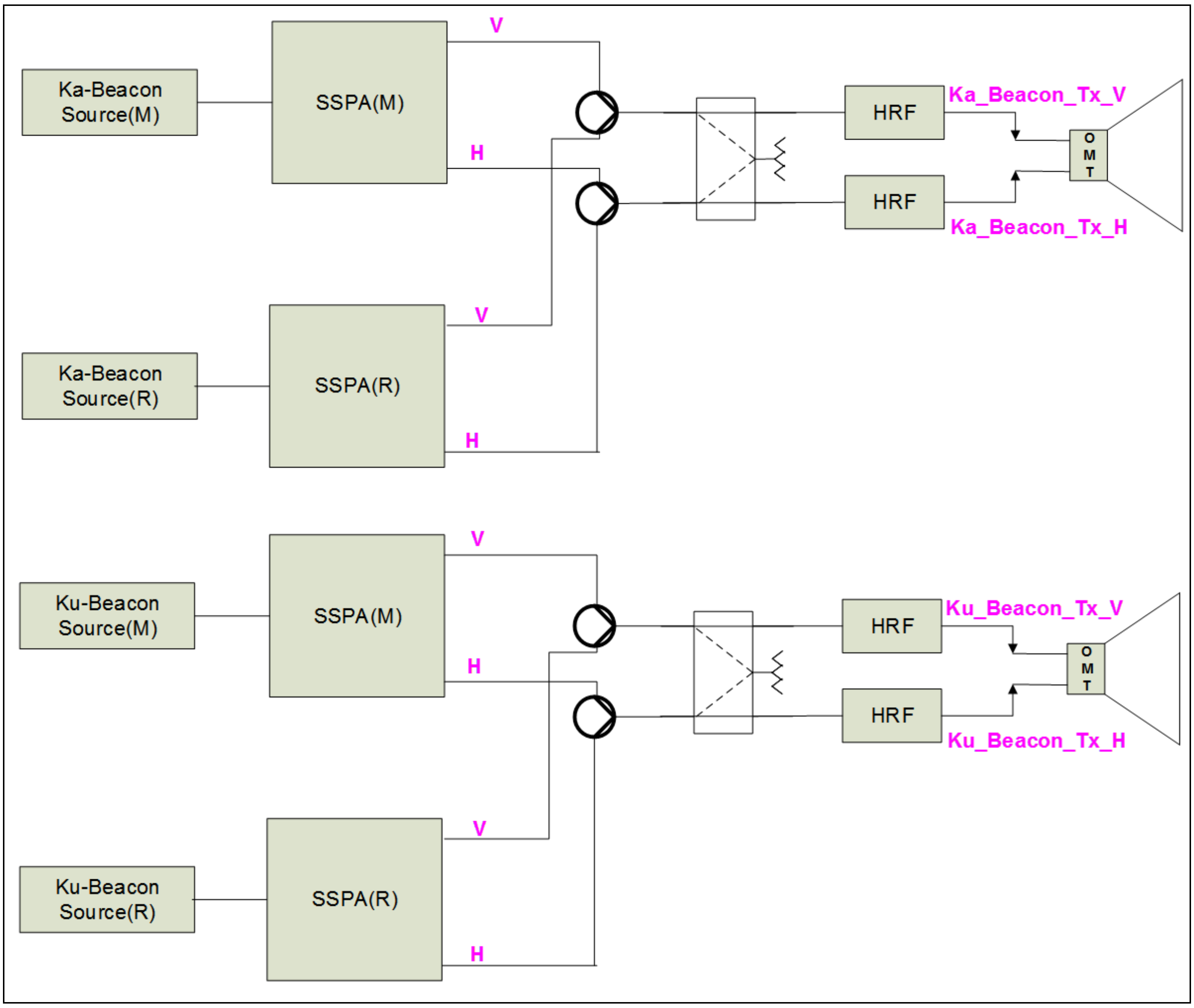


Figure 4-1 Beacon Transmitters

4.2 Frequency Plan and Polarization

Frequency plan and polarization of on-board beacon transmitters are given in the Table 4-1.

Table 4-1 : Frequency Plan for Beacon Transmitters

On Board Beacon Transmitter No.	Frequency (MHz)	Polarization
Ku-B1	10701.0	Linear – H
Ku-B2	10701.0	Linear – V
Ka-B1	19701.5	Linear – H
Ka-B2	19701.5	Linear – V

4.3 Coverage

The Ka and Ku-band Beacon transmit beam shall have India Mainland coverage, A&N and Lakshadweep Islands coverage.

4.4 Transmit EIRP

Ka-Beacon

The transmitter shall provide a Beacon EIRP of 21 dBW over lifetime at the edge of India Mainland coverage and A&N and Lakshadweep Islands.

Ku-Beacon

The transmitters shall provide a Beacon EIRP of 21 dBW over lifetime at the edge of India Mainland coverage and A&N and Lakshadweep Islands.

4.5 EIRP Stability

At any fixed point within the transmit beams of the channels, the change in saturated EIRP, at the centre frequency of the transmission channel, shall not exceed 2 dB peak to peak over any operating day and 3 dB peak-to-peak over the design life of the spacecraft, due to all causes except spacecraft pointing variations.

The in-orbit diurnal temperature variations as being experienced on the existing spacecraft shall be considered for the purpose of establishing the test-results compliance.

4.6 Transmit Frequency Stability

The Beacon Transmitters shall have the following frequency stability limits:

a) The net stability of the channel carrier frequencies, including initial tolerance shall not exceed ±10 ppm over the operating lifetime of the satellite.

b) The frequency stability over one-month period, which includes eclipse season, shall not exceed ±2.5 ppm.

c) The spurious modulation on unmodulated carriers shall not exceed the single side band noise density as specified in Table 4-2.

Table 4-2 : **Spurious Phase Noise**

Offset from Carrier	SSB Phase Noise Density, dBc / Hz
10 Hz	-33
100 Hz	-62
1.0 KHz	-81
2.0 KHz	-84
10.0 KHz	-84
100 KHz	-94
1.0 MHz	-94

4.7 Spurious Output

This section applies to spurious outputs including unwanted products resulting from frequency conversion stages, image frequencies, harmonics and leakage.

4.7.1 Spurious Outputs within 10.70- 10.95, 11.20 – 11.45 & 19.70 – 20.20 GHz Band

The total rms power resulting from the sum of all spurious signals in the assigned frequency band measured at the input to transmit antenna shall be below noise power in a 4 KHz band.

The rms power of the spurious signal from the multiplier inter-modulation products resulting from any harmonic of the local oscillator frequency shall be 60 dB below the single carrier level measured at the input to the transmit antenna.

4.7.2 Spurious Outputs outside 10.70- 10.95, 11.20 – 11.45 & 19.70 – 20.20 GHz Band

The total rms power of all spurious signals outside the desired band and also output power due to harmonics of the normal output signals measured at input to the transmit antenna shall not exceed -60 dBW in any 4 KHz band.

4.8 Cessation of Emissions

It shall be possible to turn each individual transmission channel ON or OFF by ground command.

CHAPTER - 5

5 Telemetry, Tracking & Command (TT&C) Subsystem

5.1 TT&C Functional Requirement

The TT&C system shall operate in C-band (6 GHz uplink / 4 GHz downlink) both during transfer orbit (T.O) and during synchronous on-orbit (O.O) operations.

All Tele-command operations shall be exercised through an omni-directional receive antenna. Telemetry and Ranging operations during transfer orbit (T.O) & positioning maneuvers shall be through the Omni-pattern telemetry antenna system. The on-orbit telemetry and ranging operations shall be exercised through an earth coverage global horn (GH) antenna and optionally through the above omni antenna. The TT&C subsystem shall be capable of transmitting simultaneously at two separate frequencies and receiving simultaneously at two separate frequencies. Capability shall exist for simultaneous transmission of telemetry and ranging signals.

The TT&C subsystem shall meet performance requirements specified herein during the transfer orbit, drift orbit or synchronous orbit, when transmitting signals to and receiving signals from, the prime master control facility (MCF) located within the primary coverage area and other TT&C stations as required during transfer orbit, orbit rising and drift orbit phases.

The harness and interconnect philosophy shall be in such a way that no single failure in any active component in the TT&C subsystem shall result in: loss of telemetry monitoring critical to operation, degradation of ranging accuracy, loss of any command function.

The C band uplink and downlink frequency for TT&C operation shall be in the range of 6410 to 6425 MHz and 4185 to 4200 MHz respectively.

The choice of TT&C frequency in the specified band shall be based on the co-located satellite operations at the chosen orbital slot.

5.2 Telemetry

The C-band telemetry equipment shall consist of two identical redundant encoder equipment each operating on a separate carrier frequency.
Telemetry system shall provide:

5.2.1 Capability to operate simultaneously both the telemetry encoders in normal mode and either one of the telemetry encoder in normal telemetry or in dwell telemetry modes.

5.2.2 To power ON / OFF by ground command the telemetry encoders and TTC transmitters individually.

5.2.3 Telemetry signals transmitted from the spacecraft shall uniquely identify the spacecraft.

5.2.4 Telemetry System Requirement Definition

The telemetry system shall collect and transfer data on all spacecraft subsystems in quantities, accuracies and at time intervals sufficient to determine spacecraft performance, attitude, need for commanding and to support analysis of failures. The command decoder register contents and command execute data shall also be tele-metered. The modulation format for telemetry data during all phases, including launch and transfer orbit and during normal operation phase, shall be PCM-PSK-PM.

The data rate, format and information tele-metered shall support rapid fault isolation and correction from the ground. The telemetry subsystem shall be consistent with data evaluation and resultant emergency procedure execution, either by a human operator or automatically by preprogrammed computer. The combined parameter measurement and processing accuracy shall be generally ±1% of full range of all analog signals as tele-metered by the PCM data system.

The telemetry unit shall be operational during all phases of spacecraft systems tests, pre-launch operations continuously from the time the spacecraft is transferred to internal power prior to launch and throughout the spacecraft operational life. Each telemetry unit shall receive electric power from main power bus. The operation of main subsystem shall not affect the operation and read-out accuracy of the redundant subsystem.

5.2.5 Telemetry Radio Frequency Channels

The telemetry carrier frequencies shall maintain a long-term frequency stability of 4 PPM or better for all in-orbit operation conditions. The spurious content of the telemetry carrier shall be no greater than -50dBW in any 1MHz bandwidth in any of the communications frequency bands used in the spacecraft and -55dBW in any 4 KHz band over the usable bandwidth in the payload 3.7GHz to 4.8 GHz band as measured at the transmitter output.

Each of the two C band telemetry systems shall provide a minimum EIRP of -6.5dBW during transfer orbit (TO) over ± 90 deg half cone angle around the +Yaw axis. The antenna coverage may be provided by a single or two separate antennas. Antenna gain over the 4PI steradian shall be a minimum of -10dBi for 85% coverage. In the operational phase, per channel EIRP shall be a minimum of +4dBW on the global horn antenna over the primary coverage area at the end of design life of the satellite.

The telemetry bit rate shall be 2.0 kb/s. A dwell mode operation capability shall be provided to continuously monitor a single mainframe data word or a combination of any selected 16 parameters

Ground commands shall be utilized to switch from the transfer orbit omni-antenna configuration to the global antenna. There shall be a provision to switch ON / OFF the telemetry transmitters.

5.3 Telecommand

The Tele-command equipment shall provide the capability to control the spacecraft from the ground control stations. Reception of false commands, originating either from any communications signals in any of the frequency band used in the spacecraft or from other sources, shall be precluded to the maximum extent possible. The decoder used shall be identified by the telemetry data. Ground command override of any automatic command function shall be provided. Command encoders specific to the GSAT-11 satellite shall be supplied to all TT&C Network support stations during the transfer orbit phase.

5.3.1 Telecommand System Requirement Definition

The command equipment shall be capable of operating properly when command transmissions from ground illuminate the spacecraft with flux density levels of -92dBW/m^2 within omni antenna coverage. The antenna coverage may be provided by a single or multiple antennae.

After the spacecraft is placed in to on-orbit mode, the on board command equipment shall be capable of operating from a ground control station located with in primary coverage area and illuminating the spacecraft with flux density between -72dBW/m^2 and -92dBW/m^2. The spacecraft command antenna gain over the 4PI steradian angle shall be a minimum of -10dBi for 85% coverage.

The command receiver shall be capable of receiving the frequency modulated (FM) command with a nominal carrier deviation of $\pm$400 KHz on the uplink command carrier with a modulation scheme of PSK-FM.

No permanent damage to any part of the command equipment shall occur for the flux level as high as -50dBW/m^2 and the command equipment shall not respond to signals lower than -105dBW/m^2. The center frequency of the command receivers shall remain within 50ppm of the assigned carrier frequency for the entire design life of the spacecraft.

5.4 Ranging

The TT&C subsystem shall provide redundant ranging capability through the combined use of the command and telemetry equipment. It shall be possible to conduct ranging operations simultaneously with a single telemetry unit operating without degradation or interference to any of these operations.

The command receiver shall be capable of receiving the frequency modulated (FM) ranging tones with a nominal carrier deviation of ±400 KHz on the uplink command carrier. The phase modulated (PM) ranging tones downlink shall be re-transmitted back to ground through the telemetry carrier with a nominal modulation index of 0.9 radians for range determination. The highest tone used for ranging shall be 27.777 KHz. Additional tones shall be provided for resolving range ambiguities and for reliable range measurements. The ranging subsystem shall be designed such that the slant range can be measured at-least to an accuracy of $\pm$50 meters.

5.5 TT&C Antenna Network

The TT&C antenna system shall consist of omni antenna for both uplink and downlink and a Global beam horn antenna for downlink.
The omni antenna shall consist of network of antenna on EV and AEV phase of the spacecraft to provide near 360 deg coverage. Global antenna shall have beam width of >17deg.

5.5.1 Tele-Command / Ranging Uplink Signals Polarization

Both in T.O phase and in O.O phase, the command / ranging receive (uplink) signals (both TC_1 & TC_2) shall be operated through an omni antenna system with uplink in RHCP for EV and LHCP for AEV.

5.5.2 Telemetry/ Ranging Down Link Signals Polarization

a. In T.O phase and during in-orbit emergencies, the Telemetry / Ranging transmit signals (downlink) (both TM_1 & TM_2) shall be operated through omni antenna network with LHCP for EV and RHCP AEV.

b. In O.O phase, the Telemetry/ Ranging transmit signal (downlink) shall be operated through a global horn with dual circular LHC / RHC polarization (TM1# LHCP and TM#2 RHCP)

CHAPTER-6

6 Attitude & Orbit Control Subsystem (AOCS)

The satellite attitude in the transfer orbit shall be 3-axes control mode using thrusters. The different modes of control shall be determined by the mission requirements. During the apogee engine burn, the spacecraft attitude shall be controlled by thrusters with attitude reference provided by the IRU.

The satellite attitude in O.O shall be maintained by body stabilization using momentum/reaction wheels. Body mounted magnetic torquers shall be used to provide fine attitude control and assist momentum de-saturation in roll / yaw-axis.

The spacecraft control system shall be designed to meet the Communication Payload antenna pointing requirement for the fully deployed spacecraft configuration.

6.1 Attitude Determination

6.1.1 Initial Acquisition & Reacquisition

The attitude sensors shall be compatible with the initial acquisition sequence/IRU calibration both in terms of accuracy and their field of views (FOV) for transfer orbit and the ability for subsequent reacquisition requirements.

6.1.2 Apogee Engine Burn Maneuver

The attitude control sensors shall be compatible with the 3 axes attitude control requirements for the apogee burn maneuver.

6.1.3 Attitude Measurement

Attitude is measured relative to the spacecraft axis fixed in the satellite. Attitude control errors shall be measured relative to the above referred definition, but subject to the attitude biasing provided for making antenna pattern measurements / deriving the desired antenna contours/desired attitude pointing in synchronous orbit.

Earth Sensors design shall provide a means of overcoming Sun/Moon interference. For the synchronous orbit control, the earth sensors shall have the capability to provide proper control error signals for the nominal orbital location of 74°E Longitude. However, electrical bias capability of ES may be used for other orbital slots. The attitude determination shall be done entirely using on-board equipment and ground supplied ephemeris data. Earth sensor shall be used during T.O and Earth sensor/star sensor shall be used during O.O. Attitude determination data shall be tele-metered to the ground for the purpose of spacecraft control evaluation. The spacecraft control system shall provide antenna beam pointing from an inclined orbit operation to increase spacecraft life.

6.2 Body Stabilization

6.2.1 Initial Acquisition

The initial acquisition sequence is defined as the maneuvers and events necessary for acquisition of body stabilized attitude from the initial random attitude condition. The initial acquisition sequence shall be initiated upon separation of the satellite from the launch vehicle and includes the following events:

a) Satellite de-spin, if required
b) Sun acquisition along South face of the satellite
c) Earth acquisition/IRU calibration

The spacecraft shall be taken through a series of maneuvers to achieve the necessary orientation for the firing of LAM. The spacecraft shall be 3-axes controlled using RCS thrusters during the burn of LAM. The attitude reference would be provided by the IRU alone. The control system shall be designed for a first apogee maneuver in the event of contingency. Any on-board sequence shall be capable of being overridden by the ground commands.

6.3 Re-Acquisition

Provision shall be made for reacquisition of the Earth pointing attitude after a loss of control due to either equipment failure or inadvertent human error.

6.3.1 Operational Attitude Control

The satellite shall be body stabilized using a skewed momentum wheel configuration having two skewed pitch wheels and a yaw wheel mounted in the pitch/yaw plane. The skew angle is determined by the control system design. The normal control of the spacecraft pointing shall be performed in a close-loop fashion entirely using on-board equipment.

In the in-orbit mode (for momentum bias case), two skewed pitch wheels shall operate in primary mode and in the event of failure of any one of the pitch wheels, the system would operate using reaction wheel and the remaining momentum wheel. The system shall be capable of meeting the pointing and stability requirements in this mode also. The minimum impulse bit requirements from the thrusters would be compatible with the mission requirements even in the back-up wheel configuration.

The control system configuration shall have a capability for ground over ride of the AOCS thrusters and redundancy selection. The configuration shall have the capability for attitude bias for different modes of controls and activate AOCS thrusters, accordingly.

6.4 Design Requirements

The Attitude determination and control subsystem shall be 100% redundant, except for the magnetic torquing coil(s); the torquers shall be working with reduced capacity in case of one coil failure. The momentum wheel scheme for the body stabilized case shall have three-for-two redundancy. The control system design shall provide for the solar array sun tracking capability either in open loop mode or close-loop mode using array mounted sun sensor. The Solar Array Drive Motor (SADM) shall have redundant motor winding. Sensors and their control electronics units shall be redundant. Cross strapping between redundant control electronics and other elements of the subsystem including reference sensors, thrusters, momentum wheels, solar array drive electronics/solar array drive assembly and magnetic torquer shall be provided. The thrusters-firing for apogee rise motor or station- keeping maneuvers shall be totally controlled from ground commands. Suitable safe mode shall be incorporated in the control system to account for loss of attitude reference and unusual long thruster pulse.

The control system shall have the capability of controlling either the pitch alone or pitch and roll together in the normal in-orbit for momentum biased system mode of operation through ground commands. The stability requirement would need to be met only in the later case.

The control system shall have capability to bias the satellite to have the capability for antenna pattern measurements in different planes using MCF and / or any other designated communication earth stations.

6.5 AOCS Telemetry Requirements

Diagnostic telemetry shall be provided to permit long term evaluation and detection of performance trends. The following parameters shall be monitored in the telemetry main frame as a minimum:

a. Different Attitude sensors data
b. Momentum/Reaction wheel speeds
c. Momentum/Reaction wheel currents
d. IRU processed data
e. Magnetic torquer currents
f. Solar array drive angle read outs
g. Thruster accumulated firing time history
h. Pitch, Roll and Yaw control error signals
i. Angular rate information
j. Status of all AOCS subsystem

6.6 Antenna Pointing Error

The design of the spacecraft shall provide for controlling the (maximum 3 sigma) antenna pointing errors to within the limits as described in para 2.9 of chapter #2.

CHAPTER-7

7 Propulsion Subsystem

The propulsion subsystem shall meet the entire operational requirements of apogee maneuvers and attitude & orbit control functions for the satellite for its operational life span of 15 years and a design life of 18 years. In addition, the propulsion system shall also provide a onetime repositioning capability as decided by mission requirement at a rate of 1 deg per day or as decided by mission requirement.

The total budgeted quantity of propellant shall be available to either of the redundant AOCS thruster groups and the apogee motor. The AOCS thrusters shall be capable of providing back-up to the liquid apogee motor (LAM) as a mission contingency. An isolation system capable of being enabled by ground command shall be provided.

7.1 Propellant Storage and Feed system

The storage sizing shall accommodate sufficient propellant with margin to take care of 3 Sigma launch vehicle errors and a minimum one year of life after meeting all budgeted requirements for 15 year’s life span. A combination of regulated pressure and blow down mode shall be used for propellant expulsion. Bubble-free propellant shall be maintained at the outlet of the tanks under all conditions, using a suitable propellant management device.

7.2 Leakage

The subsystem shall be of an all welded design. It shall be ensured that leakage level has no adverse effects on the required performance over the 15 years of design life.

7.3 Thermal

The thermal regime shall ensure that freezing of propellant does not occur, with a minimum margin of 5^0C. Temperature Sensors shall be provided to monitor the health of the propulsion system. Redundant heaters shall be provided on tanks, plumb lines, thruster valves etc.

7.4 Thruster performance

Mission requirement for each thruster shall be specified assuming that out of the two redundant thrusters one may not be operable throughout the entire mission; this shall be increased by a factor of 1.5 as performance margin and each thruster shall be capable of meeting this requirement both for firing duration and number of pulses. The impulse predictability (3 sigma total impulse) shall be ± 4% or better for continuous firing lasting (1000 seconds) or longer. The thruster shall be operable both in pulse mode and in the continuous mode. There shall be no thermal constraints on duty cycle. In the continuous mode, the apogee motor shall deliver at least a nominal vacuum specific impulse of 315 seconds for LAM and 285 seconds for AOCS thrusters. The performance of these thrusters shall not degrade if operated at:

a) +10% of the maximum initial propellant supply pressure
b) -15% of the minimum end of life propellant supply pressure

Thruster performance repeatability, predictability and also the leakage integrity shall be maintained throughout for at least 1.5 times of the maximum cycle life.
Each thruster shall have an average steady state thrust level within 50% of the qualification levels under equivalent conditions. The environmental conditions shall not be a constraint for the thruster restart. The thruster assembly shall be capable of restart and continued operations within specifications at all temperatures of the valve and thrust chamber temperatures which might result from thermal soak after any operation during the mission.

7.5 Total Propellant Budget

Sufficient propellant shall be provided to ensure attainment of synchronous orbit despite of launcher errors (3σ) and reacquisition in case of one attitude loss, and to provide the necessary orbit and attitude control for a 15 year mission plus 1 year margin thereafter, including one repositioning of the satellite. The total propellant and pressurant mass budget to meet the above requirements shall be based on thruster performance test data, mission analysis and tankage expulsion efficiency.

7.6 Safety

The propulsion subsystem and related handling and servicing equipment and procedures shall be designed to conform to applicable sections of the Launcher Safety Policy and ISRO Safety Requirements.

7.7 Instrumentation

Supply pressure from each group/branch of propellant and pressurant tanks shall be telemetered. Temperatures of thruster flow control valves, tanks and other specified locations of propulsion system shall be telemetered. Status indications shall be provided for Thruster operation and Latching valve open / close positions.

The monitoring periodicity of telemetered data shall be so chosen to mission requirements and to ensure proper operation of the subsystem.

CHAPTER -8

8 Electrical Power Subsystem

The electrical power subsystem shall generate its power by solar arrays. Electrochemical batteries shall store the required electrical energy during the sunlit period to provide power during peak demands and eclipse periods. The power output from the solar array and batteries shall be controlled for use by the various spacecraft loads. Appropriate controls for battery charge, battery discharge and other functions shall be provided for operational purposes. The electrical power subsystem and its interface shall have protection features to protect the system and the loads from anomalies that may occur, such as bus under voltage, battery under voltage/over voltage, battery over temperature, single component failure and overload.

Power system distribution is through power bus bar which is protected from fault propagation, in case of an inadvertent short in bus. Battery's excess current detection & isolation logic shall be incorporated. Provisions to telemeter all the important parameters and functions shall be made. Suitable wiring, shielding and grounding technique to minimize electromagnetic interference problems shall be adhered to.

In summary, judicious sizing of solar array and battery, coupled with single point failure protected power control and associated interface shall meet the requirements of the various subsystems and payloads throughout the span of 15 years mission life.

8.1 Solar Array

Solar array shall be optimized using appropriate solar cells to meet mission requirement with sufficient margin even with one string failure. Sizing of the solar array shall be done such that output power of the solar array shall meet the estimated average load requirements throughout the 15 year operational life with adequate margins. These average load requirements shall include all losses such as distribution losses, battery charging, thermal control and electric drive. The active area of each solar cell shall be adequately protected from radiation in the space environment. Solar Array Drive Assembly (SADA) for power & signal transfer shall be properly de-rated to meet 15 year mission life. The projected end of 15 year life power margin shall be a minimum of 7.5% above the actual measured average load.

8.2 Batteries

Energy storage shall be provided by Li-ion battery. The batteries shall provide power to the spacecraft, whenever required throughout the 15 years operational and 15 years design life of the spacecraft. The batteries shall support complete load for the predicted spacecraft solar eclipse through each battery and its sizing shall be such that the depth of discharge of each battery shall not exceed 65% of the nameplate capacity and the discharge can be up to 70% in case of one cell failure. The batteries shall be provided with a charge control circuit for charging the batteries with current limited rates. Batteries shall attain the fully charged condition in orbit at normal charging rate in between all operating modes involving battery discharge. The batteries shall be protected for open & short circuit failure of any cell.

Due consideration shall be given to over voltage protection due to over charge condition. Provision shall be made for automatic implementation of battery operating modes such as battery charge/discharge and under voltage disconnect functions with command backup.

The tolerances for battery temperature, recharge capacity, overcharge protection; etc shall be selected to ensure the minimum 15 year mission life with satisfactory performance, following a maximum of 5 year ground storage.

8.3 Power Control & Regulation

The power output from the solar array and batteries shall be controlled for use by the spacecraft loads. Main power bus shall be provided for supplying power to the spacecraft subsystems. The power bus is fully regulated, single 70V bus. Provision shall be made for un-interrupted bus for selected loads.

No single component failure in the spacecraft shall open or short circuit a main electrical power bus.

The design of the power subsystem shall be such that under all conditions during design lifetime, including operations in eclipse with one battery cell failure, the main bus voltage shall be equal to or exceed the minimum required to meet the eclipse communication payload specifications. The command equipment shall never be disconnected from the bus.

CHAPTER -9

9 Thermal Control

The spacecraft shall be constructed to maintain all equipment and the structure within design temperature ranges under all expected conditions of pre-launch, launch, transfer orbit, parking orbit and synchronous orbit environments and operating modes and test conditions specified in the detailed test plan. Design shall include thermo-optical performance degradation of thermal control surfaces based on previous flight data, as well as conditions resulting from individual failures, apogee motor / RCS thruster plume heating and post fire soak back.

9.1 Thermal control of critical subsystems

Thermal control shall take care of the temperature control requirements of critical bus subsystems like propulsion, batteries, Earth Sensors and sensitive payloads subsystems.

9.2 Temperature control management

Temperature control may be achieved by the use of embedded heat pipes along with the use of passive means augmented by heaters. Temperature of various subsystems shall be telemetered for maintaining thermal regime during various phases of mission including on-orbit operation throughout the operational / design life. Provision shall be made for automatic temperature control (ATC) with programmability and enable / disable features.

9.3 Thermal Design margins

Thermal Design margins of 10^0C for qualification tests and 5^0C for acceptance tests, on maximum and minimum expected operating temperatures shall be established at the time of the design and shall be verified by tests.

CHAPTER-10

10 Structure

The spacecraft structure shall be designed to meet the overall spacecraft and launcher constraints. Compatibility with chosen launch vehicle shall be ensured. High structural design efficiency shall be achieved by proper choice of materials and methods of construction suitable for the selected configuration geometry. Technological elements proven on earlier spacecraft or otherwise fully demonstrated on ground only shall be employed for the structure to ensure reliability.

Specified positive margins of safety shall be provided on all structural components for the ground handling, different launch phases and on orbit loads. Dimensional stability and alignment requirements of structure shall be as per the spacecraft functional requirements providing a platform without degradation well beyond the required service life. The structural dynamics behavior shall be such as to meet the requirements of chosen launcher, control and stabilization and also to ensure safety of spacecraft payload elements. The parts and materials for realizing the structure shall be in line with the antenna pointing accuracy requirements over the environmental conditions.
The static envelope of the spacecraft shall be compatible with the chosen launcher.

CHAPTER-11

11 Spacecraft Support Equipment & Transportability

11.1 Spacecraft Support Equipment

Spacecraft support equipment shall include all such equipment essential to the protection and support of the spacecraft during assembly, integration and testing operations as well as its interface with the identified satellite launch vehicle. The support equipment shall consist of, but not limited to the following items:

a. Mechanical interface equipment like panel fixtures, spacecraft fixtures, clamp band and mounting adapters and handling equipment etc.
b. Electrical interface equipment for spacecraft powering, commanding, health monitoring and testing etc.
c. Any other support system as deemed necessary for interfacing with the launch vehicle.

11.2 Transportability

The spacecraft and associated equipment shall be capable of transportation by both un-pressurized aircraft and or by road transport carrier. The spacecraft system, its subsystems and subassemblies and also separately delivered spacecraft equipment shall be suitably designed or protected to withstand such transportation environment. The environment during transportation shall be controlled so as not to exceed the flight environment and in no case it shall exceed the design limits of the hardware. A specially designed transportation container shall be used for the transportation of spacecraft. Similar arrangements shall be made for the transportation of separately delivered spacecraft hardware if any.

CHAPTER-12

12 Electromagnetic Interference/Compatibility & Space Charge

12.1 Electromagnetic Interference/Compatibility

The spacecraft shall be designed to assure adequate electromagnetic compatibility among various subsystems under all operating modes and shall be designed to preclude degradation of communication transponders or spacecraft support electronics, as a result of electromagnetic interference including those arising out of launch operations. Electromagnetic compatibility shall be ensured for the environment resulting from the launch vehicle system. Design guidelines for ensuring electromagnetic compatibility shall be derived using MIL-STD-1541A and MIL-STD-461C specifications. Additionally, the electromagnetic compatibility of the spacecraft and its subsystems shall be verified against the provisions of MIL-STD-462 test procedures.

12.2 Space Charging and Grounding

The Spacecraft design shall take into consideration space charging phenomena. The Spacecraft design shall provide for the electrical connection of all electrically conductive structural elements of the spacecraft to form a common spacecraft ground. Provisions shall be made to electrically ground all conductive external surfaces of the spacecraft. All interface circuits shall be designed to minimize electrical interference on signal lines, signal returns, power bus & returns and spacecraft structure.

CHAPTER–13

13 Safety & Handling Hazardous Systems/ Equipments

13.1 Safety

The design of spacecraft shall adhere to all applicable launch pad safety, range safety, handling and transport safety regulations. Special safety regulations shall be enforced, if necessary, during the assembly, integration testing and pre-launch operations on the spacecraft.

13.2 Handling of Hazardous Systems/Equipment

Handling of hazardous systems/equipment and operations shall be as per the predetermined sequence and procedures under controlled environments as per the requirements. The spacecraft elements that fall under this category are mainly the propulsion elements like propulsion tanks, propellants etc, electro explosive devices like pyro cutters, and chemical batteries.

APPENDIX-A

Operational Environments

1. Launch Loads

These shall be governed by the chosen launch vehicle.

2. Orbital Environment

The vibration and static loads during transfer and synchronous orbits are less than the launch loads. But the structural frequencies of the appendages shall be so specified as to avoid vibration coupling between spacecraft attitude control system and elastic motions of the structure in the synchronous orbit deployed configuration.

3. Radiation Environment

The configuration of the spacecraft and the selection of parts and material used shall ensure that the performance and life requirements are met during the operation in the actual space environment. For this performance, the radiation environment shall be specified and the data shall be used to determine the acceptability of the design for radiation degradation. This shall also include the proton flux levels for two anomalously large solar flare events combined with the trapped geo-synchronous proton flux levels. The solar array degradation shall be based upon this data.

The solar energy distribution shall be according to the normalized Johnson curve for a solar constant of 1353 +67/-65 W/m^2. Seasonal variations in solar energy distribution shall also be taken into account.

APPENDIX-B

Environmental Test Requirements

1. spacecraft environmental test

These requirements shall include the fabrication, acoustical and temperature requirements and shall satisfy the test requirements for the chosen launcher. The actual qualification and acceptance test environments shall be specified in Environmental Test Level Specifications (ETLS) document. The specified temperature shall be based upon expected temperature extremes predicted by the thermal model for transfer orbit and on-orbit condition with $\pm 10^0$C qualification margin and $\pm 5^0$C acceptance margins. Adequacy of the test environment shall be ensured.

2. Component Environmental Test Requirement

These shall be as per the document (ETLS) specified for this purpose and any subscription thereof.

Acronyms

ALC	Automatic Level Controller
AM	Amplitude Modulation
AOCS	Attitude and Orbit Control Systems
ATC	Auto Temperature Controller
BW	Band Width
CF	Centre Frequency
EIRP	Effective Isotorpic Radiated Power
EOC	Edge Of Coverage
EOL	End Of Life
ES	Earth Sensor
ETLS	Environmental Test Level Specification
FGM	Fixed Gain Mode
FM	Frequency Modulation
FOV	Field Of View
FSS	Fixed Satellite Services
GAGAN	GPS Aided Geo Augmented Navigation
GEO	Geo-stationary Orbit
G/T	Gain to Noise Temperature
GHz	Giga Hertz
GSAT	Geo-Synchronous Satellite
GTO	Geo-Stationary Transfer Orbit
I-3K	3000 Ton Class Satellite
IMP	Inter Modulation Product
INLUS	Indian Land Uplink Station
INRES	Indian Reference Station
INSAT	Indian National Satellite
IRU	Inertial Reference Unit
ISRO	Indian Space Research Organisation
KHz	Kilo Hertz
LAM	Liquid Apogee Motor
LH	Linear Horizontal
LHC (P)	Left Hand Circular (Polarisation)
LV	Linear Vertical

MCF	Master Control Facility
MHz	Mega Hertz
MIL-STD	Military Standard
OO	On Orbit
PCM	Pulse Code Modulation
PM	Phase Modulation
PPM	Parts Per Million
PSK	Phase Shift Key
RCS	Reaction Control System
RF	Radio Frequency
RHC (P)	Right Hand Circular (Polarisation)
Rx	Receive
SADA	Solar Array Drive Assembly
SAW	Surface Acoustic Wave
SBAS	Satellite Based Augmentation System
SFD	Saturation Flux Density
SIS	Signal-in-space
SSB	Single Sideband
TC	Telecommand
TCXO	Temperature Controlled Crystal Oscillator
TM	Telemetry
TO	Transfer Orbit
TT&C	Telemetry, Tracking and Commanding
TWTA	Trvelling Wave Tube Amplifier
Tx	Transmit