TECHNICAL APPENDIX

Application of Kepler Communications
Inc. for U.S. Market Access Authority of
a Non-Geostationary Satellite Orbit
System in Ka- and Ku-band Frequencies

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TECHNICAL APPENDIX

In accordance with Part 25 of the Federal Communications Commission's ("Commission") rules, this document provides a technical description of the Kepler system, including, *inter alia*, detail related to its orbital configuration, beam plans, space and earth station terminals, interference assessments, and debris mitigation strategies. In addition, an Orbital Debris Assessment Report (ODAR) completed pursuant to NASA Technical Standard 8719.14B is provided as an Annex to this document.[1]

I. Overall system description

A. Space Segment

The Kepler system ("System") is designed to provide high-speed, low-latency satellite broadband services to fixed and mobile user terminals located at or above 55°N latitude. The service will be delivered from a fleet of 360 small satellites dispersed equally among 12 near-polar orbital planes. User service links are provided in Ka-band, and feeder links in Ku-band. The primary TTC (tracking, telemetry, command, and control) system is in Ku-band, with a redundant TTC system in S-band, through which Kepler will maintain direct control over all spacecraft using dedicated TTC ground stations. The S-band systems will primarily be used during Launch and Early Operations Phase ("LEOP"), emergency procedures, or for commanding propulsive

[1] Kepler recognizes that the Commission has just issued updated rules for orbital debris mitigation and is also seeking comment on further changes to those rules. Kepler will provide additional information to the Commission as necessary as changes to these rules take effect.

maneuvers (orbit raising, de-orbiting, or conjunction avoidance). Nominally, routine TTC operations are carried out through commands sent through its feeder links in Ku-band.

The Kepler System is designed to meet the bandwidth demands of customers living, working, or travelling throughout northern high latitude regions. By focusing solely on coverage in northern high latitudes, the System minimizes the total number of satellites and gateways needed to provide a powerful broadband service from an NGSO constellation. High latitude constraints also act to minimize the potential for interference to other NGSO, GSO and Fixed networks.

1. Orbit Parameters

The Kepler System space segment is comprised of 360 operational NGSO satellites in 12 near-polar orbital planes (89.5º inclination), each in a circular orbit with an altitude of 600 ± 50km. The numeric orbital parameters are summarized in Table 1.

Table 1: Constellation design showing altitudes and inclination.

Altitude	Inclination	Planes	RAAN Spacing	Satellites per Plane	In-Plane Separation	Number of Satellites
600 ±50 km	89.5º	12	15º	30	12º	360

Satellites will have a mean mission duration of 5 to 7 years. At the end of mission lifetime, satellites will execute a deorbit maneuver to bring their orbital altitude to around 300km to ensure rapid re-entry. The full constellation lifetime will be 15 years, accounting for between one to two full replacements of the satellite assets over that period.

2. Deployment and Coverage Area

Once fully deployed, the Kepler System will provide continuous coverage above 55°N latitude, including the Arctic Ocean and the U.S. State of Alaska. Coverage maps of this network are shown in Figure 1 below.

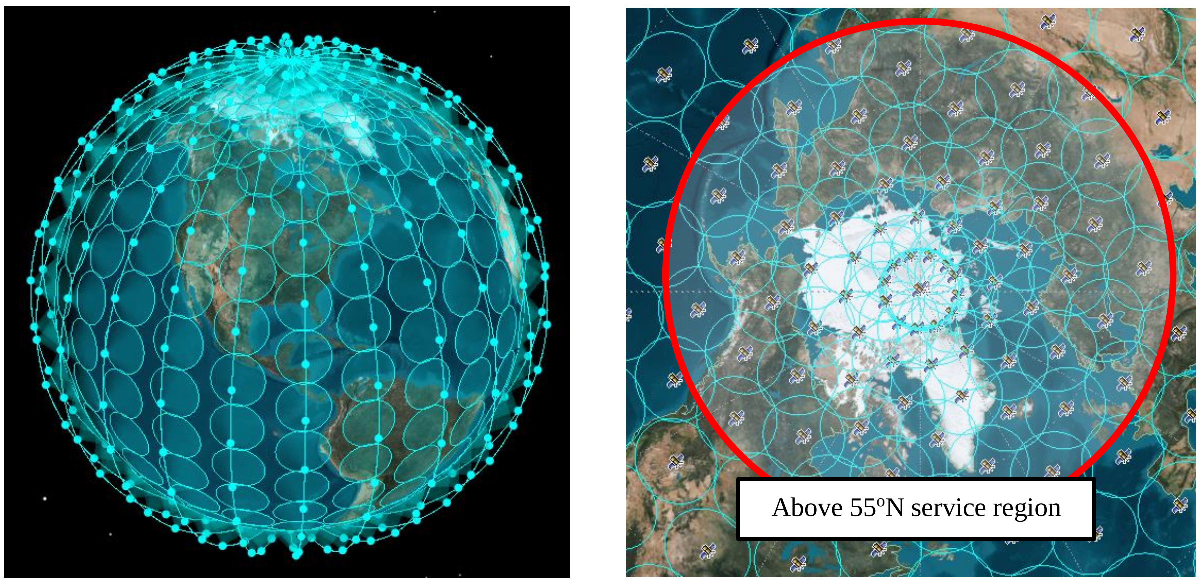

Figure 1: Network coverage showing (left) isometric and (right) top-down views. The red circle indicates the region of supported commercial coverage available.

Gateway sites will be positioned such that satellites passing over the service region will be able to maintain constant communication to the ground.

B. Ground Segment

A range of compatible ground terminals are considered for both service and feeder links. All terminals employ transmission masks to limit interference to other systems within line-of-sight, accounting for avoidance of the horizon and the GSO arc as appropriate. Service and gateway terminals alike may use phased-array and mechanically steered parabolic architectures to communicate with the Kepler System. Representative specification and performance details of these terminals are provided in Table 2 below.

Table 2: Example link performance for representative service antennas.

Antenna Characteristic Diameter	Ground Antenna Link Specifications		Link Performance [Mbps]	
	EIRP (dBW)	G/T (dB/K)	Downlink	Uplink
Service Links				
30 cm	49	9	40	10
45 cm	51	13	100	20
1.2 m	54	23	800	40

Moreover, all terminals equipped to use the Kepler network will make use of adaptive coding and modulation schemes to maximize achievable data rates and optimize use of available spectrum.

1. Customer Terminals

The System can interface with many existing, off-the-shelf mechanically or electrically steerable terminals designed for aeronautic, mobile, and land-mobile applications. Where necessary for LEO compatibility, Kepler will support upgrades to certain terminals in collaboration with the OEM to ensure their LEO tracking functionality and mechanical actuation reliability meets Kepler's service standards.

2. Gateway Terminals

Gateway sites will be distributed throughout the service area to serve as feeder link ingestion points to/from the space segment. Traffic from gateway sites will be securely routed to the global Internet, as depicted in Figure 2. At times, transmissions may be restricted to specific gateway facilities based on informational security, or for local regulation. To ensure that all satellites operating within the commercial service area have available connections to the ground, many co-located gateway terminals will need to be installed at gateway sites. Exact distribution will be a function of their geographical location, satellite connection demand, network optimization, and interference considerations.

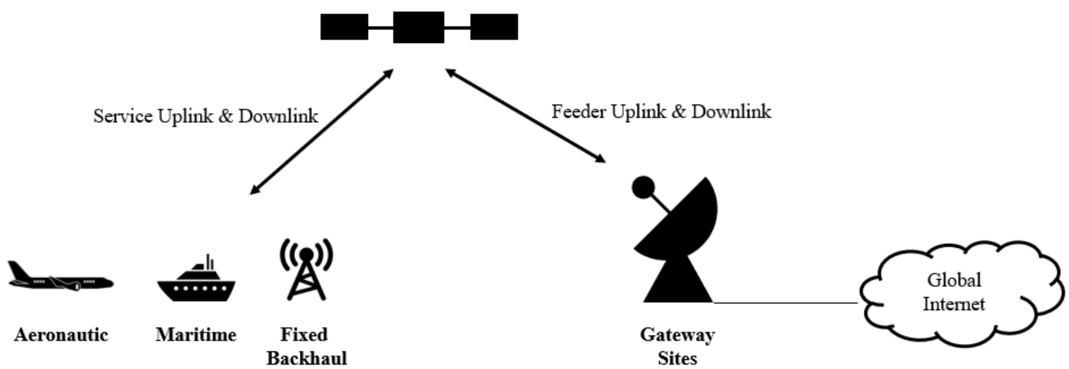

Figure 2: System network architecture

3. Kepler System Control

Kepler maintains a network operations center at its headquarters in Toronto, Canada, from which it will control the satellite system through its ground station network. Kepler presently operates several dedicated gateways and TTC terminals in the bands used by the proposed system. New ground station facilities will be brought online to accompany the progressive deployment of the constellation throughout its lifetime.

II. Frequency Plan

C. Beams

The constellation frequency scheme is separated distinctly by function into Ku- and Ka-band according to the configurations illustrated in Table 3 below.[2]

Table 3: System beam summary.

Type	Direction	# of Beams	Gain (dBi)	Max G/T (dB/K)	Service	Polarization	Operating Band (MHz)		Channel Bandwidth (MHz)		Max EIRP (dBW)	Max EIRP Density (dBW/Hz)	Max Steering Angle
							Min	Max	Min	Max			
Service	Down[3]	1	27	-	FSS	RHCP/LHCP	17800	18600	10	500	35	-41	45°
					FSS	RHCP/LHCP	18800	19400	10	500	35	-41	
					FSS/MSS	RHCP/LHCP	19700	20200	10	500	35	-41	
	Up	4	15	-14.8 (nadir)	FSS	RHCP/LHCP	27500	29100	10	500	-	-	45°
					FSS/MSS	RHCP/LHCP	29500	30000	10	500			
Feeder	Down	2	18	-	FSS	RHCP/LHCP	10700	12700	10	500	20	-50	63.4°
	Up	2	21	-7.8 (nadir)	FSS	RHCP/LHCP	12750	13250	10	500	-	-	63.4°
					FSS	RHCP/LHCP	13800	14000	10	250			
					FSS/MSS	RHCP/LHCP	14000	14500	10	500			

1. Service Links

Satellites are equipped with a Ka-band phased array antenna system in the uplink and downlink directions. Beam center frequency, channel bandwidth, and polarization handedness are selectable, providing substantial flexibility to both technical operations and interference mitigation. Service beams can be steered throughout a satellite's FOV to provide coverage to users in different locations. The maximum off-boresight steering angle for the service link beams is set

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[2] As discussed above, Kepler is not seeking authority to operate on the S band frequencies within the U.S. These frequencies are noted merely for informative purposes.

[3] The downlink service beam characteristics above represent the worst-case max EIRP density. Kepler's system has the capability to combine multiple channels into a single beam and perform adaptive power control while adhering to the max EIRP density presented.

to 45° to minimize side-lobe emissions and maintain scanning losses to an acceptable degree. Using a maximum steering angle of 45° off-boresight, the per-satellite coverage area is 1,300km diameter (approximately 1,330,000km^2) when at the maximum operational altitude of 650km. The service link TX and RX HPBW (half power beamwidths) at nadir are respectively around 3° and 13°, corresponding to a 68km and 300km diameter beam on the ground at the maximum 650km altitude. The resulting satellite FOV, overlaid with approximately TX and RX beam sizes, are shown in Figure 3.

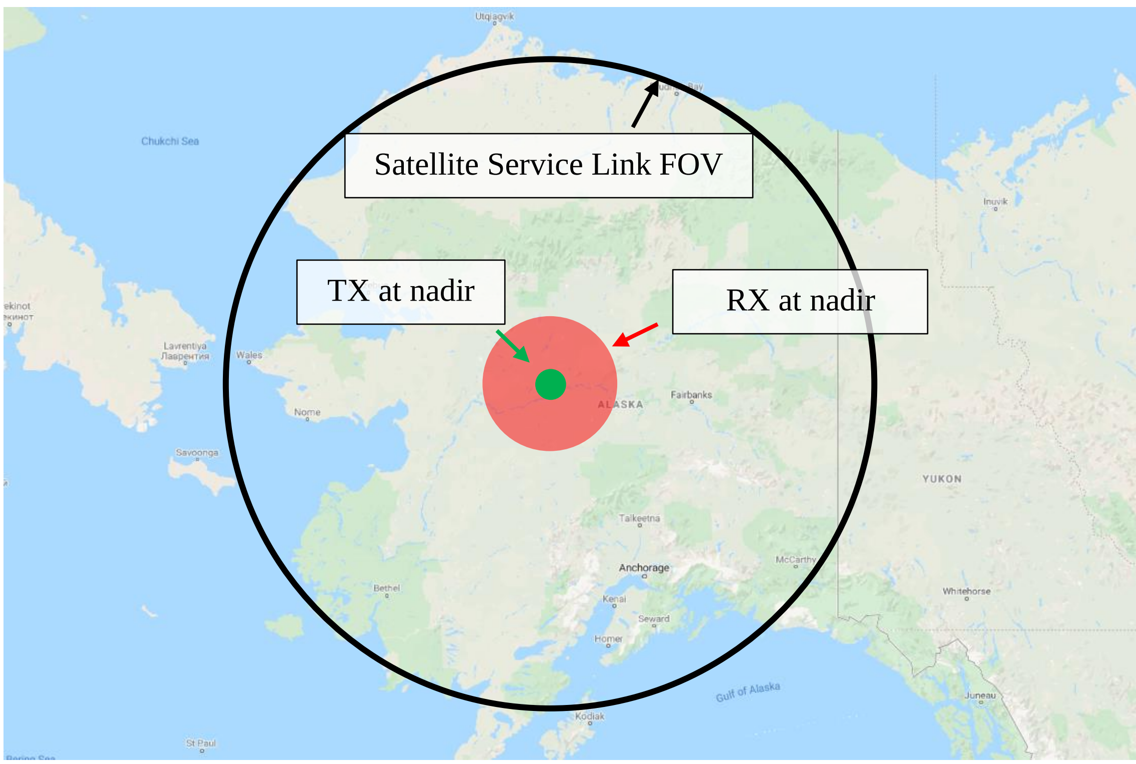

Figure 3: Approximate coverage area for a single satellite (black circle) at maximum operational altitude of 650km. Example nadir TX and RX beam sizes (red and green circles) correspond to 3° and 13° HPBW (68 km and 300 km diameter) respectively at 650km altitude. These beamwidths increase to roughly 6° and 15° when steered at maximum slant.

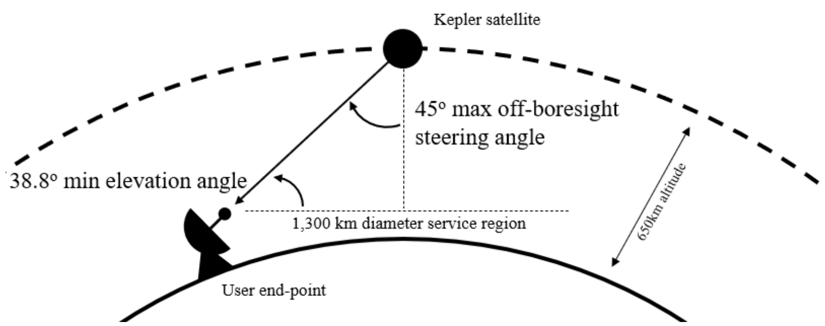

Figure 4: Service link beam steering and minimum elevation angle, which corresponds to the maximum altitude of 650km.

Service beam contours are circular when pointed directly along boresight and will widen and become elliptical as steering angle is increased. Figure 5 below shows example transmit and receive beam contours when pointed nadir and at the maximum steering angle. It is noted that the RX fully-steered beam contour of Figure 5 (bottom right) is cut-off because the beam extends into space at the cut-off location.

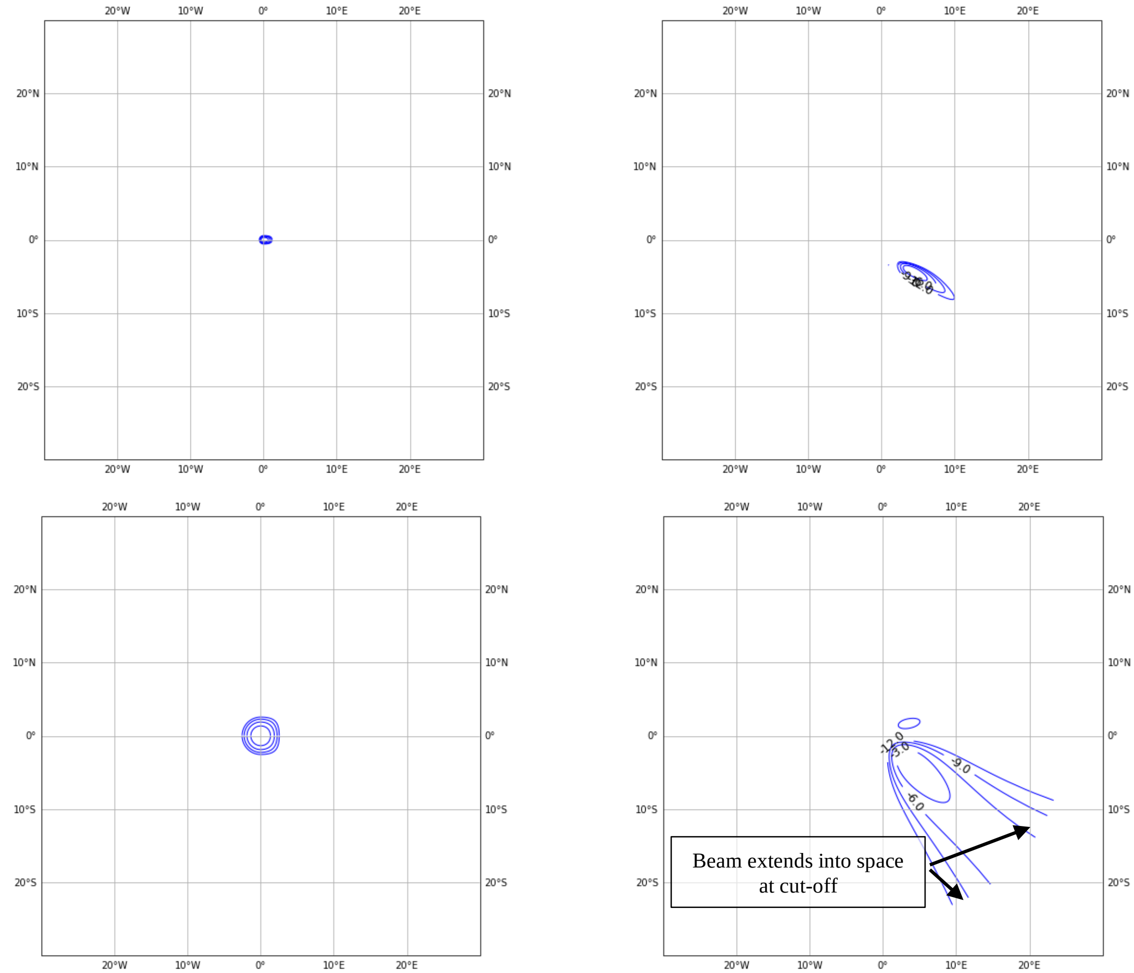

Figure 5: Service link beam gain contours at operational altitude of 650km. (Top row) Transmit beam contours shown when pointed nadir (left) and at full 45º steering (right). (Bottom row) Receive beam contours shown when pointed nadir (left) and at full 45º steering (right).

2. Feeder Links

Satellites can support up to 2 spatially independent Ku-band feeder beams in the uplink and downlink directions. Feeder link center frequency, channel bandwidth, and polarization handedness are selectable to further increase operational flexibility and/or address variable coordination requirements. Feeder beams operate in a similar way to their service link counterparts, with two main distinctions. The first is that the maximum steering angle (and therefore beam field-of-view) is expanded to 63.4° for wider ground coverage. This is achieved through a phased array antenna which is mechanically off-axis assisted to minimize steering angle of the phased array antenna. The second is that when within the service region, at least one feeder beam will always be connected to an Internet-connected ground station. The remaining available feeder beam will seek to establish a connection with the next optimal ground station in advance of the satellite losing sight of the first ground station (make-before-break connection). A seamless handover will be performed when both gateway connections are established, as shown in Figure 6 below.

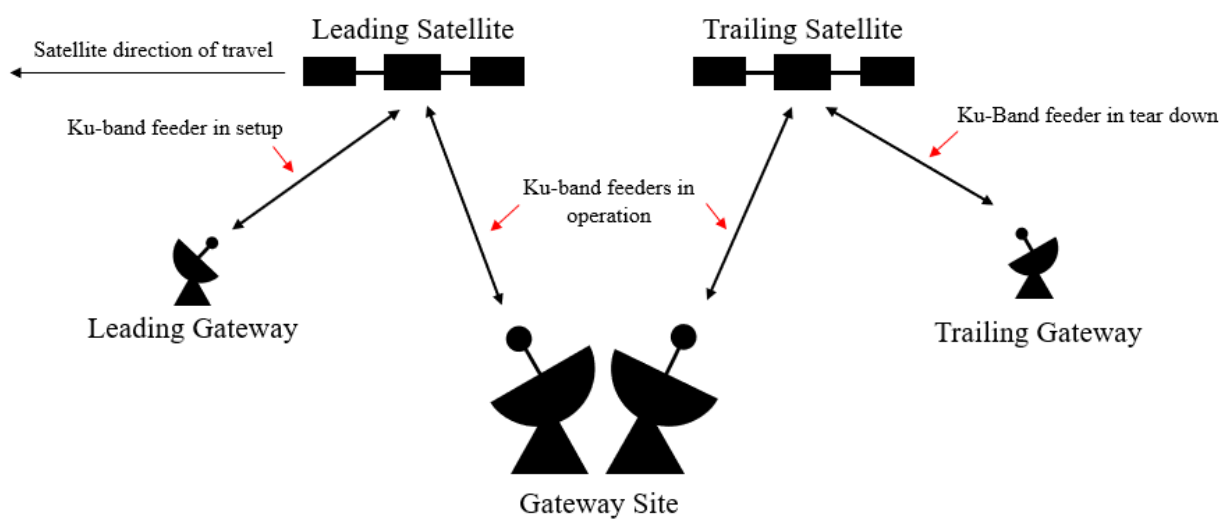

Figure 6: Depiction of dual-beam handoff between gateway feeder links.

Using a maximum steering angle of 63.4° off-boresight, the per-satellite coverage area for feeder links is 2,590km diameter (approximately 5,270,000km^2). The feeder link TX and RX HPBW at nadir are both approximately 8° corresponding to a 180km diameter beam on the ground at the maximum 650km altitude. The resulting satellite FOV, overlaid with approximately TX and RX beam sizes, are shown in Figure 7.

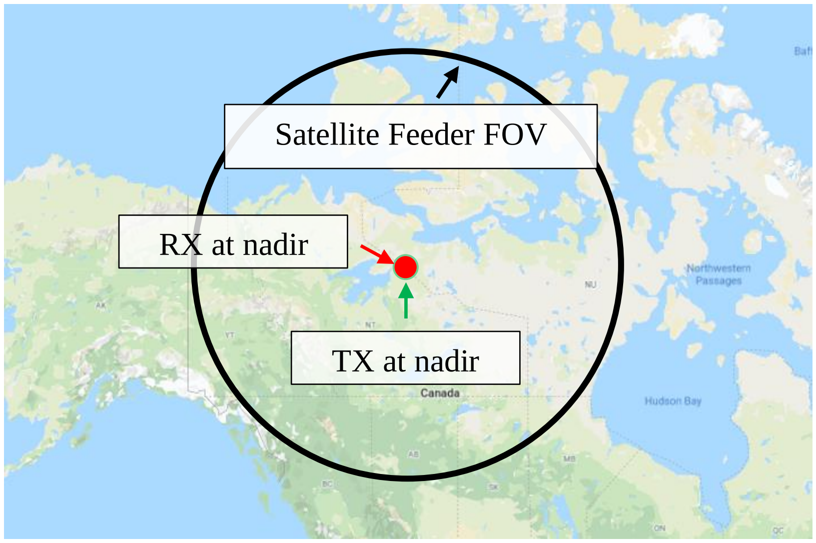

Figure 7: Feeder link coverage areas (minimum and maximum service altitude) for a single satellite (black circles). Example nadir TX and RX beam sizes (red and green circles) are shown at operational altitude, corresponding to 8° and 7° (180km and 160km diameter) respectively. These beamwidths both increase to about 10° when steered at maximum slant.

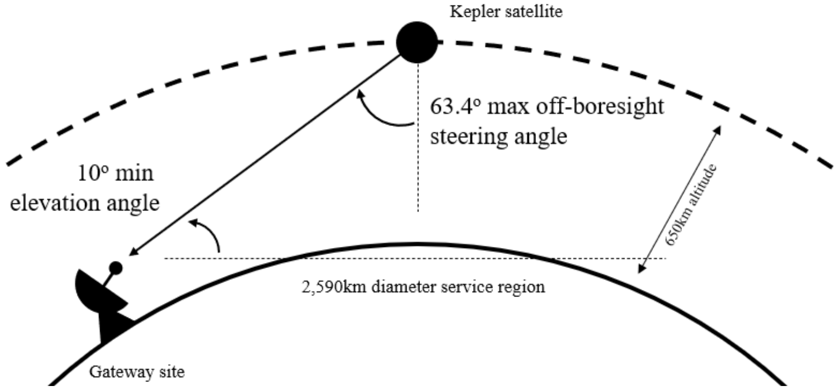

Figure 8: Feeder link beam steering and corresponding minimum elevation angle of 10°.

Similar to the service link beams, the satellite feeder link beam projection widens as the beam is steered away from boresight. Figure 9 shows example transmit and receive beam contours when pointed nadir and at the maximum steering angle off-boresight. It is noted also that the fully steered beams in Figure 9 (right column) are cut off because the beams extend into space at the cut-off location.

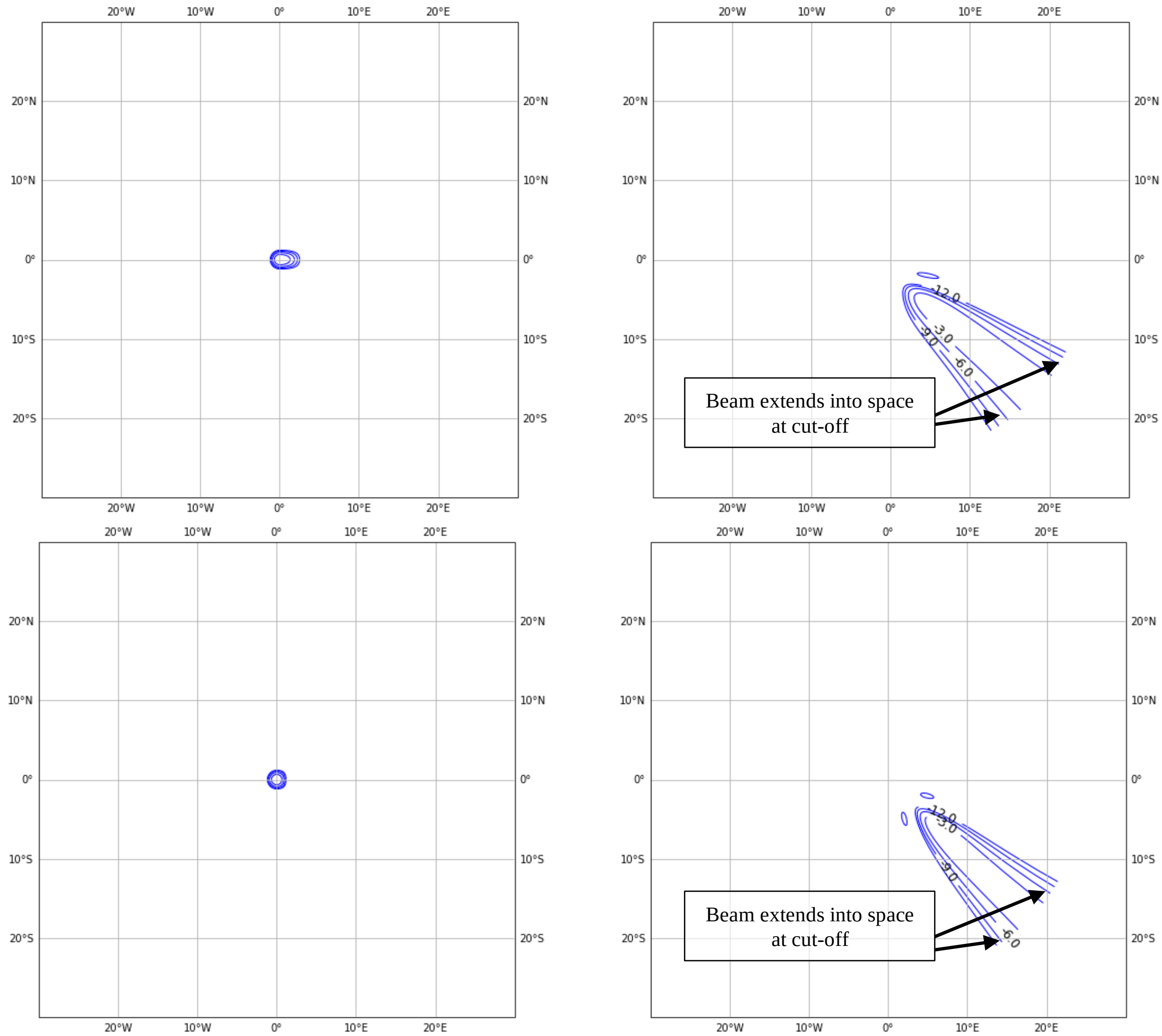

Figure 9: Feeder link beam gain contours at max operational altitude of 650 km. (Top row) Transmit beam contours shown when pointed nadir (left) and at full 63.4° steering (right). (Bottom row) Receive beam contours shown when pointed nadir (left) and at full 63.4° steering (right).

3. TTC Links

Kepler's satellites are equipped with dedicated TTC antennas that operate within the Space Operation Service in the 2025 – 2110 and 2200 – 2290 MHz bands for uplink and downlink respectively. Kepler is not requesting market access authority to operate in its TTC bands within the United States. Kepler will control its network using S-band TTC control stations situated outside of the United States. Therefore, no S-band information was provided in the associated Schedule S. For the purposes of coordination, satellites can select any center frequency within these frequency ranges. All emissions are designed to meet ITU-RR Article 21 PFD limits.

D. Limits and Standards

1. Certifications

Systems requesting market access authorization are required to abide by a number of technical rules and standards outlined in 47 C.F.R Part 25. Kepler hereby certifies compliance with those applicable rules, as follows:

- That carrier frequencies of all proposed space station transmitters will be maintained within 0.002% of their reference frequency, in accordance with 47 C.F.R. § 25.202 (e)
- The mean power of emissions will be attenuated below mean transmitter output power as set forth in paragraphs f(1) through f(4) of § 25.202.
- The System employs both orthogonal polarizations and spatially independent beams to maximize its frequency re-use, and in doing so satisfies the requirements of 47 C.F.R § 25.210.
- The control of all antenna beams (sub-system and power amplifier) can be enabled or disabled individually via ground control commands sent over secure TTC links during LEOP, nominal operations, and any orbit raising/de-orbiting maneuvers. In the event of a prolonged period with no communication with the ground, satellites will automatically execute a cessation of emissions in order to prevent the possibility of a satellite failing in a

transmitting state. Such attributes are in compliance with the emission cessation requirements of § 25.207.

2. PFD Compliance

a. Feeder Beams

The maximum power flux density (PFD) produced by feeder downlink beams is illustrated in Figure 10 in the worst-case scenario of a 10 MHz channel. As shown, resultant feeder link transmission power falls below the limits set by ITU-RR No. 21 and 47 C.F.R. § 25.208. In practice, feeder links will operate down to a minimum angle of arrival of 10^{o}. Scanning loss incurred on antenna gain as a result of steering was conservatively modelled using a $\cos(\theta)$ factor (where θ is the off-boresight angle). The maximum EIRP density for the feeder downlink assuming a 10 MHz channel is -50 dBW/Hz at a 90^{o} angle of arrival corresponding to a 0^{o} off-boresight steering angle. Nominal operations with a 100 MHz channel results in an EIRP density of -60 dBW/Hz.

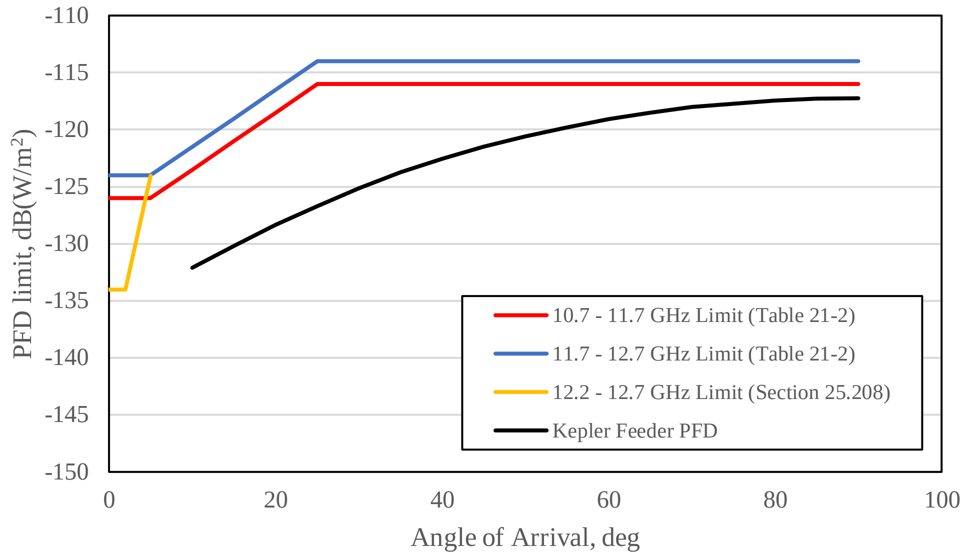

Figure 10: Feeder downlink PFD in 1 MHz reference bandwidth using the worst-cast scenario link bandwidth of 10MHz.

b. Service Beams

The maximum PFD produced by service downlink beams is illustrated in Figure 11 in the worst-case scenario of a 10 MHz channel. As shown, resultant service link transmission power falls well below the limits set by ITU-RR No. 21 and 47 C.F.R. § 25.208. In practice, service links will operate down to a minimum angle of arrival of 38.8°. Scanning loss incurred as a result of steering was conservatively modelled using a cos(θ) factor (where θ is the off-boresight angle). The maximum EIRP density for the service downlink is -41 dBW/Hz at a 90° angle of arrival corresponding to a 0° off-boresight steering angle using a 4 x 10 MHz channel[4]. In nominal operations with an 80 MHz channel, EIRP density is -50 dBW/Hz.

[4] Alternatively, EIRP can be reduced to 29 dBW for a single 10 MHz channel to meet the same -41 dBW/Hz EIRP density.

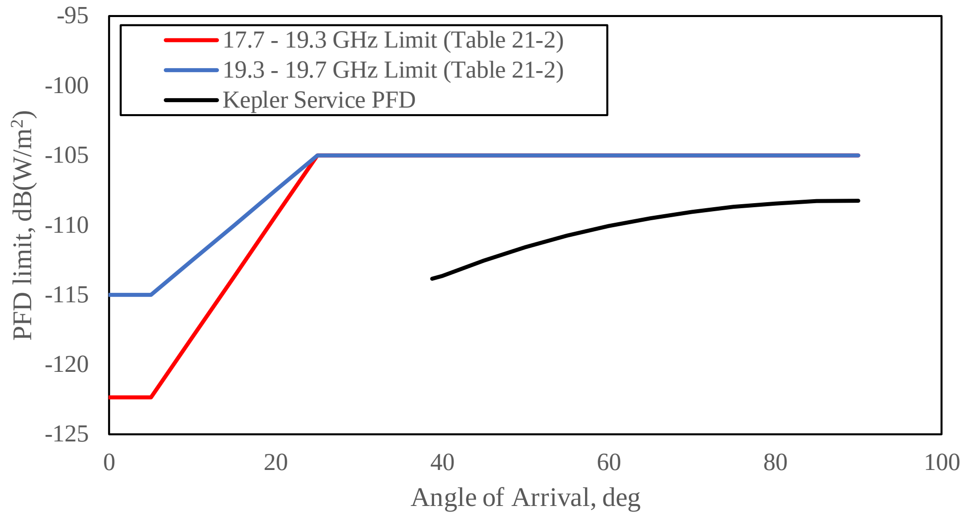

Figure 11: Service downlink PFD in 1 MHz reference bandwidth in the worst-case scenario of a 10MHz channel.

3. EPFD Compliance

Kepler hereby certifies that its proposed system will comply with all applicable EPFD limits stipulated under ITU-RR Article 22. For more information on mitigation of interference to geostationary space and earth stations, see Section III-B: GSO below.

III. Interference

A. Fixed

Kepler's System is designed to co-exist with terrestrial fixed service ("FS") operations. Uplink operations from both user terminals and gateways protect the fixed service by maintaining minimum horizon elevation angles of 38.8° and 10° respectively. These comparatively large elevation angles result in sufficiently sidelobe isolation to protect FS operations from harmful interference.

Kepler's system downlink EIRP levels comply with all applicable FCC and ITU power flux density ("PFD") limits designed to protect the FS operations from downlink interference, as provided in Section II-D-2: PFD Compliance.

B. GSO

The System complies with all applicable EPFD limits stipulated by ITU-RR Article 22 in its TTC, service, and feeder bands. For the service link, inline events with the GSO arc do not occur because of the high latitude and minimum elevation angle. This is depicted in Figure 12 for user terminals across the service region beginning at 55°N latitude. As shown, the minimum separation angle between a Kepler satellite and the GSO arc on the service link is 11.5°.

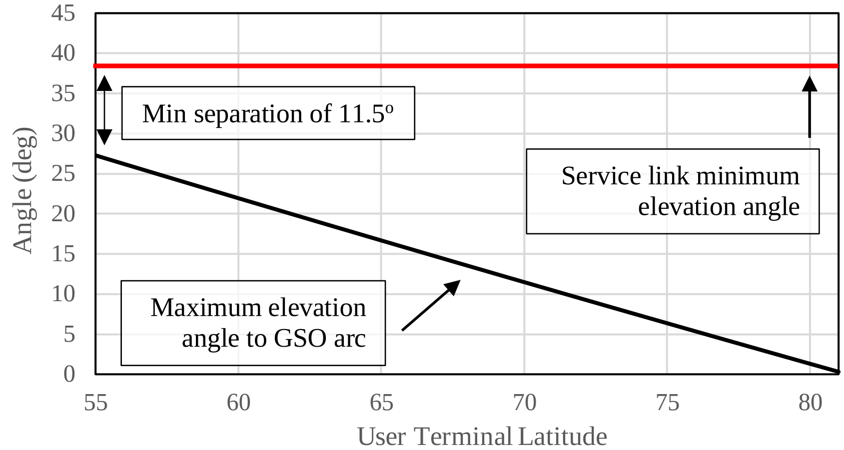

Figure 12: Minimum separation to GSO based on latitude of user terminal.

To meet EPFD limits on feeder links, Kepler actively employs exclusion masks on both its space and ground stations alike. As with the service link, the high latitude service regions (and corresponding high latitude gateway locations) inherently reduces the overall amount of time that Kepler space stations are viewed in-line with the GSO arc from the ground. This means that, even with exclusion masks employed, the aggregate sidelobe power from all ground and space stations into GSO systems is reduced over any prolonged duration. These exclusion masks are latitude dependent, as the minimum separation angle between Kepler satellites and the GSO arc is also latitude dependent, as shown in Figure 13.

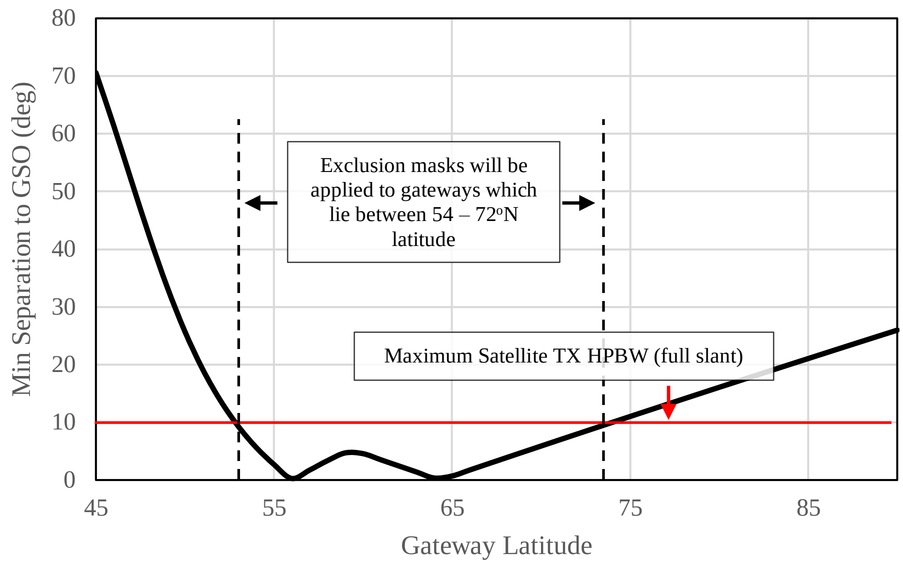

Figure 13: Minimum separation between Kepler satellite and GSO by latitude, as viewed from the ground.

For gateways that reside within 54 – 72°N, an exclusion mask threshold of 1.5° will be applied, in accordance with the GSO-separation limit of No. 21.2. Most importantly, this does not capture a gateway's ability to use satellite and site diversity to select available satellites in view which are not undergoing in-line events with the GSO arc. Satellite selection optimization will act to eliminate nearly all instances of in-line geometry with GSO, reducing the need for exclusion masks. Due to these compounding effects, Kepler has requested a waiver for presenting an exhaustive EPFD compliance analysis for the system, as the present EPFD limits set by Article 22 and Resolution 85 were developed in consideration of systems with much higher inherent potential for interference to GSO than the one proposed.

C. NGSO

The System is designed to facilitate coordination and spectrum sharing with other Ka and Ku-band NGSO FSS systems. Each satellite is capable of implementing a variety of coordination strategies that include, among other things, diversity, pointing,[5] selectable polarization, dynamic center frequency selection, dynamic channel sizing, and custom waveforms.[6] Kepler's incidence of in-line interference events is helped by the relatively limited number of satellite systems providing comparable services to Kepler's in the high north. Therefore, Kepler believes it is highly unlikely that it will be unable to find mutually agreeable strategies for coordination with other NGSO operators using the same bands, but if necessary, its network can easily accommodate the default band splitting strategy outlined by § 25.261. Kepler looks forward to working with system operators via good faith coordination, information exchange, and otherwise to ensure compatible operations and maximize the use of Ka and Ku-band spectrum and orbital resources.

IV. Orbital Debris Mitigation

The System has been designed with specific attention to space safety. The selected orbital configuration ensures timely de-orbit and safe disposal of satellites even in the event of a loss of maneuverability in the worst-case scenario during mission operations. Under nominal conditions, space stations are actively removed from orbit via on-board propulsion. Kepler is already registered with the Combined Space Operations Center (CSpOC) and will be capable of commanding any of the proposed satellites from its current ground station network in real-time after receiving a conjunction data message (CDMs) from CSpOC. The following sections discuss

[5] Because many authorized systems use emission bandwidths smaller than the entire authorized band in each direction of transmission, it is possible that there will be no frequency overlap during an in-line event. Even with frequency overlap, systems operating with different polarizations may not experience harmful interference.

[6] The satellite bent-pipe architecture is transparent to the type of waveforms implemented.

how Kepler's design and operational strategies will act to mitigate orbital debris risks, as required by Section 25.114(d)(14) of the Commission's rules.

D. Small Debris Mitigation

Kepler has assessed and limited the probability of a Kepler satellite becoming a source of debris through collisions with small debris (<10cm), demonstrating that the per-satellite probability of collision for these circumstances are less than 0.01, consistent with the requirements of NASA-STD-8719.14B. Supporting calculations for these claims are provided in the full orbital debris assessment report (ODAR) in Annex A.

E. Large Debris mitigation

Kepler has assessed the risk of collision with large debris (>10cm) and other space stations, and implemented several measures into the spacecraft design to minimize risk of collisions, including:

- Equipping each satellite with propulsive capabilities to allow for collision avoidance maneuvers to be performed in the event of a CDM
- Active disposal of the satellites through atmospheric reentry at their end of life
- Initially launching each satellite into a low-altitude orbit of 500 km with rapid decay times to allow system check-out prior to executing an orbit raising maneuver into the operational orbit
- Using a maximum operational orbit of 650km will still allow the satellites to naturally decay within the 25-year requirements in the event of propulsion failure while in the operational orbit
- Setting the parking orbit, operational orbit, and disposal orbit well outside the altitude range of the International Space Station (ISS) to minimize risk of collision

- Large satellite separation at orbit intersections

With these design considerations to minimize debris risk, Kepler has assessed and limited the probability of collision with large debris, demonstrating that the per-satellite probability of collision with large debris is less than 0.001, consistent within the requirements of NASA-STD-8719.14B. Moreover, Kepler also demonstrates that the aggregate probability of collision across the entire constellation is less than 0.001, exceeding the requirements of NASA-STD-8719.14B. Supporting calculations are provided in the full ODAR in Annex A.

F. Planned debris release

The Kepler satellites are designed such that there will be no planned release of debris during normal deployment, operations, and disposal.

G. Accidental explosions

Kepler has assessed and minimized the probability of accidental explosions both during and after competition of mission operations. The only foreseen risk of explosion would be a result of battery overheating and fracture in the event of failed pressure release safety, leading to the low probability of cell explosion. A failure mode and effects analysis (FMEA) was performed to demonstrate that the probability of this occurring is exceptionally low. The baseline electric propulsion system employed uses unpressurized and non-explosive propellant storage, and the indium/gallium solid/liquid fuel is chemically inert. No hypergolics or explosive actuation devices will be used. A backup propellant system will use pressurized, yet inert, Krypton/Xenon mixtures. FMEA describes how all the propulsion methods improbable failure modes would not result in catastrophic failure. Propellant budget and power system lifetimes are designed to incorporate these de-orbit operations. Supporting calculations pertaining to the risk of accidental explosion are provided in the full ODAR in Annex A.

H. Accuracy of orbital parameter maintenance

The Kepler satellites have on-board propulsion that will maintain orbital parameters to within the tolerances described in Table 4.

Table 4: Orbital parameter tolerances.

Orbital Parameter	Tolerance
Apogee + Perigee	± 25 km
Inclination	± 1.5^{o}
RAAN	± 10^{o}

I. Post-mission disposal

After the completion of nominal mission operations[7], the Kepler satellites will execute a de-orbit maneuver of each satellite from the operational (max) 650km circular orbit into a disposal orbit (300km circular orbit). This de-orbit maneuver will be performed in around 27 days, during which time conjunction avoidance maneuvers will be performed as needed. Once inserted into the disposal orbit, the spacecraft will naturally decay in approximately 0.12 years through atmospheric drag in an uncontrolled reentry. This de-orbit maneuver will allow for an expedited and orbit-normal transit through the orbit range of the ISS, minimizing the risk of collision. The maximum

[7] Nominal mission operations will conclude until the earliest of (i) battery capacity has reached 25% of BOL, or (ii) propulsion fuel levels have reached 25% of BOL. This is expected within 5-7 years of start of mission.

operational altitude of 650km also ensures that satellites will passively deorbit in under 25 years in the event where a failure occurs at the operational altitude that prevents active disposal. Supporting calculations for Kepler's post-mission disposal plans are provided in the full ODAR in Annex A, demonstrating that the System is fully compliant with NASA-STD-8719.14.

J. Casualty risk assessment

Kepler's analysis indicates that none of the on-board systems or components are expected to survive re-entry. Therefore, the proposed system presents a re-entry casualty risk of zero.

ENGINEERING CERTIFICATION

I hereby certify that I am the technically qualified person responsible for preparation of the engineering information contained in this Application of Kepler Communications Inc. for Market Access Authority for a Non-Geostationary Satellite Orbit System in Ka- and Ku-band Frequencies, that I am familiar with Part 25 of the Commission's rules, that I have either prepared or reviewed the engineering information submitted in this application, and that it is complete and accurate to the best of my knowledge and belief.

/s/ Jared Bottoms
Jared Bottoms
Head of Satellite Programs and Launch
Kepler Communications Inc.

Dated: 26 May 2020

ANNEX A: ORBITAL DEBRIS ASSESSMENT REPORT

Report Version: 1.0

Date of Delivery: 26/05/2020

Required Signatures: See Engineering Certification

Document Data is Not Restricted

This document contains no proprietary, ITAR, or export-controlled information

This report is presented in compliance with NASA-STD-8719.14B, Appendix A

DAS Software Version Used in Analysis: 3.0.1

A. Self-Assessment of the ODAR Using the Format in Appendix A.2 of NASA-STD 8719.14B

An assessment is provided in Table 5 in accordance with the assessment format provided in Appendix A.1 of NASA-STD-8719.14B.

Table 5: Orbital debris self-assessment report. Based upon ODAR version 1.0 dated 26/05/2020.

Requirement Number	Launch Vehicle				Spacecraft			Comments
	Compliant	Not Compliant	Incomplete	Standard Non-Compliant	Compliant	Not Compliant	Incomplete	
4.3-1.a			X		X			No debris released in LEO
4.3-1.b			X		X			No debris released in LEO
4.3-2			X		X			No debris released in GEO
4.4-1			X		X			No credible scenario of explosion
4.4-2			X		X			No credible scenario of explosion
4.4-3			X		X			No planned breakups
4.4-4			X		X			No planned breakups
4.5-1			X		X			Collision probability within requirements
4.5-2					X			Collision probability within requirements
4.6-1(a)			X		X			Re-entry in 24.8 yrs (max) from mission start
4.6-1(b)			X		X			Spacecraft not using storage orbit option
4.6-1(c)			X		X			Spacecraft not using direct retrieval
4.6-2			X		X			Spacecraft not in GEO
4.6-3			X		X			Spacecraft not between LEO and GEO
4.6-4			X		X			Disposal systems designed for >0.90 reliability
4.7-1			X		X			No controlled reentry
4.8-1					X			No tethers used

B. Assessment Report Format

This ODAR follows the format recommended in NASA-STD-8719.14B, Appendix A.1. This report includes the orbital debris mitigation approach for the spacecraft and not for any launch vehicle (Sections 2 through 8 in NASA-STD-8719.14B, Appendix A.1).

C. Project Management and Mission Overview

HQ Mission Directorate: N/A

Program Executive: N/A

Project Manager: Kepler Communications Inc.

Foreign Government or Space Agency Participation: N/A

NASA's Responsibilities: N/A

Senior Scientific & Management Personnel

- Nickolas Spina, Director Regulatory Affairs
- Jared Bottoms, Head of Satellite and Launch Programs
- Wen Cheng Chong, Chief Technology Officer
- Mark Michael, Chief Architect

Table 6: Mission design and development milestones.

Milestone	Est. Dates
Demonstration Mission PDR	2020
Demonstration Mission CDR	2021
Constellation PDR	2022
Constellation CDR	2022

Mission Overview:

The System will be a constellation of 360 NGSO small satellites in a near-polar (89.5^{o} inclination) circular orbit at an altitude of 600 ±50km. The 360 satellites will be spread across 12 planes, each with 30 satellites per plane. The mission lifetime for each satellite is a minimum of 5 years (up to 7 years) with a total maximum time-to-disposal after end of service of 24.8 years.

Anticipated Launch Vehicle & Launch Site: To be determined

Table 7: Proposed launch date and mission duration

Milestone	Est. Dates
Start of Launch Campaign	July 2023
Launch Completion	June 2024
End of First Network Operational Life	Between 2028 - 2030

Description of the Launch and Deployment Profile:

Satellites will be launched into a circular orbit with a 500km altitude and 89.5^{o} inclination. After systems check-out on each satellite, an orbit raising maneuver will be performed to raise the spacecraft into the operational orbit of 600 ± 50km. This orbit raising maneuver will require around 27 days to complete using on-board electric propulsion. The satellites will carry-out their mission for between 5-7 years until the earliest of (i) battery capacity has reduced to below 30% BOL, or (ii) propulsion fuel levels have reached 62% BOL. At this point, the satellites will execute a de-orbit maneuver which will transfer the satellites into a 300km circular disposal orbit. At this point, atmospheric effects will result in a rapid uncontrolled de-orbit within a maximum of 44 days.

Description of Spacecraft Maneuvering Capability:

Attitude control is provided by 3-axis reaction wheels and magnetorquers. Orbit control is provided by ion and hall effect electric propulsion.

Reason for Operational Orbit Selection:

The operational orbit was selected because (i) the relatively low altitude ensures de-orbit within the 25-year requirement in the event of propulsion failure, (ii) the relatively low debris environment at this altitude range minimizes risk of collision, and (iii) the altitude provides good coverage in the target service region, minimizing the number of satellites needed.

Potential Physical Interference with Other Operational Spacecraft:

The System has a risk of physical interference with any operational spacecraft at or below the maximum 650km operational altitude of Kepler's system. A truncated list of those considered to be a high concern of physical interference is included in Table 8.

Table 8: Identified operational spacecraft considered as high concern for physical interference (as of 26 May 2020).

Spacecraft	No. of Satellites	Altitude
International Space Station	1	350 - 450 km
Hubble Space Telescope	1	568 km
Spire Global Constellation	> 80	500 km
Planet Constellation	> 200	575 km
Starlink Constellation	417	350 – 400 km
OneWeb Constellation	74	550 - 600 km

D. Spacecraft Description

Physical Description:

The Kepler satellites are approximately 22 kg wet mass (approximately 0.7kg of fuel), with physical dimensions of around 360 x 750 x 115 mm^3 (stowed). The satellites contain a single long deployable and mechanically actuated solar array, with deployed planar dimension of approximately 0.45m^2. The payload consists of a bent-pipe Ku-to-Ka radio, a Ka-band service link phased array antenna (TX and RX) and two Ku-band feeder links which are electrically steered phased array antennas with mechanical off-axis assist (TX and RX). The antennas are physically separated from the main avionics package (BUS). The main avionics box contains the platform electronics, including battery bank, on-board processing, propulsion, electrical power, and attitude control systems. The payload control is maintained in a separate and distinct enclosure for thermal safety of the battery and other critical health systems. An illustration of the spacecraft in mission operation configuration is shown in Figure 14, and some key spacecraft parameters are provided in Table 9. Nadir is in the positive z-direction, whereas the velocity vector varies about the z-axis depending on the right ascension of the ascending node (RAAN) to maximize power generation, as shown in Figure 14.

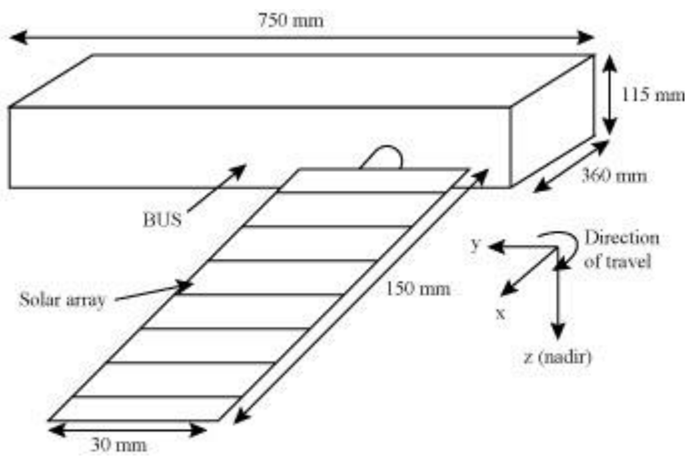

Figure 14: Conceptual design of Kepler satellite

Table 9: Spacecraft parameters.

Parameter	Value or Description
Total Mass at Launch	22kg
Dry Mass at Launch	21.3kg
Fluids	N/A
Propulsion	Electric propulsion (ion or hall)
Attitude Control	3-axis reaction wheels and magnetorquers
Nominal Attitude	See Figure 14
Range Safety or Pyrotechnics	N/A
Electrical Generation and Storage	20 string solar array (172W EOL OAP) and 15 cell battery bank (290 W-hr capacity)
Radioactive Materials	No radioactive materials on board
Proximity Operations	N/A

The load bearing structure for each spacecraft is comprised of a single aluminum structural piece (machined or additive manufactured) that provides a primary thermal transport as well as the entirety of the load bearing structure for load transfer to/from the launch vehicle. The antenna assemblies, payload and avionics enclosures are fastened and affixed to this structure. Solar arrays are deployed through spring loaded hinges and actuated through non-explosive actuators. Actuation of solar arrays is enabled at a set time post separation from the launch vehicle

E. Assessment of Spacecraft Debris Released During Normal Operations

Not applicable as there is no intentional release of debris.

F. Assessment of Spacecraft Intentional Breakups and Potential for Explosions

Potential causes of spacecraft breakup during deployment and mission operations:

There is no foreseen scenario in which the spacecraft would breakup during normal deployment and operations.

Summary of failure modes and effects analysis of all credible failure modes which may lead to an accidental explosion:

The first foreseen risk of explosion would be battery overheating and the resulting low probability of cell explosion. A FMEA (failure mode and effects analysis) was done to demonstrate the combined, mutually exclusive failures that must occur to result in the potential for accidental explosion of the batteries. Seven independent scenarios were analyzed to consider the risk that battery explosion could occur, with an exceptionally low cumulative probability that explosion will occur (see subheading *Supporting Rationale and FMEA details* below).

Detailed plan for any designed spacecraft breakup, including explosions and intentional collisions: Not applicable as there are no planned breakups.

List of components which are passivated at EOM: N/A

Rational for all items which are required to be passivated, but cannot due to their design: The batteries will not be passivated upon entering the disposal orbit as a means to provide telemetry, control, and collision avoidance capability until end of life with re-entry. This is not deemed a risk, as the total combined stored energy remains small at less than 1,044 kJ.

Assessment of spacecraft compliance with Requirements 4.4-1 through 4.4-4:

Requirement 4.4-1: Limiting the risk to other space systems from accidental Orbital Debris explosions during deployment and mission operations which in orbit about Earth or the Moon:

For each spacecraft and launch vehicle orbital stage employed for a mission (i.e., every individual free-flying structural object), the program or project shall demonstrate, via failure mode and effects analyses or equivalent analyses, that the integrated probability of explosion for all creditable failures modes of each spacecraft and launch vehicle is less than 0.001 (excluding small particle impacts).

Compliance statement: The expected probability of any accidental explosion caused by failure on-board the spacecraft is predicted to be significantly less than that requires. Kepler maintains this as a top requirement for ongoing trade-studies and FMEA.

Required Probability: 0.001
Expected Probability: << 0.001

Supporting Rationale and FMEA Details:

Battery explosion:

Effect: The following presents all failure modes that may theoretically result in battery explosion, with the possibility of debris generation. However, in the unlikely event of battery explosion the low stored chemical energy combined with being located internal to the spacecraft container minimizes the risk of debris generation because of the lack of penetration energy.

Probability: Extremely low. Battery explosion in all failure modes requires two simultaneous faults or systematic user failure that are extremely low probability in themselves.

Failure mode 1: Short circuit
Mitigation: Functional testing of charge/discharge, as well as environmental testing including shock, vibration, thermal cycling, and vacuum testing to prove no internal short circuit sensitivity exists. Cells & protection circuits are COTS which extensive flight heritage across many ranges of space vehicles
Combined Faults Required: Failure of environmental testing **AND** failure for functional testing to discover short circuit sensitivities **AND** failure of COTS vendor in testing components.

Failure Mode 2: Above-nominal current draw inducing battery thermal rise
Mitigation: Thermal cycling on cells to test upper temperature limits, testing of batteries at above-nominal discharge levels. All cells and assemblies include features to allow for prevention and physical safety for conditions such as over-temperature, over-discharge, etc.
Combined Faults Required: Inaccurate spacecraft thermal design **AND** over-current failure to detect off-nominal discharge rates

Failure Mode 3: Terminal contact with conductors not at battery voltage levels causing above-nominal current draw.

Mitigation: Qualification-test short circuit protection on each circuit, design of battery holders to ensure to unintended conductor contact without mechanical failure, minimize risk of mechanical failures via shock, vibration testing.
Combined Faults Required: Mechanical failure induced short circuit **AND** failure of over-current protection.

Failure Mode 4: Inoperable vents on battery holders.
Mitigation: Battery vents are not inhibited by the battery holder design or the spacecraft.
Combined Faults Required: Assembler fails to install proper venting **AND** failure to identify inoperable venting during final inspection.

Failure Mode 5: Batteries are crushed during operation
Mitigation: Spacecraft is tested to simulate launch loadings to ensure no damage to cells. All components in proximity to battery are constrained to the rigid primary structure.
Combined Faults Required: A catastrophic mechanical failure has occurred via impact **AND** the failure must be sufficient to cause a crushing of the batteries, leading to an internal short circuit **AND** the satellite must be in a naturally sustained orbit at the time of crushing.

Failure Mode 6: Low level current leakage or short circuit through battery holder due to moisture-induced degradation of insulators
Mitigation: Non-conductive delrin for battery holders, operation in vacuum thus moisture cannot affect insulators
Combined Faults Required: Failure of battery insulators **AND** dislocation of batteries in holder **AND** failure to detect fault in environmental testing

Failure Mode 7: Thermal rise due to space environment and high discharge combined
Mitigation: Spacecraft thermal design to prevent situation, testing of batteries above expected operating conditions, use of COTS components with flight heritage. FDIR implements latching current limiters and active load-shed capability in response to high temperature event detection.
Combined Faults Required: Inaccurate thermal design **AND** failure to detect fault in thermal cycling and environmental testing **AND** failure of COTS supplier to provide qualified batteries.

Requirement 4.4-2: Design for passivation after completion of mission operations while in orbit about Earth or the Moon:

Design of all spacecraft and launch vehicle orbital stages shall include the ability and a plan to either 1) deplete all onboard sources of stored energy and disconnect all energy generation sources when they are no longer required for mission operations or postmission disposal or 2) control to a level which cannot cause an explosion or deflagration large enough to release orbital debris or break up the spacecraft. The design of depletion burns and venting should minimize the probability of accidental collision with tracked objects in space.

Compliance statement: At EOM, a de-orbit maneuver will be executed to bring each spacecraft to a disposal altitude of approximately 300km. Once the spacecraft has reached this altitude atmospheric drag will result in an uncontrolled re-entry in under 44 days without any further need for de-orbit maneuvers.

Requirement 4.4-3: Limiting the long-term risk to other space systems from planned breakups for Earth and lunar missions.
Compliance statement: Not applicable as there are no planned breakups.

Requirement 4.4-4: Limiting the short-term risk to other space systems from planned breakups.
Compliance statement: Not applicable as there are no planned breakups.

G. Assessment of Spacecraft Potential for On-Orbit Collisions

Assessment of spacecraft compliance with Requirements 4.5-1 and 4.5-2:

Requirement 4.5-1: Limiting debris generated by collisions with large objects when in Earth orbit:

For each spacecraft and launch vehicle orbital stage in or passing through LEO, the program or project shall demonstrate that, during the orbital lifetime of each spacecraft and orbital stage, the probability of accidental collision with space objects larger than 10 cm in diameter does not exceed 0.001. For spacecraft and orbital stages passing through the protected region +/- 200km and +/- 15 degrees of geostationary orbit, the probability of accidental collision with space objects larger than 10 cm in diameter shall not exceed 0.001 when integrated over 100 years from time of launch.

Compliance Statement: A probability of collision analysis was performed using NASA's DAS 3.0.1 software. This analysis was done for the entire constellation sustained over a constellation duration of 15 years. With a minimum lifetime of 5 years, the constellation of 360 operational orbiting spacecraft will be built and replenished through the launch of at maximum 1,080 total satellites over the course of 15 years.

In the nominal scenario, the mission is subdivided into five stages, shown in Table 10.

Table 10: Stages of a nominal spacecraft mission.

Stage	Duration	Description
Stage 1: LEOP	1 month	Spacecraft is launched to a circular parking orbit of 500km to perform system check-outs.
Stage 2: Orbit Raising	27 days	Spacecraft performs an orbit raising maneuver to raise the satellite from the parking orbit into the 600 ± 50km orbit.
Stage 3: Operations	5 – 7 yrs	Spacecraft performs its mission from its operational orbit.
Stage 4: Disposal Burn	18 days	Spacecraft performs a de-orbit maneuver to enter the disposal orbit (300km circular orbit)
Stage 5: Disposal	2 months	Spacecraft naturally decays due to atmospheric drag.

In the nominal scenario, the risk of collision is assumed to be **zero** because of the inclusion of propulsion on-board each spacecraft, consistent with long-standing practices by the Commission.[8]

[8] See *Mitigation of Orbital Debris in the New Space Age, Notice of Proposed Rulemaking*, FCC 18-159 (Nov. 19, 2018), ¶ 26.

Propulsion allows a spacecraft to respond to a Conjunction Data Message (CDM) event within a very small response time, typically within a single orbit. Kepler is registered for CDM notifications with CSpOC and will have the capacity to command a satellite in real-time upon learning of a CDM. When a CDM is responded to within a single orbit, collisions are deterministic rather than probabilistic; there is near certainty on the range between spacecraft during a conjunction event when a projection is made within a single orbit. Because of this ability to respond to deterministic collisions rather than probabilistic, a functioning propulsion system implies a zero probability of collision.

In addition to this nominal scenario, the worst-case scenario was considered where the spacecraft propulsion system fails at some point over the course of the mission. The following driving assumptions were used across this worst-case scenario analysis:

- Over the course of 15 years, 1,080 Kepler spacecraft are placed into operational orbit
- Propulsion reliability equals 99%, equal to 11 failed satellites without propulsion
- Propulsion failure is only considered to occur at the end-of-mission lifetime, equating to an orbit epoch year of 2028
- Spacecraft is assumed to be in a 'tumbling' configuration (see *Assessment of Spacecraft Postmission Disposal Plans and Procedures,* as well as Figure 14) resulting in an effective drag area of $0.28m^2$
- An altitude of 650km was assumed, representing the worst-case disposal time for the orbit range of 600 ± 50km as well as debris environment
- A lifetime of 24.8 years is assumed per spacecraft, consistent with the maximum orbital decay time (see *Assessment of Spacecraft Postmission Disposal Plans and Procedures*)

Based on these assumptions, the following collision risk statistics were derived from DAS:

- Single spacecraft collision probability over 24.8 years when propulsion fails in operational orbit: 1.88E-05
- Aggregate constellation probability of collision: 2.07E-04

It is worth emphasizing that Kepler has conducted this analysis over the **total sum of foreseen spacecraft** needed to replenish the constellation over a 15-year span while demonstrating compliance to a requirement intended to guide the design of **a single spacecraft**.

Collision Risk Output:

The DAS logs below show the inputs and outputs from the above analysis:

==============
Run Data
==============

INPUT

Space Structure Name = PolarSat
Space Structure Type = Payload
Perigee Altitude = 650.000 (km)
Apogee Altitude = 650.000 (km)
Inclination = 89.500 (deg)
RAAN = 0.000 (deg)
Argument of Perigee = 0.000 (deg)
Mean Anomaly = 0.000 (deg)
Final Area-To-Mass Ratio = 0.0130 (m^2/kg)
Start Year = 2028.000 (yr)
Initial Mass = 22.000 (kg)
Final Mass = 21.300 (kg)
Duration = 24.800 (yr)
Station-Kept = False
Abandoned = True

OUTPUT

Collision Probability = 1.8788E-05
Returned Message: Normal Processing
Date Range Message: Normal Date Range
Status = Pass

==============

=============== End of Requirement 4.5-1 ===============

Requirement 4.5-2: Limiting debris generated by collisions with small objects when operating in Earth or lunar orbit**:**

For each spacecraft, the program or project shall demonstrate that, during the mission of the spacecraft, the probability of accidental collision with orbital debris and meteoroids sufficient to prevent compliance with the applicable postmission disposal requirements is less than 0.01.

It is assumed for this analysis that small objects less than 10cm in characteristic diameter are not capable of tracking. Because of this, propulsion is incapable of responding to a CDM because of the lack of *a priori* knowledge of a conjunction event. For this analysis, the following assumptions were used:

- Nominally operating spacecraft are assumed to be in a ‘nominal’ configuration (see *Assessment of Spacecraft Postmission Disposal Plans and Procedures,* as well as Figure 14) resulting in an worst case average effective drag area of $0.28m^2$
- Nominally operating spacecraft have an assumed 7-year lifetime,

Based on these assumptions, the following collision risk statistics were derived from DAS:

- Single nominally operating spacecraft collision probability resulting in PMD failure: 2.3621E-5

Collision Risk Output:

The DAS logs below show the inputs and outputs from the above analysis, where the single nominally operating spacecraft collision probability is the cumulative probabilities listed below:

```
==================================================
Spacecraft = Gen2-1
Critical Surface = Avionics +Z
==================================================

**INPUT**

    Apogee Altitude = 650.000000 (km)
    Perigee Altitude = 650.000000 (km)
    Orbital Inclination = 89.500 (deg)
    RAAN = 0.000000 (deg)
    Argument of Perigee = 0.000000 (deg)
    Mean Anomaly = 0.000000 (deg)
    Final Area-To-Mass = 0.013000 (m^2/kg)
    Initial Mass = 22.000000 (kg)
    Final Mass = 22.000000 (kg)
    Station Kept = No
    Start Year = 2023.000000 (yr)
    Duration = 7.000000 (yr)
    Orientation = Fixed Oriented
    CS Areal Density = 2.900000 (g/cm^2)
    CS Surface Area = 0.07 (m^2)
    Vector = (1.000000 (u), 0.000000 (v), 0.000000 (w))
    CS Pressurized = No
    Outer Wall 1 Density: 2.9 [g/cm^2] Separation 1.000 [cm]
    Outer Wall 2 Density: 2.9 [g/cm^2] Separation 2.000 [cm]

**OUTPUT**

    Probability of Penetration = 1.432E-08
    Returned Error Message: Normal Processing
    Date Range Error Message: Normal Date Range

==================================================
Spacecraft = Gen2-1
Critical Surface = Avionics -Z
==================================================

**INPUT**

    Apogee Altitude = 650.000000 (km)
    Perigee Altitude = 650.000000 (km)
    Orbital Inclination = 89.500 (deg)
    RAAN = 0.000000 (deg)
    Argument of Perigee = 0.000000 (deg)
    Mean Anomaly = 0.000000 (deg)
    Final Area-To-Mass = 0.013000 (m^2/kg)
```

Initial Mass = 22.000000 (kg)
Final Mass = 22.000000 (kg)
Station Kept = No
Start Year = 2023.000000 (yr)
Duration = 7.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.900000 (g/cm^2)
CS Surface Area = 0.07 (m^2)
Vector = (-1.000000 (u), 0.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 2.9 [g/cm^2] Separation 1.000 [cm]
Outer Wall 2 Density: 2.9 [g/cm^2] Separation 2.000 [cm]

OUTPUT

Probability of Penetration = 7.7800E-09
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

==
Spacecraft = Gen2-1
Critical Surface = Avionics +X
==

INPUT

Apogee Altitude = 650.000000 (km)
Perigee Altitude = 650.000000 (km)
Orbital Inclination = 89.500 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.013000 (m^2/kg)
Initial Mass = 22.000000 (kg)
Final Mass = 22.000000 (kg)
Station Kept = No
Start Year = 2023.000000 (yr)
Duration = 7.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.900000 (g/cm^2)
CS Surface Area = 0.03 (m^2)
Vector = (0.000000 (u), 1.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 2.9 [g/cm^2] Separation 4.000 [cm]
Outer Wall 2 Density: 2.9 [g/cm^2] Separation 8.000 [cm]

OUTPUT

Probability of Penetration = 1.3138E-05
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

==
Spacecraft = Gen2-1
Critical Surface = Avionics +X
==

INPUT

Apogee Altitude = 650.000000 (km)
Perigee Altitude = 650.000000 (km)
Orbital Inclination = 89.500 (deg)
RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.013000 (m^2/kg)
Initial Mass = 22.000000 (kg)
Final Mass = 22.000000 (kg)
Station Kept = No
Start Year = 2023.000000 (yr)
Duration = 7.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.900000 (g/cm^2)
CS Surface Area = 0.03 (m^2)
Vector = (0.000000 (u), -1.000000 (v), 0.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 2.9 [g/cm^2] Separation 4.000 [cm]
Outer Wall 2 Density: 2.9 [g/cm^2] Separation 8.000 [cm]

OUTPUT

Probability of Penetration = 5.6536E-10
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

==
Spacecraft = Gen2-1
Critical Surface = Avionics +Y
==

INPUT

Apogee Altitude = 650.000000 (km)
Perigee Altitude = 650.000000 (km)
Orbital Inclination = 89.500 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.013000 (m^2/kg)
Initial Mass = 22.000000 (kg)
Final Mass = 22.000000 (kg)
Station Kept = No
Start Year = 2023.000000 (yr)
Duration = 7.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.900000 (g/cm^2)
CS Surface Area = 0.03 (m^2)
Vector = (0.000000 (u), 0.000000 (v), 1.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 2.9 [g/cm^2] Separation 10.000 [cm]
Outer Wall 2 Density: 2.9 [g/cm^2] Separation 8.000 [cm]
Outer Wall 3 Density: 2.9 [g/cm^2] Separation 5.000 [cm]

OUTPUT

Probability of Penetration = 1.3794E-06
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

==
Spacecraft = Gen2-1
Critical Surface = Avionics +Y
==

INPUT

Apogee Altitude = 650.000000 (km)
Perigee Altitude = 650.000000 (km)
Orbital Inclination = 89.500 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Mean Anomaly = 0.000000 (deg)
Final Area-To-Mass = 0.013000 (m^2/kg)
Initial Mass = 22.000000 (kg)
Final Mass = 22.000000 (kg)
Station Kept = No
Start Year = 2023.000000 (yr)
Duration = 7.000000 (yr)
Orientation = Fixed Oriented
CS Areal Density = 2.900000 (g/cm^2)
CS Surface Area = 0.05 (m^2)
Vector = (0.000000 (u), 0.000000 (v), -1.000000 (w))
CS Pressurized = No
Outer Wall 1 Density: 2.9 [g/cm^2] Separation 4.000 [cm]

OUTPUT

Probability of Penetration = 9.0804E-06
Returned Error Message: Normal Processing
Date Range Error Message: Normal Date Range

H. Assessment of Spacecraft Postmission Disposal Plans and Procedures

Description of spacecraft disposal option selected:

The satellite will execute a de-orbit maneuver for a fast reentry in approximately 1/4 of a year from EOM. In the event of a failure of a de-orbit maneuver, the spacecraft will naturally decay within a maximum of 24.8 years after end of mission.

Identification of all systems or components required to accomplish any postmission disposal maneuvers. Plan for any spacecraft maneuvers required to accomplish postmission disposal:

Postmission disposal requires the following subsystems: propulsion, attitude control to maintain a stable attitude during maneuver, power system (solar panels, batteries, EPS) to supply electrical energy to the propulsion and attitude control system, and TT&C link to maintain ground communications during maneuver.

Calculation of area-to-mass ratio after postmission disposal, if the controlled reentry option is not selected

The nominal spacecraft configuration is dependent upon the orbit RAAN and varies from a maximum of 0.49m^2 (velocity along y-axis and solar panel perpendicular to direction of travel) to a minimum of 0.04m^2 (velocity along y-axis and solar panel parallel to direction of travel), with reference to Figure 14. The tumble configuration is essentially an average of all surface areas and reflects the situation when the spacecraft has an uncontrolled attitude.

EOL Spacecraft Mass:	21.3kg	
Cross-Sectional Area:	Max Nominal Configuration:	0.49 m^2
	Min Nominal Configuration:	0.04 m^2
	Tumble Configuration:	0.28 m^2
Area to Mass Ratio:	Max Nominal Configuration:	0.023 m^2/kg
	Min Nominal Configuration:	0.002 m^2/kg
	Tumble Configuration:	0.013 m^2/kg

If appropriate, preliminary plan for spacecraft controlled reentry: Not applicable

Assessment of spacecraft compliance with Requirements 4.6-1 and 4.6-2:

Requirement 4.6-1: Disposal for space structures in or passing through LEO:

A spacecraft or orbital stage with a perigee altitude below 2,000 km shall be disposed of by one of the following three methods:

a. Atmospheric reentry option:

1. Leave the space structure in an orbit in which natural forces will lead to atmospheric reentry within 25 years after the completion of mission; or

2. Maneuver the space structure into a controlled de-orbit trajectory as soon as practical after completion of mission.

b. Storage orbit option: Maneuver the space structure into an orbit with a perigee altitude above 2000 km and ensure its apogee altitude will be below 19,700 km, both for a minimum of 100 years.

c. Direct retrieval: Retrieve the space structure and remove it from orbit within 10 years after completion of mission.

Compliance Statement: At the end of operational life, each spacecraft will perform a de-orbit maneuver which will reduce the spacecraft into the disposal orbit with an altitude of 300km and zero eccentricity. From here, the satellite will reenter the atmosphere through natural decay in less than 1 year. DAS was used to estimate the expected on-orbit lifetime from this disposal orbit, which resulted in reentry within 0.12yrs (44 days), well within the 25-year guidelines. The DAS logs below show the inputs and outputs from the above analysis:

INPUT

Start Year = 2028.000000 (yr)
Perigee Altitude = 300.000000 (km)
Apogee Altitude = 300.000000 (km)
Inclination = 89.500000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Area-To-Mass Ratio = 0.013000 (m^2/kg)

OUTPUT

Orbital Lifetime from Startyr = 0.120465 (yr)
Time Spent in LEO during Lifetime = 0.120465 (yr)
Last year of Propagation = 2028 (yr)
Returned Error Message: Object reentered

The worst-case scenario of a failure resulting in an inability to perform the de-orbit maneuver was also considered. This scenario considered the worst-case scenario where failure occurs at the maximum altitude of 650km. DAS was used to estimate the expect on-orbit lifetime in this worst-case scenario, which resulted in a reentry in 24.8 years, or slightly less than the 25-year requirement. The DAS logs below show the inputs and outputs from the above analysis:

INPUT

Start Year = 2028.000000 (yr)
Perigee Altitude = 650.000000 (km)
Apogee Altitude = 650.000000 (km)
Inclination = 89.500000 (deg)
RAAN = 0.000000 (deg)
Argument of Perigee = 0.000000 (deg)
Area-To-Mass Ratio = 0.013000 (m^2/kg)

OUTPUT

Orbital Lifetime from Startyr = 24.788501 (yr)
Time Spent in LEO during Lifetime = 24.788501 (yr)
Last year of Propagation = 2052 (yr)
Returned Error Message: Object reentered

Requirement 4.6-2: Disposal for space structures near GEO:
Compliance Statement: Not applicable

Requirement 4.6-3: Disposal for space structures between LEO and GEO
Compliance Statement: Not applicable

Requirement 4.6-4: Reliability of postmission disposal maneuver operations in Earth orbit:

NASA space programs and projects shall ensure that all postmission disposal operations to meet Requirements 4.6-1, 4.6-2, and/or 4.6-3 are design for a probability of success as follows:

a. *Be no less than 0.90 at EOM, and*

b. *For controlled reentry, the probability of success at the time of reentry burn must be sufficiently high so as not to cause a violation of Requirement 4.7-1 pertaining to limiting the risk of human casualty.*

Compliance Statement: Onboard propulsion to execute the de-orbit maneuver for the nominal fast reentry scenario will be designed to ensure a greater than 0.99 probability of success at EOM. In addition to on-ground testing to ensure this reliability, the propulsion system will be thoroughly tested in the 500km parking orbit prior to executing the orbit raising maneuver. Failing that, the relatively low altitude should still ensure reentry within the 25-year guidelines if the propulsion system fails to operate as designed.

I. Assessment of Spacecraft Reentry Hazards

Assessment of spacecraft compliance with Requirement 4.7-1

Requirement 4.7-1: Limit the risk of human casualty:

The potential for human casualty is assumed for any object with an impacting kinetic energy in excess of 15 joules:

a. *For uncontrolled reentry, the risk of human casualty from surviving debris shall not exceed 0.0001 (1:10,000).*

b. *For controlled reentry, the selected trajectory shall ensure that no surviving debris impact with a kinetic energy greater than 15 joules is closer than 370 km from foreign landmasses, or is within 50 km from the continental U.S., territories of the U.S., and the permanent ice pack of Antarctica.*

c. *For controlled reentries, the product of the probability of failure to execute the reentry burn and the risk of human casualty assuming uncontrolled reentry shall not exceed 0.0001 (1:10,000).*

Compliance Statement: Analysis performed using DAS shows that no part of the satellite is expected to survive the uncontrolled reentry (option a), therefore the risk of human casualty of a single satellite, and the aggregate constellation is 0. The DAS input and outputs are shown below, including a detailed description of spacecraft components.

==============

Project Data

==============

INPUT

Space Structure Name = Gen2-1

Space Structure Type = Payload

Perigee Altitude = 650.000000 (km)

Apogee Altitude = 650.000000 (km)

Inclination = 89.500000 (deg)

RAAN = 0.000000 (deg)

Argument of Perigee = 0.000000 (deg)

Mean Anomaly = 0.000000 (deg)

Area-To-Mass Ratio = 0.013000 (m^2/kg)

Start Year = 2023.000000 (yr)

Initial Mass = 23.000000 (kg)

Final Mass = 22.000000 (kg)

Duration = 5.000000 (yr)

Station Kept = False

Abandoned = False

PMD Perigee Altitude = 300.000000 (km)

PMD Apogee Altitude = 650.000000 (km)

PMD Inclination = 89.500000 (deg)

PMD RAAN = 0.000000 (deg)

PMD Argument of Perigee = 0.000000 (deg)

PMD Mean Anomaly = 0.000000 (deg)

OUTPUT

Suggested Perigee Altitude = 300.000000 (km)

Suggested Apogee Altitude = 650.000000 (km)

Returned Error Message = Passes LEO reentry orbit criteria.

Released Year = 2029 (yr)

Requirement = 61

Compliance Status = Pass

==============

=============== End of Requirement 4.6 ===============

05 25 2020; 14:53:10PM   *********Processing Requirement 4.7-1

Return Status : Passed

***********INPUT****

Item Number = 1

name = Gen2-1

quantity = 1

parent = 0

materialID = 8

type = Box

Aero Mass = 22.000000

Thermal Mass = 22.000000

Diameter/Width = 0.360000

Length = 0.740000

Height = 0.140000

name = Primary Structure

quantity = 1

parent = 1

materialID = 9

type = Flat Plate

Aero Mass = 4.140000

Thermal Mass = 3.500000

Diameter/Width = 0.360000

Length = 0.740000

name = HDRMs

quantity = 4

parent = 2

materialID = 59

type = Cylinder

Aero Mass = 0.160000

Thermal Mass = 0.160000

Diameter/Width = 0.030000

Length = 0.030000

name = Avionics Box

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 12.300000

Thermal Mass = 1.000000

Diameter/Width = 0.360000

Length = 0.360000

Height = 0.140000

name = Avionics Package

quantity = 2

parent = 4

materialID = 19

type = Box

Aero Mass = 2.000000

Thermal Mass = 2.000000

Diameter/Width = 0.070000

Length = 0.200000

Height = 0.030000

name = RWs Assembly

quantity = 3

parent = 4

materialID = 58

type = Cylinder

Aero Mass = 0.300000

Thermal Mass = 0.300000

Diameter/Width = 0.040000

Length = 0.030000

name = Payload Box

quantity = 1

parent = 4

materialID = 8

type = Box

Aero Mass = 3.600000

Thermal Mass = 1.200000

Diameter/Width = 0.100000

Length = 0.220000

Height = 0.083000

name = Payload Cards

quantity = 8

parent = 7

materialID = 8

type = Box

Aero Mass = 0.300000

Thermal Mass = 0.300000

Diameter/Width = 0.090000

Length = 0.090000

Height = 0.015000

name = Propulsion Enclosure

quantity = 1

parent = 4

materialID = 8

type = Box

Aero Mass = 0.800000

Thermal Mass = 0.600000

Diameter/Width = 0.090000

Length = 0.140000

Height = 0.090000

name = Propellant Canister

quantity = 1

parent = 9

materialID = 8

type = Cylinder

Aero Mass = 0.200000

Thermal Mass = 0.200000

Diameter/Width = 0.080000

Length = 0.080000

name = SA motor

quantity = 2

parent = 4

materialID = 59

type = Cylinder

Aero Mass = 0.100000

Thermal Mass = 0.100000

Diameter/Width = 0.020000

Length = 0.060000

name = Battery

quantity = 1

parent = 4

materialID = 77

type = Box

Aero Mass = 1.800000

Thermal Mass = 1.800000

Diameter/Width = 0.090000

Length = 0.320000

Height = 0.050000

name = Antenna Array

quantity = 1

parent = 1

materialID = 8

type = Box

Aero Mass = 1.000000

Thermal Mass = 1.000000

Diameter/Width = 0.300000

Length = 0.300000

Height = 0.010000

**************OUTPUT****

Item Number = 1

name = Gen2-1

Demise Altitude = 77.999420

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Primary Structure

Demise Altitude = 72.072021

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = HDRMs

Demise Altitude = 53.024700

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Avionics Box

Demise Altitude = 76.195274

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Avionics Package

Demise Altitude = 64.815315

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = RWs Assembly

Demise Altitude = 60.098995

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Payload Box

Demise Altitude = 70.903732

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Payload Cards

Demise Altitude = 64.874725

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Propulsion Enclosure

Demise Altitude = 72.183800

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Propellant Canister

Demise Altitude = 68.798172

Debris Casualty Area = = 0.000000

Impact Kinetic Energy = = 0.000000

****

name = SA motor

Demise Altitude = 68.744598

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Battery

Demise Altitude = 74.197617

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

name = Antenna Array

Demise Altitude = 73.316193

Debris Casualty Area = 0.000000

Impact Kinetic Energy = 0.000000

****

=============== End of Requirement 4.7-1 ===============

J. Assessment for Tether Missions

Not applicable as there are no tethers in the mission.