RAD-AI: Radiation-Mitigated Edge AI for Autonomous Space Operations

UCCS CubeSat Design Project

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1. Executive Summary

RAD-AI is a 6U CubeSat mission designed to validate commercial-off-the-shelf (COTS) artificial intelligence hardware with radiation mitigation techniques in Low Earth Orbit (LEO). The mission addresses a critical capability gap in autonomous space computing: radiation-hardened processors such as the BAE RAD750 lack the computational throughput required for real-time AI inference, while commercial AI accelerators deliver adequate performance but exhibit unacceptable failure rates in the space radiation environment [1, 2].

The primary objective of RAD-AI is to demonstrate AI-driven autonomous computing with radiation-aware operation for a minimum of 12 months in LEO. The payload couples a SiFive U74 RISC-V quad-core processor with an FPGA-based AI accelerator, implementing Triple Modular Redundancy (TMR) in software, selective tantalum shielding, watchdog timers, and Error Detection and Correction (EDAC) memory protection [3, 4]. Unlike previous AI CubeSat demonstrations such as ESA's Phi-Sat-1, which performed static cloud detection, RAD-AI demonstrates real-time radiation event detection and autonomous mode switching—the system detects high-radiation zones including the South Atlantic Anomaly (SAA) and automatically adjusts processing parameters to prevent data corruption [5, 6].

The mission targets the 2025-2027 development window before NASA's High Performance Spaceflight Computing (HPSC) processors become widely available [7]. RAD-AI provides empirical flight data on COTS AI degradation patterns and mitigation effectiveness, informing both LEO constellation designs and HPSC integration strategies for deep-space missions supporting Artemis and Mars Sample Return [8, 9].

The spacecraft utilizes a hybrid development approach combining a COTS 6U bus with a custom AI payload. The 6U form factor (20×10×34 cm, <14 kg) allocates 1.5U for the AI payload, 1.5U for power systems, and 3U for the spacecraft bus [10]. Power requirements average approximately 36 W with AI processing peaks of 15-30 W, supplied by GaAs solar arrays generating 45 W at beginning of life and a 60 Wh lithium-ion battery [11].

Total development cost is estimated at $100,000-$120,000, with launch services provided through NASA's CubeSat Launch Initiative (CSLI) valued at approximately $250,000 [12]. The project timeline spans three years: design and development (months 1-24), integration and testing (months 17-30), and flight operations (months 31-42).

2. Introduction

2.1 Objective

The primary objective of the RAD-AI mission is to demonstrate AI-driven autonomous computing with radiation-aware operation in Low Earth Orbit for a duration of 12 months. This demonstration shall validate that COTS AI hardware, when coupled with appropriate radiation mitigation techniques, can provide reliable real-time inference capabilities in the LEO radiation environment.

Secondary objectives for the mission are as follows:

1. **Measure radiation effects on mitigated COTS hardware**: Quantify Single Event Upset (SEU) rates and Total Ionizing Dose (TID) accumulation on the RISC-V processor and FPGA AI accelerator with TMR software mitigation and selective shielding implemented [13].

2. **Validate real-time radiation detection and adaptive processing**: Demonstrate that the onboard system can detect radiation environment changes with latency less than one minute and autonomously transition between operational modes to maintain data integrity [14].

3. **Generate flight dataset for future designs**: Produce a comprehensive dataset of AI performance metrics, radiation event correlations, and degradation patterns to inform future LEO constellation computing architectures and NASA HPSC integration strategies [7].

4. **Track AI accuracy degradation over mission lifetime**: Continuously monitor inference accuracy on standardized benchmarks to characterize the relationship between cumulative radiation exposure and AI model performance degradation [15].

Success criteria for the RAD-AI mission are defined at three levels:

- **Minimum Success (30 days)**: Spacecraft achieves stable orbit, establishes ground communications, and returns valid telemetry data including radiation sensor readings and AI processor health metrics.

- **Baseline Success (6 months)**: Continuous AI processing operations with documented SAA passages, validated autonomous mode transitions, and preliminary degradation trend data.

- **Full Success (12 months)**: Complete mission duration with demonstrated radiation-aware adaptive behavior, comprehensive degradation curves, and sufficient data for statistical analysis of mitigation effectiveness.

2.2 Problem Statement

Autonomous onboard computing is rapidly becoming essential for advanced space missions. NASA's Artemis program identifies autonomous precision landing as a cornerstone technology for lunar surface operations [8]. The Mars Sample Return campaign requires onboard hazard avoidance and terrain-relative navigation capabilities that cannot rely on ground control due to one-way light time delays ranging from 6 to 44 minutes [9]. NASA's Technology Area 4 (TA4) explicitly identifies "radiation-tolerant autonomy" as a critical capability gap requiring near-term investment [16].

The fundamental challenge is a performance-reliability tradeoff in space computing hardware. Radiation-hardened processors such as the BAE RAD750, which has flown on over 100 missions including Mars rovers and deep space probes, operate at approximately 200 MHz with no dedicated AI acceleration capabilities [1]. This computational throughput is insufficient for real-time computer vision applications requiring 10-30 Hz inference rates on high-resolution imagery [17]. In contrast, commercial AI processors such as the NVIDIA Jetson platform deliver computational performance exceeding 500 GFLOPS, but radiation testing has demonstrated unacceptable vulnerability to the space environment. Total ionizing dose testing of NVIDIA Jetson Nano GPUs showed functional degradation beginning at approximately 20 krad (Si), with recent characterization of the Jetson Orin platform concluding it is only "marginally sufficient for a three-year LEO mission" [2, 18].

NASA has recognized this capability gap and initiated the High Performance Spaceflight Computing (HPSC) program to develop a radiation-hardened processor providing 100x performance improvement over current flight heritage computers [7]. The HPSC program, with first processors expected in early 2025 and flight qualification extending through 2027, represents the authoritative long-term solution for deep-space autonomous computing [19]. However, a transition gap exists: near-term LEO constellations, commercial Earth observation satellites, and university research missions require autonomous computing solutions during the 2025-2027 period before HPSC becomes widely available and affordable [20].

Recent commercial developments validate market demand for COTS-with-mitigation solutions. Cosmic Shielding Corporation successfully demonstrated radiation-shielded NVIDIA Jetson Orin NX hardware on the Aethero CubeSat platform in 2024, attracting both commercial and military customers [21]. The Air Force Research Laboratory (AFRL) actively partners with industry on COTS radiation mitigation approaches, confirming that government and defense stakeholders recognize the need for near-term solutions [21]. The OPTOS CubeSat, launched in 2019, demonstrated that a radiation-tolerant collaborative computer architecture using COTS components with system-level hardening could operate for three years in LEO with "no System Error," proving that system reliability can exceed individual component reliability through appropriate design techniques [22].

RAD-AI addresses this validated need by providing empirical orbit data on COTS AI performance with mitigation during the critical 2025-2027 window. The mission is positioned as complementary bridge technology rather than a competitor to HPSC—the flight data generated will directly inform HPSC integration strategies while enabling near-term missions that cannot wait for fully radiation-hardened solutions [7, 20].

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3. Project Planning and Management

The RAD-AI project follows a hybrid development strategy combining COTS spacecraft bus components with a custom AI payload, consistent with CubeSat 101 best practices for first-time developers seeking to balance risk, cost, and schedule [23]. The development approach leverages flight-proven subsystems where available while focusing custom development effort on the novel radiation-mitigated AI payload that represents the mission's primary contribution.

Development Philosophy

Following CubeSat 101 guidance, three hardware configurations shall be developed [23, Ch. 2]:

1. **Engineering Test Unit (ETU)**: First integrated assembly used for practice integration, fit checks, and initial testing. Mistakes discovered on the ETU prevent costly rework on flight hardware.

2. **FlatSat Configuration**: Components mounted on a flat board without flight structure, enabling rapid software development, debugging, and troubleshooting with full accessibility to all interfaces.

3. **Flight Units (2)**: Two complete flight-qualified spacecraft are fabricated. Producing multiple flight units simultaneously reduces per-unit cost and provides redundancy for launch opportunities.

Project Timeline

The RAD-AI project spans approximately 42 months from initiation through completion of flight operations, organized into three major phases:

Phase 1: Design and Development (Months 1-16)

Milestone	Timeframe	Key Activities
Concept Development	Months 1-3	Requirements definition, trade studies, preliminary design
Preliminary Design Review (PDR)	Months 7-10	Subsystem designs, interface definitions, test planning
Critical Design Review (CDR)	Months 11-16	Detailed designs, procurement initiation, software development

Phase 2: Integration and Test (Months 17-30)

Milestone	Timeframe	Key Activities
ETU Integration	Months 17-20	First hardware integration, initial functional testing
FlatSat Testing	Months 18-22	Software validation, HIL simulation, radiation injection testing
Flight Unit Integration	Months 21-26	Flight hardware assembly, workmanship verification
Environmental Testing	Months 24-28	Vibration, thermal-vacuum, EMI/EMC per NASA GEVS [24]
Flight Readiness Review (FRR)	Months 28-30	Final verification, documentation completion, delivery preparation

Phase 3: Launch and Operations (Months 31-42)

Milestone	Timeframe	Key Activities
Delivery to Integrator	Month 30	CubeSat delivery to CSLI mission integrator
Launch Campaign	Months 31-33	Integration with dispenser, launch vehicle integration
Commissioning	Months 33-34	On-orbit checkout, sensor calibration, baseline establishment
Science Operations	Months 34-42	Continuous autonomous operation, data collection, analysis

Budget Management

Following CubeSat 101 guidance that "Budget includes 10%+ reserve for unexpected events" [23, Ch. 2, p. 17], the RAD-AI project maintains a 20% reserve allocation given the novel nature of the AI payload development.

Table 3.1: RAD-AI Budget Summary

Category	Estimated Cost	Notes
AI Payload Development	$35,000-40,000	RISC-V processor, FPGA, sensors, shielding
Spacecraft Bus (COTS)	$40,000-50,000	Blue Canyon XACT or equivalent [25]
Environmental Testing	$10,000-15,000	Vibration, thermal-vacuum, radiation
Ground Segment	$5,000-8,000	Station upgrades, cloud services
Travel and Reviews	$3,000-5,000	PDR, CDR, FRR, delivery
Reserve (20%)	$18,000-24,000	Contingency for unexpected costs
Total Development	**$100,000-120,000**	Excluding launch services
Launch Services (CSLI)	~$250,000 value	Provided by NASA at no cost [12]

CSLI Compliance

The RAD-AI mission design incorporates CSLI requirements from project initiation [23]:

- **No pyrotechnics**: All deployable mechanisms use burn-wire release systems
- **RF licensing**: Amateur radio frequencies selected; FCC application submitted within 30 days of manifesting per CSLI requirements
- **NASA benefit**: Mission directly supports NASA Technology Area 4 autonomy objectives and Artemis/Mars Sample Return technology development [16]
- **Orbital debris compliance**: 400-600 km altitude ensures natural decay within 25 years per NASA-STD-8719.14 [26]
- **Flexibility**: Mission accepts any CSLI-compatible LEO orbit; no specific inclination or altitude requirements

Risk Management

Key programmatic risks and mitigation strategies are summarized below:

Risk	Likelihood	Impact	Mitigation
CSLI selection not achieved	Medium	High	Identify backup commercial launch options; maintain ORS/NRO rideshare eligibility
Component availability delays	Medium	Medium	Early procurement; identify alternate sources; maintain component buffer stock
AI payload development challenges	Medium	High	Extensive FlatSat testing; incremental capability demonstration; descope options defined
Ground station communication gaps	Low	Medium	SatNOGS network backup; automated retry protocols; onboard data storage
Budget overrun	Medium	Medium	20% reserve; phased procurement; value engineering reviews

4. Problem Specifications

4.1 Project Requirements

The RAD-AI requirements are derived from four primary sources: (1) NASA CSLI and CubeSat Design Specification constraints [10, 23], (2) mission-specific science and technology demonstration objectives, (3) the LEO radiation environment at 400-600 km altitude [27], and (4) spacecraft bus capabilities and interfaces. Requirements are organized into five subsystem categories following the structure established in the reference design [28].

4.1.1 Communications Requirements List

The communications subsystem shall provide reliable bidirectional data transfer between the spacecraft and ground segment, supporting both command uplink and science/telemetry downlink functions.

Table 4.1: Communications Requirements

ID	Requirement	Rationale	Verification
COM-1	The communications subsystem shall provide command uplink capability at a minimum data rate of 1200 bps.	Ground operators must be able to upload commands, software updates, and configuration changes throughout the mission [29].	Demonstration
COM-2	The communications subsystem shall provide telemetry and science data downlink at a minimum data rate of 9600 bps.	Target daily data volume of 100 MB requires sustained downlink throughput during ground station passes [30].	Demonstration
COM-3	The communications subsystem shall operate in the UHF amateur radio band (430-440 MHz) with appropriate licensing.	Amateur frequencies minimize licensing complexity and enable SatNOGS network backup; compliant with FCC Part 97 [31].	Inspection
COM-4	The RF transmitter shall not exceed 8 W output power.	Power budget allocation; compliant with amateur radio power limits for satellite operations [31].	Test
COM-5	The communications subsystem shall support store-and-forward operation with minimum 72-hour data buffering.	Ensures no data loss during periods without ground contact or during anomaly recovery [32].	Demonstration
COM-6	The antenna system shall deploy reliably from stowed configuration after orbit insertion.	Launch vehicle constraints require stowed antennas; deployment must succeed for mission viability [10].	Test
COM-7	The communications subsystem shall implement AX.25 protocol compatible with amateur radio ground infrastructure.	Enables use of existing amateur satellite tracking networks and SatNOGS ground stations [33].	Demonstration

ID	Requirement	Rationale	Verification
COM-8	The spacecraft shall transmit a unique identifier beacon at minimum 60-second intervals when power-positive.	Supports orbital tracking and identification; required for amateur radio license compliance [31].	Demonstration

4.1.2 Power Requirements List

The electrical power subsystem shall generate, store, regulate, and distribute electrical power to all spacecraft subsystems throughout all mission phases and operational modes.

Table 4.2: Power Requirements

ID	Requirement	Rationale	Verification
POW-1	The power subsystem shall generate minimum 45 W at beginning of life (BOL) from solar arrays.	AI payload peak demand of 30 W plus bus loads of 10 W plus 20% margin requires 48 W; 45 W BOL provides adequate margin with degradation [11].	Test
POW-2	The power subsystem shall provide minimum 60 Wh energy storage capacity.	Supports 45-minute eclipse operations at average power plus one full AI processing cycle during eclipse [34].	Test
POW-3	The power subsystem shall regulate bus voltage to 5.0 V ± 0.25 V for payload interfaces.	RISC-V processor and FPGA require stable 5V supply; ±5% tolerance per component specifications [3].	Test
POW-4	The power subsystem shall regulate bus voltage to 3.3 V ± 0.17 V for sensor interfaces.	Sensors and low-power peripherals operate at 3.3V nominal [35].	Test
POW-5	The power subsystem shall provide battery charge regulation preventing overcharge above 4.2 V per cell.	Lithium-ion safety requirement; prevents thermal runaway and capacity degradation [36].	Test
POW-6	The power subsystem shall implement undervoltage load shedding at 3.0 V per cell.	Protects battery from deep discharge damage; ensures recovery capability [36].	Demonstration
POW-7	The power subsystem shall support autonomous load shedding in priority sequence during power-negative conditions.	AI payload is lowest priority; critical bus functions maintained during anomalies [34].	Demonstration

ID	Requirement	Rationale	Verification
POW-8	The power subsystem shall provide current-limited outputs with overcurrent protection on all power rails.	Prevents fault propagation; isolates failed components [37].	Test

4.1.3 Telemetry and Control Requirements List

The telemetry and control subsystem shall acquire sensor data, manage onboard data storage, execute autonomous control algorithms, and coordinate spacecraft operational modes.

Table 4.3: Telemetry and Control Requirements

ID	Requirement	Rationale	Verification
TEL-1	The telemetry system shall sample radiation sensors (RADFETs, particle detector) at minimum 1 Hz during science operations.	Captures SAA transitions and SEU correlation data with sufficient temporal resolution [27].	Demonstration
TEL-2	The telemetry system shall sample housekeeping sensors (temperatures, voltages, currents) at minimum 0.1 Hz.	Adequate for thermal and power trending; reduces data volume [38].	Demonstration
TEL-3	The telemetry system shall timestamp all data to ±1 second accuracy using GPS-synchronized onboard time.	Enables correlation of radiation events with orbital position and ground-based space weather data [27].	Test
TEL-4	The telemetry system shall store minimum 7 days of full-rate science and housekeeping data onboard.	Provides margin for ground station outages and anomaly investigation [32].	Inspection
CON-1	The control system shall execute autonomous mode transitions between Normal, Protected, and Safe modes based on radiation environment assessment.	Core mission objective: demonstrate radiation-aware adaptive computing [14].	Demonstration
CON-2	The control system shall complete radiation environment assessment and mode transition decision within 60 seconds of sensor update.	Ensures timely response to SAA entry; prevents data corruption [14].	Test
CON-3	The control system shall implement watchdog timers with automatic reset capability for all processors.	Recovers from radiation-induced processor lockups without ground intervention [4].	Demonstration

ID	Requirement	Rationale	Verification
CON-4	The control system shall maintain operational state and resume autonomous operation following any processor reset.	Ensures mission continuity; no ground intervention required for nominal recovery [39].	Demonstration
CON-5	The C&DH system shall be physically isolated from the AI payload processor.	Prevents AI payload faults from affecting critical spacecraft functions; defense-in-depth architecture [40].	Inspection

4.1.4 Mechanical Requirements List

The mechanical subsystem shall provide structural support, environmental protection, and mechanical interfaces for all spacecraft components while maintaining compliance with CubeSat Design Specification and launch vehicle requirements.

Table 4.4: Mechanical Requirements

ID	Requirement	Rationale	Verification
MEC-1	The spacecraft shall conform to 6U CubeSat form factor: 20.0 cm × 10.0 cm × 34.05 cm maximum dimensions.	CubeSat Design Specification Rev. 14 compliance; required for CSLI dispensers [10].	Inspection
MEC-2	The spacecraft total mass shall not exceed 14.0 kg.	CDS Rev. 14 6U mass limit [10].	Test
MEC-3	The spacecraft center of mass shall be located within 2.0 cm of geometric center in all axes.	CDS Rev. 14 requirement for dispenser compatibility and deployment stability [10].	Test
MEC-4	The structure shall withstand quasi-static loads of 7.5 g in all axes.	Derived from NASA GEVS launch environment requirements [24].	Analysis
MEC-5	The structure shall withstand random vibration environment of 14.1 Grms per NASA GEVS.	Qualification level for CubeSat dispensers [24].	Test
MEC-6	All deployable components shall be constrained during launch with positive retention requiring active release.	CDS requirement; prevents premature deployment that could interfere with launch vehicle or other payloads [10].	Inspection
MEC-7	Rail contact surfaces shall be hard anodized aluminum with surface roughness ≤1.6 μm Ra.	CDS dispenser interface requirement [10].	Inspection
MEC-8	The structure shall provide mounting provisions for 2 mm tantalum shielding around AI processor and FPGA.	Selective shielding approach for radiation mitigation [4].	Inspection

ID	Requirement	Rationale	Verification
MEC-9	The structure shall accommodate radiation sensor placement with unobstructed field of view for particle detector.	Accurate radiation measurement requires minimized structural interference [27].	Inspection

4.1.5 Thermal Requirements List

The thermal subsystem shall maintain all spacecraft components within operational temperature limits throughout all mission phases including eclipse transients and end-of-life degraded conditions.

Table 4.5: Thermal Requirements

ID	Requirement	Rationale	Verification
THE-1	The thermal subsystem shall maintain battery temperature between 0°C and 45°C during all operational phases.	Lithium-ion cell operational limits; charging prohibited below 0°C [36].	Analysis, Test
THE-2	The thermal subsystem shall maintain AI processor temperature between -20°C and 70°C during operation.	RISC-V processor operational temperature range [3].	Analysis, Test
THE-3	The thermal subsystem shall maintain FPGA temperature between -40°C and 85°C during operation.	Lattice CrossLink-NX industrial temperature grade limits [41].	Analysis, Test
THE-4	The thermal subsystem shall limit AI processor temperature rate of change to ≤5°C per minute.	Prevents thermal stress on die attach and solder joints [42].	Analysis
THE-5	The thermal design shall dissipate minimum 30 W from the AI payload during peak processing.	AI processing thermal load with margin for worst-case hot conditions [11].	Analysis
THE-6	The thermal subsystem shall provide minimum 10 W heater capacity for battery survival heating.	Maintains battery above 0°C during cold-case eclipse scenarios [36].	Analysis, Test
THE-7	All spacecraft components shall survive non-operational temperature range of -40°C to +60°C.	Encompasses all credible mission scenarios including safe mode and anomaly conditions [24].	Test
THE-8	The thermal design shall accommodate increased power dissipation from TMR processing overhead.	TMR implementation increases computational load and associated heat generation [4].	Analysis

Requirements Traceability

A complete requirements verification matrix including verification methods, success criteria, and verification status tracking is provided in Appendix C. Requirements are traced to parent mission objectives and derived from applicable standards including:

- CubeSat Design Specification Rev. 14 [10]
- NASA General Environmental Verification Standard (GEVS) [24]
- NASA-STD-8719.14 Process for Limiting Orbital Debris [26]
- NASA CubeSat 101 [23]
- FCC Part 97 Amateur Radio Service Rules [31]

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5. Conceptual Design

The conceptual design phase evaluated multiple alternatives for mission focus, electrical architecture, and structural configuration. Each alternative was assessed against mission objectives, technical feasibility, cost constraints, and schedule risk. The evaluation process follows CubeSat 101 guidance emphasizing simplicity, use of flight-proven components, and designs that do not push CDS envelope limits [43].

5.1 Missions

A variety of mission concepts were evaluated for the RAD-AI payload. The mission selection is critical in determining spacecraft requirements, component selection, development complexity, and ultimately the contribution to the field of radiation-tolerant autonomous computing.

Mission Concept 1: Static Cloud Detection (Phi-Sat-1 Replication)

This concept replicates the ESA Phi-Sat-1 mission approach, implementing a convolutional neural network for cloud detection in Earth observation imagery [44]. The AI system would classify image tiles as cloudy or clear, enabling selective downlink of scientifically valuable cloud-free imagery.

Advantages:

- Proven flight heritage from Phi-Sat-1 (2020)
- Well-documented neural network architecture and training methodology
- Straightforward success metrics (classification accuracy)
- Lower development risk due to available reference implementation

Disadvantages:

- Does not advance state-of-the-art beyond existing demonstration
- Static application with no radiation-adaptive behavior
- Limited contribution to autonomous computing for deep-space missions
- No real-time radiation environment awareness

Assessment: While technically feasible and low-risk, this concept offers limited novelty and does not address the core mission objective of demonstrating radiation-aware autonomous computing.

Mission Concept 2: Radiation-Aware Autonomous Computing (Selected)

This concept implements real-time radiation environment monitoring coupled with autonomous processing mode adaptation. The AI system performs star-field tracking for attitude determination while simultaneously monitoring radiation sensors and orbital position to detect high-radiation zones. Upon detecting elevated radiation (e.g., SAA entry), the system autonomously transitions to protected processing modes with increased redundancy [45].

Advantages:

- Novel contribution: first demonstration of AI-driven radiation-adaptive computing
- Directly addresses NASA TA4 "radiation-tolerant autonomy" gap [46]
- Generates flight data valuable for HPSC integration planning
- Demonstrates capability critical for Artemis and Mars Sample Return missions
- Multiple operational modes provide rich dataset for analysis

Disadvantages:

- Higher development complexity than static applications
- Requires integration of radiation sensors with AI decision system
- Mode transition logic requires careful design to prevent thrashing
- More complex verification and validation process

Assessment: Selected as primary mission. The increased complexity is justified by the significant contribution to autonomous space computing technology and strong alignment with NASA strategic objectives.

Mission Concept 3: Radiation Effects Monitoring Only

This concept focuses exclusively on characterizing radiation effects on COTS AI hardware without implementing adaptive behaviors. The system would operate continuously in a single mode while logging all radiation events and correlating them with AI inference errors [47].

Advantages:

- Simpler implementation than adaptive system
- Clean dataset without mode transition confounds
- Directly comparable to ground radiation testing results

Disadvantages:

- Does not demonstrate autonomous adaptation capability
- Passive monitoring provides less operational value
- Limited advancement toward deep-space autonomy goals
- Similar data available from dedicated radiation monitoring payloads

Assessment: While valuable for hardware characterization, this concept does not demonstrate the autonomous adaptation that represents RAD-AI's unique contribution.

Mission Concept 4: High-Performance Computing Benchmark

This concept maximizes AI processing performance to establish CubeSat computing benchmarks, running standardized inference workloads (MLPerf) continuously without radiation adaptation [48].

Advantages:

- Provides standardized performance metrics
- Enables direct comparison with ground-based systems
- Useful for commercial constellation computing planning

Disadvantages:

- No radiation awareness or adaptation
- Risk of premature failure without protective modes
- Does not advance autonomous computing for harsh environments
- Benchmark focus may not translate to operational applications

Assessment: Rejected due to lack of radiation-aware features essential to mission objectives.

Mission Selection Summary

Table 5.1: Mission Concept Evaluation Matrix

Criterion	Weight	Concept 1	Concept 2	Concept 3	Concept 4
NASA Alignment	25%	2	5	3	2
Technical Novelty	25%	1	5	3	2
Feasibility	20%	5	3	4	4
Data Value	20%	2	5	4	3
Risk Level	10%	5	3	4	2
Weighted Score	100%	**2.5**	**4.4**	**3.5**	**2.6**

Mission Concept 2 (Radiation-Aware Autonomous Computing) was selected based on superior alignment with NASA objectives, highest technical novelty, and greatest value of generated flight data.

5.2 Electrical

The electrical subsystem conceptual design evaluated alternatives for each major component: main processor, AI accelerator, radiation sensors, communications, and power system.

Main Processor Options

Three processor architectures were evaluated for the main computing platform:

Option A: ARM Cortex-M Series (Microcontroller)

ARM Cortex-M4/M7 microcontrollers offer low power consumption (<100 mW), extensive flight heritage in CubeSat missions, and mature development ecosystems [49]. However, limited computational resources (single core, <500 MHz, no FPU on lower variants) preclude real-time AI inference at required rates.

Option B: ARM Cortex-A Series (Application Processor)

ARM Cortex-A53/A72 processors provide significantly higher performance (multi-core, >1 GHz, NEON SIMD) with moderate power consumption (1-5 W) [50]. These processors can run full Linux operating systems, simplifying software development. However, radiation susceptibility is comparable to Option C without the architectural advantages for fault tolerance.

Option C: RISC-V Architecture (Selected)

RISC-V processors, specifically the SiFive U74 quad-core (1.5 GHz), offer several advantages for radiation-tolerant computing [51]:

- Open instruction set architecture enables custom radiation-hardening extensions
- Simpler pipeline architecture facilitates TMR implementation
- Growing space industry adoption (TRISAT-R demonstrated fault-tolerant RISC-V in 2022) [52]
- No licensing fees reduce program cost
- Active research community developing radiation-tolerant RISC-V variants

Table 5.2: Main Processor Comparison

Parameter	ARM Cortex-M7	ARM Cortex-A72	SiFive U74 (RISC-V)
Clock Speed	480 MHz	1.8 GHz	1.5 GHz
Cores	1	4	4
Power (typical)	0.1 W	4 W	3 W
AI Capability	Very Limited	Moderate	Moderate
TMR Feasibility	Limited	Moderate	Good
Flight Heritage	Extensive	Limited	Emerging (TRISAT-R)
Cost	$15	$50	$40
Selection	Rejected	Backup	**Primary**

The SiFive U74 RISC-V processor was selected as the primary computing platform based on superior TMR implementation potential and alignment with emerging space computing trends.

AI Accelerator Options

Three AI acceleration approaches were evaluated:

Option A: GPU-Based (NVIDIA Jetson)

NVIDIA Jetson platforms offer exceptional AI performance (>500 GFLOPS on Jetson Nano, >100 TOPS on Orin) with mature software ecosystem (CUDA, TensorRT) [53]. However, radiation testing shows vulnerability beginning at ~20 krad TID, and high power consumption (10-40 W) challenges CubeSat power budgets [54]. Recent Cosmic Shielding demonstrations with Jetson Orin required significant shielding mass [55].

Option B: FPGA-Based (Selected)

Field Programmable Gate Arrays enable custom AI inference engine implementation with several advantages [56]:

- Inherent radiation tolerance superior to GPUs (configuration memory can be protected)
- Lower power consumption (2-5 W for equivalent inference performance)
- Reconfigurable architecture enables in-flight algorithm updates
- SEU effects localized to specific logic blocks rather than systemic failures
- Lattice CrossLink-NX specifically designed for edge AI applications

Option C: Dedicated AI ASIC (Google Edge TPU)

Application-specific AI accelerators offer excellent performance-per-watt but limited radiation characterization data and fixed functionality prevent adaptation [57].

Table 5.3: AI Accelerator Comparison

Parameter	NVIDIA Jetson Nano	Lattice CrossLink-NX	Google Edge TPU
AI Performance	472 GFLOPS	10 TOPS (INT8)	4 TOPS
Power	10 W	2 W	2 W
TID Tolerance	~20 krad	~50 krad (est.)	Unknown
Reconfigurable	No	Yes	No
Development Effort	Low	High	Medium
Selection	Rejected	**Primary**	Rejected

The Lattice CrossLink-NX FPGA was selected for AI acceleration based on superior radiation tolerance, reconfigurability, and acceptable power consumption.

Radiation Sensor Options

Radiation environment monitoring requires sensors for both Total Ionizing Dose (TID) accumulation and Single Event Effects (SEE):

TID Monitoring: RADFETs

Radiation-sensitive Field Effect Transistors provide real-time TID measurement through threshold voltage shift proportional to accumulated dose [58]. Multiple RADFETs with different oxide thicknesses enable dose measurement across the expected mission range (5-30 krad). Selected for flight implementation.

SEE Monitoring: Particle Telescope

A simple cosmic ray telescope using stacked silicon detectors enables correlation between particle flux and observed SEU events [59]. This sensor provides direct measurement of the instantaneous radiation environment for mode transition decisions. Selected for flight implementation.

Supplementary: Orbital Position

GPS-derived orbital position enables prediction of SAA encounters using onboard radiation belt models (AP-8, AE-8) [60]. This predictive capability supplements real-time sensor data for mode transition decisions.

Communications Options

Option A: S-Band System

S-band (2.0-2.3 GHz) offers higher data rates (>1 Mbps) and smaller antennas but requires more complex ground station infrastructure and licensing [61].

Option B: UHF System (Selected)

UHF amateur band (430-440 MHz) provides adequate data rate (9600 bps) for mission requirements with simplified licensing (amateur radio) and access to global SatNOGS ground station network for backup [62]. Lower cost and complexity align with budget constraints.

Option C: Combined S-Band/UHF

Dual-band system provides high-rate S-band downlink with UHF backup. Rejected due to mass, power, and cost impact exceeding available margins.

Power System Options

Option A: Body-Mounted Solar Cells

Solar cells mounted directly on spacecraft body panels provide simplicity and reliability but limited power generation (~15 W for 6U) insufficient for AI payload requirements [63].

Option B: Deployable Solar Arrays (Selected)

Deployable solar panels provide required power generation (~45 W) within 6U form factor. GaAs triple-junction cells selected for high efficiency (~30%) and radiation tolerance [64].

Option C: Deployable Arrays with Battery Augmentation

Larger battery capacity (>100 Wh) enables extended eclipse operations but exceeds mass budget. Rejected in favor of operational constraints during eclipse.

5.3 Structure

The structural design evaluated configuration options for the 6U form factor, shielding approaches, and thermal management strategies.

6U Configuration Options

Option A: 3U + 3U Stacked

Two 3U sections stacked along the long axis, with AI payload in one section and bus in the other. Simple integration but suboptimal mass distribution and thermal coupling [65].

Option B: Integrated 6U (Selected)

Single integrated 6U structure with distributed subsystem placement optimized for mass properties and thermal management. Payload and bus components interleaved for optimal center of mass location.

Option C: Modular Design

Separable payload and bus modules enabling independent development. Rejected due to interface complexity and mass penalty.

Table 5.4: Volume Allocation

Subsystem	Volume Allocation	Rationale
AI Payload (processor, FPGA, sensors)	1.5U	Core mission hardware with shielding
Power System (batteries, regulators)	1.5U	60 Wh battery pack, power electronics
Communications (radio, antenna stowage)	0.5U	UHF transceiver and deployment mechanism
ADCS (star tracker, magnetorquers)	0.5U	Attitude determination and control
C&DH and harness	0.5U	Flight computer, data storage, wiring
Structure and margin	1.5U	Primary structure, brackets, contingency
Total	**6.0U**	

Shielding Approach Options

Option A: Full Enclosure Shielding

Complete aluminum or tantalum enclosure around all electronics. Provides uniform protection but significant mass penalty (>2 kg for meaningful attenuation) [66].

Option B: Selective Shielding (Selected)

Targeted shielding of most radiation-sensitive components (AI processor, FPGA, critical memory) with 2 mm tantalum (equivalent to ~6 mm aluminum). Provides 10x dose reduction for ~500 g mass penalty [67].

Option C: No Dedicated Shielding

Rely entirely on software mitigation (TMR, EDAC) without physical shielding. Lowest mass but highest SEU rate and TID accumulation. Rejected as insufficient for 12-month mission duration.

Table 5.5: Shielding Mass Budget

Component	Shielding Material	Thickness	Mass
RISC-V Processor	Tantalum	2 mm	180 g
FPGA	Tantalum	2 mm	220 g
Critical Memory	Tantalum	1 mm	80 g
Total Shielding			**480 g**

Thermal Management Options

Option A: Passive Thermal Control (Selected)

Thermal design using surface coatings, MLI blankets, and conductive coupling to radiator surfaces. Adequate for expected thermal loads with appropriate component placement [68].

Option B: Active Thermal Control

Heat pipes or pumped fluid loops for high-power components. Rejected due to complexity and mass penalty not justified by thermal requirements.

Option C: Thermoelectric Cooling

Peltier coolers for AI processor thermal management. Rejected due to power consumption and limited effectiveness in vacuum.

---

6. Electrical Testing Methods and Results

The electrical testing program verifies functionality, performance, and environmental tolerance of all electrical subsystems. Testing follows NASA General Environmental Verification Standard (GEVS) guidelines and CubeSat 101 recommendations [69, 70]. Per CubeSat 101, all testing shall be completed at least one month before Mission Readiness Review [70, Ch. 2].

6.1 Sensor Integration and Calibration

Test Objective

Verify that all sensors interface correctly with the flight computer and provide accurate measurements within specified tolerances.

Test Configuration

The sensor integration test utilizes the FlatSat configuration with all flight sensors connected to the engineering model processor. Sensors under test include:

- BME280 temperature/pressure/humidity sensor (I2C interface)
- RADFETs (analog interface via ADC)
- DS18B20 digital temperature probes (1-Wire interface)
- INA219 current/voltage monitors (I2C interface)
- Camera module (CSI interface)

Test Procedure

1. Power system initialization and voltage verification
2. Sequential sensor activation and communication verification
3. Sensor polling at operational rates (1 Hz for radiation sensors, 0.1 Hz for housekeeping)
4. Data logging for minimum 300 seconds (300 samples at 1 Hz)
5. Comparison of measured values against calibrated reference instruments

Expected Results

Table 6.1: Sensor Calibration Acceptance Criteria

Sensor	Parameter	Accuracy Requirement	Reference Standard
BME280	Temperature	±1.0°C	NIST-traceable thermometer
BME280	Pressure	±1.0 hPa	Calibrated barometer
DS18B20	Temperature	±0.5°C	NIST-traceable thermometer
INA219	Voltage	±1%	Calibrated multimeter
INA219	Current	±2%	Calibrated shunt reference
RADFET	Dose	±10%	Calibration curve from vendor

Preliminary Results

Breadboard testing of sensor interfaces has verified communication protocols and basic functionality. Full calibration testing is scheduled for Month 18 (ETU integration phase).

6.2 Radiation Testing

Test Objective

Characterize radiation tolerance of the AI payload assembly including processor, FPGA, and memory under representative space radiation conditions.

Test Facility

Radiation testing shall be conducted at the Lawrence Berkeley National Laboratory (LBNL) 88-Inch Cyclotron, which provides proton beams suitable for space radiation effects testing [71]. Test cost is approximately $2,500 per day.

Test Configuration

The Device Under Test (DUT) consists of the integrated AI payload board including:

- SiFive U74 RISC-V processor
- Lattice CrossLink-NX FPGA with AI inference engine
- DDR4 memory with EDAC protection
- Supporting power regulation circuitry

The DUT is mounted in a vacuum chamber with thermal control. Beam parameters simulate LEO proton environment.

Test Procedure

TID Testing:

1. Configure DUT in operational mode running AI inference benchmark
2. Expose to proton beam at accelerated dose rate (100-1000 rad/hr)
3. Monitor AI inference accuracy, processor health, and memory errors continuously
4. Continue exposure to 30 krad total dose (3x expected annual dose)

5. Characterize any performance degradation versus accumulated dose

SEU Testing:

1. Configure DUT in operational mode with TMR enabled and disabled
2. Expose to proton beam while monitoring for upsets
3. Measure SEU cross-section (upsets per unit fluence)
4. Verify TMR effectiveness by comparing error rates
5. Characterize autonomous recovery from detected upsets

Expected Results

Table 6.2: Radiation Test Success Criteria

Parameter	Requirement	Rationale
TID Tolerance	>20 krad functional	2x annual dose with margin
SEU Cross-Section	$<10^{-14}$ cm^2/bit (with TMR)	Acceptable error rate for 12-month mission
TMR Effectiveness	>99% error masking	Demonstrated redundancy benefit
Recovery Time	<10 seconds from SEU	Minimal data loss from upsets
Latchup Immunity	No destructive latchup to 80 MeV-cm^2/mg	Survival of heavy ion environment

Preliminary Analysis

Monte Carlo simulations using SPENVIS orbital radiation models predict annual TID of 5-10 krad at 500 km altitude with 2 mm tantalum shielding [72]. The 30 krad test level provides adequate margin for mission duration with shielding degradation.

6.3 TMR Validation Testing

Test Objective

Verify that Triple Modular Redundancy implementation correctly masks single-point errors and maintains computational integrity.

Test Configuration

Software-based fault injection testing using the FlatSat configuration. Test software deliberately corrupts memory locations and register values to simulate radiation-induced bit flips.

Test Procedure

1. Initialize system with TMR enabled
2. Execute AI inference benchmark establishing baseline accuracy

3. Inject single-bit errors at random locations during execution
4. Verify voter logic correctly identifies and masks errors
5. Inject multi-bit errors to characterize TMR limitations
6. Measure computational overhead from TMR (expected 2.5-3x baseline)

Expected Results

Table 6.3: TMR Validation Criteria

Test Case	Injected Errors	Expected Behavior
Single-bit in one module	1	Masked, no output error
Single-bit in two modules (same location)	2	Detected, output error flagged
Multi-bit in one module	3+	Masked if independent locations
Voter logic corruption	1	Detected by watchdog, system reset

6.4 Power System Testing

Test Objective

Verify power system performance under all operational modes and environmental conditions.

Test Procedure

1. Measure current draw in each operational mode (Safe, Protected, Normal)
2. Verify voltage regulation stability under load transients
3. Test battery charge/discharge cycling
4. Verify undervoltage load shedding operation
5. Measure system efficiency from solar input to loads

Expected Results

Table 6.4: Power System Test Criteria

Mode	Expected Power	Voltage Stability	Duration Tested
Safe Mode	5-8 W	±2%	8 hours
Protected Mode	15-20 W	±3%	4 hours
Normal Mode	30-36 W	±5%	2 hours
Peak Processing	40-45 W	±5%	10 minutes
Eclipse (battery only)	20 W average	±3%	45 minutes

6.5 Communications Testing

Test Objective

Verify end-to-end communication link performance including range, data integrity, and protocol compliance.

Test Configuration

Flight radio connected to calibrated RF test equipment and ground station emulator. Long-range testing uses university rooftop antenna and mobile ground station.

Test Procedure

Bench Testing:

1. Verify transmitter output power (8 W nominal)
2. Measure receiver sensitivity
3. Verify AX.25 protocol compliance
4. Test store-and-forward data handling

Range Testing:

1. Position spacecraft emulator at elevated location (building rooftop)
2. Establish link with mobile ground station at increasing distances
3. Measure signal strength and bit error rate versus distance
4. Verify successful command uplink and telemetry downlink at >50 km

Expected Results

Link budget analysis predicts positive margin at 2000 km slant range (worst-case LEO geometry) with 3 dB margin. Range testing target of 50 km terrestrial validates RF performance with atmospheric losses.

6.6 Complete System Bench Test

Test Objective

Verify integrated system functionality with all subsystems operating simultaneously in flight-like configuration.

Test Configuration

Flight-equivalent hardware in FlatSat configuration with solar array simulator, battery pack, and RF link to ground station emulator.

Test Procedure

1. Execute full mission simulation including:
    - Power-on and initialization sequence
    - Sensor activation and calibration
    - AI inference benchmark execution
    - Mode transition (Normal → Protected → Safe → Normal)
    - Telemetry downlink and command uplink

- Simulated eclipse operations

2. Duration: 24 hours continuous operation
3. Inject simulated radiation events to trigger mode transitions

Expected Results

System completes 24-hour test with:

- Zero unplanned resets
- All mode transitions executed correctly
- AI inference accuracy within 1% of baseline
- All telemetry parameters within limits
- Successful command response for all uplinked commands

---

7. Structural Testing Methods and Results

The structural testing program verifies mechanical integrity under launch and on-orbit environmental conditions. Testing follows NASA GEVS requirements for CubeSat-class spacecraft [69].

7.1 Mass Properties Verification

Test Objective

Verify spacecraft mass, center of mass location, and moments of inertia meet CDS and mission requirements.

Test Equipment

- Calibrated scale (resolution 1 g)
- Mass properties measurement fixture
- Spin table for moment of inertia measurement

Test Procedure

1. Measure total spacecraft mass
2. Measure center of mass in three axes using reaction method
3. Measure moments of inertia about principal axes

Acceptance Criteria

Table 7.1: Mass Properties Requirements

Parameter	Requirement	Measurement Uncertainty
Total Mass	≤14.0 kg	±10 g
CoM X-axis	Within 2.0 cm of geometric center	±2 mm
CoM Y-axis	Within 2.0 cm of geometric center	±2 mm

Parameter	Requirement	Measurement Uncertainty
CoM Z-axis	Within 2.0 cm of geometric center	±2 mm

7.2 Fit Check and Dimensional Verification

Test Objective

Verify spacecraft dimensions comply with CDS Rev. 14 6U envelope and rail interface requirements.

Test Equipment

- CubeSat test dispenser (6U compatible)
- Calibrated calipers and gauge blocks
- Coordinate measuring machine (CMM) if available

Test Procedure

1. Measure overall dimensions at multiple locations
2. Verify rail geometry and surface finish
3. Insert spacecraft into test dispenser
4. Verify smooth insertion and extraction
5. Check deployment switch actuation

Acceptance Criteria

Table 7.2: Dimensional Requirements (CDS Rev. 14)

Parameter	Requirement
Width (X)	100.0 mm ± 0.1 mm
Depth (Y)	226.3 mm max
Height (Z)	340.5 mm max
Rail Width	8.5 mm × 8.5 mm
Rail Surface	Hard anodized, Ra ≤ 1.6 µm

7.3 Vibration Testing

Test Objective

Verify structural integrity under launch vibration environment per NASA GEVS requirements.

Test Equipment

Electrodynamic shaker system with:

- Sine and random vibration capability

- 6U CubeSat test fixture
- Tri-axial accelerometers
- Data acquisition system

Test Configuration

Spacecraft in flight configuration mounted in test fixture simulating dispenser interface. All deployables in stowed/constrained configuration.

Test Procedure

Sine Sweep (Workmanship):

1. Low-level sine sweep (0.25 g, 5-2000 Hz) pre-test
2. Document resonant frequencies
3. Low-level sine sweep post-test
4. Compare resonances (shift >5% indicates damage)

Random Vibration (Qualification):

1. Apply qualification-level random vibration per axis
2. Duration: 2 minutes per axis
3. Level: 14.1 Grms per NASA GEVS

Table 7.3: Random Vibration Spectrum (NASA GEVS)

Frequency (Hz)	ASD (g^2/Hz)
20	0.026
50	0.16
800	0.16
2000	0.026
Overall	14.1 Grms

Acceptance Criteria

- No structural failure or permanent deformation
- Resonant frequency shift <5%
- All components remain secured
- Post-test functional verification successful

7.4 Thermal Vacuum Testing

Test Objective

Verify spacecraft operation across the expected on-orbit temperature range under vacuum conditions.

Test Equipment

- Thermal vacuum chamber
- Thermal control system (heater plates, shrouds)
- Vacuum system (<10^{-5} Torr)
- Temperature sensors and data acquisition

Test Configuration

Spacecraft in flight configuration mounted on thermal interface plate. Electrical interfaces through chamber feedthroughs enable functional testing during thermal exposure.

Test Procedure

Thermal Cycling:

1. Initial functional test at ambient
2. Reduce pressure to <10^{-5} Torr
3. Cold soak at -40°C for 2 hours
4. Functional test at cold extreme
5. Hot soak at +60°C for 2 hours
6. Functional test at hot extreme
7. Complete 4 thermal cycles
8. Return to ambient, functional test

Thermal Balance (Optional): If schedule permits, perform thermal balance test to validate thermal model predictions.

Acceptance Criteria

Table 7.4: Thermal Vacuum Test Criteria

Parameter	Requirement
Cold Survival	-40°C for 2 hours, non-operational
Cold Operational	-20°C with AI processing active
Hot Survival	+60°C for 2 hours, non-operational
Hot Operational	+50°C with AI processing active
Functional Test	All parameters within specification at temperature extremes

7.5 EMI/EMC Testing

Test Objective

Verify electromagnetic compatibility between spacecraft subsystems and with external environment.

Test Procedure

Conducted Emissions:

1. Measure conducted emissions on power lines
2. Verify compliance with MIL-STD-461G CE102

Radiated Emissions:

1. Measure radiated emissions from spacecraft
2. Verify no interference with communications subsystem
3. Verify compliance with MIL-STD-461G RE102

Susceptibility:

1. Verify AI payload operation during RF transmission
2. Verify no upset from external RF fields

7.6 Deployment Testing

Test Objective

Verify reliable deployment of solar arrays and antennas.

Test Procedure

1. Constrain deployables in flight configuration
2. Actuate deployment mechanism
3. Verify complete deployment
4. Repeat for minimum 10 cycles
5. Test at temperature extremes (-20°C and +50°C)

Acceptance Criteria

- 100% deployment success (10/10 cycles minimum)
- Deployment time within specification
- No damage to deployment mechanisms or deployed elements

---

8. Final Design

The final design incorporates the selected alternatives from the conceptual design phase, refined through detailed analysis and preliminary testing. This section provides comprehensive specifications for the flight configuration.

8.1 Mission

The finalized mission for RAD-AI is **Radiation-Aware Autonomous Computing Demonstration**, selected based on the trade study presented in Section 5.1. This mission directly addresses NASA Technology Area 4 requirements for radiation-tolerant autonomy while providing empirical flight data during the critical 2025-2027 period before HPSC availability [73].

Mission Profile

Orbit Parameters:

- Altitude: 400-600 km (CSLI-compatible range)
- Inclination: Any (mission is inclination-agnostic)
- Expected orbital lifetime: 8-15 years (natural decay within 25-year requirement) [74]
- SAA exposure: 6-10 passes per day at 500 km altitude

Mission Duration:

- Operational lifetime: 12 months minimum
- Extended operations: Up to 24 months if spacecraft health permits
- Data collection: Continuous throughout operational phase

Operational Modes

The RAD-AI spacecraft operates in three distinct modes, with autonomous transitions based on radiation environment assessment:

Normal Mode:

- AI inference rate: 10 Hz (star-field tracking)
- TMR: Standard implementation (3 parallel threads)
- Power consumption: 30-36 W
- Data generation: ~50 MB/day
- Activation: Default mode when radiation environment is nominal

Protected Mode:

- AI inference rate: 3 Hz (reduced processing)
- TMR: Enhanced implementation (5 parallel threads with 2 spare)
- Power consumption: 25-30 W
- Data generation: ~30 MB/day
- Activation: Triggered by SAA entry detection or elevated particle flux
- Duration: Maintained until 10 minutes after SAA exit confirmed

Safe Mode:

- AI inference: Suspended
- C&DH: Housekeeping only
- Power consumption: 8-12 W
- Data generation: ~5 MB/day
- Activation: Triggered by multiple SEU detection, anomaly condition, or ground command
- Recovery: Automatic after 30 minutes or by ground command

Table 8.1: Operational Mode Summary

Parameter	Normal	Protected	Safe
AI Processing	10 Hz	3 Hz	Off
TMR Level	Standard	Enhanced	N/A

Parameter	Normal	Protected	Safe
Power	30-36 W	25-30 W	8-12 W
Data Rate	50 MB/day	30 MB/day	5 MB/day
Trigger	Default	SAA/High flux	Anomaly

Autonomous Adaptation Algorithm

The radiation-aware mode transition algorithm implements the following logic:

```
INPUTS:
  - RADFET dose rate (mrad/hr)
  - Particle telescope count rate (counts/sec)
  - GPS-derived orbital position
  - SAA boundary model (AP-8)
  - SEU counter (events in last 60 seconds)

DECISION LOGIC:
  IF (SEU_count > 3 in 60 sec) OR (anomaly_flag):
    TRANSITION → Safe Mode
  ELSE IF (in_SAA OR dose_rate > 50 mrad/hr OR particle_rate > threshold):
    TRANSITION → Protected Mode
  ELSE IF (out_of_SAA AND dose_rate < 20 mrad/hr for 10 min):
    TRANSITION → Normal Mode

HYSTERESIS:
  - Minimum 5 minutes in any mode before transition
  - SAA exit requires 10-minute confirmation
  - Mode transitions logged with timestamp and trigger condition
```

The algorithm includes hysteresis to prevent mode thrashing at boundary conditions. All transitions are logged for post-mission analysis [75].

8.2 Electrical

The electrical design integrates the selected components from the conceptual design phase into a cohesive system architecture. A complete circuit diagram is provided in Appendix B.

Main Computer System

Flight Computer (C&DH):

- Processor: ARM Cortex-M4 (STM32F4 series)
- Clock: 168 MHz
- Memory: 256 KB SRAM, 1 MB Flash
- Storage: 2× 64 GB SD cards (redundant)
- Interfaces: I2C, SPI, UART, GPIO
- Function: Spacecraft housekeeping, communications, mode management

- Power: 0.5 W typical

The flight computer is physically and electrically isolated from the AI payload to prevent fault propagation. Critical spacecraft functions (attitude control, communications, power management) remain operational even if the AI payload experiences radiation-induced failure [76].

AI Payload Processor:

- Processor: SiFive U74 RISC-V quad-core
- Clock: 1.5 GHz
- Memory: 4 GB DDR4 with EDAC
- Storage: 32 GB eMMC
- Operating System: Linux (Yocto-based minimal distribution)
- Function: AI inference, radiation monitoring, autonomous mode control
- Power: 3-5 W typical

AI Accelerator:

- Device: Lattice CrossLink-NX FPGA (LIFCL-40)
- Logic Cells: 39,000
- DSP Blocks: 56
- Memory: 2.5 Mb embedded
- Configuration: Custom neural network inference engine
- Function: Star-field feature extraction, radiation classifier
- Power: 1-3 W typical

Table 8.2: AI Payload Specifications

Component	Model	Key Specs	Power	Mass
RISC-V Processor	SiFive U74	4-core, 1.5 GHz	3-5 W	15 g
FPGA Accelerator	Lattice LIFCL-40	39K LUTs, 56 DSP	1-3 W	10 g
DDR4 Memory	Micron 4 GB	EDAC protected	1-2 W	5 g
eMMC Storage	32 GB	Wear-leveled	0.5 W	3 g
Total AI Payload			**8-15 W**	**~200 g**

Radiation Mitigation Implementation

Triple Modular Redundancy (TMR):

TMR is implemented in software on the RISC-V processor, executing three parallel instances of critical algorithms with majority voting [77]:

```
Architecture:
  Thread 1 ──┐
  Thread 2 ──┼── Voter ── Output
  Thread 3 ──┘
```

```
Implementation:
  - Separate memory regions for each thread
  - Voter executes on dedicated core
  - Disagreement triggers error flag and SEU counter increment
  - Persistent disagreement (>3 consecutive) triggers Safe Mode
```

Computational overhead is approximately 2.8× baseline (less than theoretical 3× due to shared I/O operations).

Error Detection and Correction (EDAC):

DDR4 memory implements SECDED (Single Error Correct, Double Error Detect) using dedicated ECC bits [78]:

- Single-bit errors: Automatically corrected, logged
- Double-bit errors: Detected, flagged, triggers memory scrub
- Scrub rate: Complete memory scan every 60 seconds

Selective Shielding:

Tantalum shielding (density 16.6 g/cm³) provides 10× dose reduction compared to equivalent aluminum mass [79]:

Table 8.3: Shielding Configuration

Component	Shielding	Thickness	Dose Reduction	Mass
RISC-V Processor	Tantalum	2.0 mm	10×	180 g
FPGA	Tantalum	2.0 mm	10×	220 g
DDR4 Memory	Tantalum	1.0 mm	5×	80 g
eMMC Storage	Aluminum (structure)	3.0 mm	2×	Included
Total				**480 g**

Watchdog Timers:

Multiple watchdog timers provide defense-in-depth against processor lockup [80]:

- Software watchdog: 10-second timeout, resets AI application
- Hardware watchdog: 60-second timeout, resets AI processor
- System watchdog: 300-second timeout, power-cycles AI payload

Sensor Package

Table 8.4: Sensor Specifications

Sensor	Model	Measurement	Range	Interface
RADFET (×4)	RFT-300	TID	0-100 krad	Analog/ADC

Sensor	Model	Measurement	Range	Interface
Particle Telescope	Custom	Particle flux	1-10^6 counts/s	Digital
Temperature (×6)	DS18B20	Temperature	-55 to +125°C	1-Wire
Current/Voltage (×4)	INA219	Power monitoring	0-26V, 0-3.2A	I2C
IMU	BMI088	Acceleration, rotation	±24g, ±2000°/s	SPI
Magnetometer	LIS3MDL	Magnetic field	±16 gauss	I2C
Camera (×2)	OV5640	Visible imagery	5 MP, 640×480 video	CSI

Communications System

UHF Transceiver:

- Model: EnduroSat UHF Transceiver II
- Frequency: 435-438 MHz (amateur allocation)
- TX Power: 0.5-8 W selectable
- Data Rate: 1200-9600 bps
- Modulation: GMSK
- Protocol: AX.25
- Power consumption: 8 W (TX), 0.5 W (RX)

Antenna:

- Type: Deployable monopole (×4 for turnstile configuration)
- Gain: 0 dBi (omnidirectional)
- Deployment: Burn-wire release (redundant burn wires per CubeSat 101 guidance) [81]

Ground Station Interface:

- Primary: University of Colorado Colorado Springs UHF station
- Backup: SatNOGS global network (~400 stations) [82]
- Passes per day: 4-6 (primary station), 15-20 (SatNOGS network)
- Daily data volume: 80-120 MB (target)

Power System

Solar Arrays:

- Configuration: 6U deployable panels (2× 3U wings)
- Cell type: GaAs triple-junction (30% efficiency)
- Power generation: 45 W BOL, 38 W EOL (15% degradation assumed)
- Deployment: Burn-wire release, spring-driven deployment

Battery:

- Type: Lithium-ion (18650 cells)
- Configuration: 4S2P (8 cells)

- Capacity: 60 Wh (5.2 Ah at 11.5 V nominal)
- Voltage range: 12.0-16.8 V
- Operating temperature: 0-45°C (charging), -20 to 60°C (discharge)

Power Distribution:

- Bus voltage: Unregulated 12-17 V
- Regulated rails: 5.0 V (5 A max), 3.3 V (3 A max)
- Load switching: Solid-state switches with current limiting
- Fault protection: Overcurrent, overvoltage, undervoltage on all rails

Table 8.5: Power Budget

Subsystem	Nominal (W)	Peak (W)	Duty Cycle	Average (W)
AI Payload	12	30	60%	18.0
C&DH	1.5	2.0	100%	1.5
Communications (TX)	8.0	10.0	10%	0.8
Communications (RX)	0.5	0.5	100%	0.5
ADCS	2.0	4.0	100%	2.0
Thermal (heaters)	0	10.0	20%	2.0
Harness losses	—	—	—	1.0
Subtotal				**25.8**
Margin (20%)				5.2
Total Required				**31.0**
Available (EOL)				**38.0**
Margin				**+7.0 W (18%)**

8.3 Structure

The structural design provides mechanical support and environmental protection for all spacecraft components while maintaining CDS Rev. 14 compliance.

Structural Configuration

Primary Structure:

- Material: 6061-T6 Aluminum
- Configuration: Machined rails with sheet metal panels
- Finish: Hard anodized (MIL-A-8625 Type III) on rails
- Panel attachment: M3 fasteners with Helicoil inserts

Mass Budget:

Table 8.6: Mass Budget

Subsystem	Mass (kg)	Allocation
Structure (primary)	1.8	12.9%
AI Payload	0.4	2.9%
Radiation Shielding	0.5	3.6%
Power System (batteries)	1.2	8.6%
Power System (solar, electronics)	1.5	10.7%
Communications	0.6	4.3%
ADCS	0.8	5.7%
C&DH	0.3	2.1%
Sensors	0.2	1.4%
Harness	0.5	3.6%
Thermal hardware	0.3	2.1%
Subtotal	**8.1**	**57.9%**
Margin (20%)	1.6	11.4%
Total with Margin	**9.7**	**69.3%**
CDS Limit	**14.0**	**100%**
Available Margin	**4.3**	**30.7%**

Component Layout

The 6U volume is organized to optimize mass properties, thermal paths, and accessibility:

-Z Face (Nadir):

- Camera apertures (2×)
- Particle telescope aperture
- Antenna deployment mechanisms

+Z Face (Zenith):

- Solar array hinges
- Sun sensor

±X Faces (Ram/Wake):

- Body-mounted solar cells (supplementary)
- External temperature sensors

±Y Faces (Cross-track):

- Deployable solar arrays (stowed)
- Magnetorquer coils

Internal Volume:

- Layer 1 (bottom): Battery pack, power electronics
- Layer 2: AI payload with shielding enclosure
- Layer 3: C&DH, communications, ADCS electronics
- Layer 4 (top): Antenna deployment, GPS receiver

Thermal Design

Thermal Environment:

At 500 km altitude, the spacecraft experiences [83]:

- Solar flux: 1367 W/m² (direct)
- Albedo: ~30% of solar (Earth-reflected)
- Earth IR: 237 W/m²
- Eclipse duration: ~35 minutes (96-minute orbit)

Thermal Control Approach:

Passive thermal control is implemented using:

- **Surface coatings:**
    - Radiator surfaces: White paint (α=0.2, ε=0.9)
    - Solar array back: Black paint (α=0.95, ε=0.9)
    - MLI blankets: Gold-aluminized Kapton (selective surfaces)
- **Conductive paths:**
    - AI payload thermally coupled to radiator panel
    - Battery isolated from structure with low-conductivity mounts
- **Heaters:**
    - Battery heater: 5 W (thermostat controlled at 5°C)
    - Propulsion heater: N/A (no propulsion system)

Thermal Analysis Results:

Table 8.7: Predicted Temperature Ranges

Component	Cold Case (°C)	Hot Case (°C)	Limit (°C)	Margin
AI Processor	-5	+55	-20/+70	+15/+15
FPGA	-10	+60	-40/+85	+30/+25
Battery	+5	+35	0/+45	+5/+10
C&DH	-15	+50	-40/+85	+25/+35

Component	Cold Case (°C)	Hot Case (°C)	Limit (°C)	Margin
Camera	-20	+45	-30/+70	+10/+25

All components maintain positive margin in both hot and cold cases.

---

9. Concept of Operations

The Concept of Operations (CONOPS) defines the operational timeline and data flow for the RAD-AI mission from launch through end of operations.

CONOPS Overview

Figure 9.1 illustrates the operational flow of the RAD-AI spacecraft during nominal science operations.

CONOPS Diagram:

```
flowchart TD
    subgraph SPACECRAFT["🛰️ RAD-AI SPACECRAFT"]
        A["**STEP 1**
Power On"] --> B["**STEP 2**
Sensor Sample"]
        B --> C["**STEP 3**
AI Inference"]
        C --> D{"**STEP 4**
Mode Decision"}
        D --> E[("**STEP 5**
Data Storage")]
        E --> F["**STEP 6**
Comms Queue"]
        F --> G["**STEP 7**
Downlink to GND"]
        G -.->|"Loop"| B
    end

    subgraph GROUND["📡 GROUND SEGMENT"]
        H["**GROUND STATION**
• Command Processing
• Data Analysis
• Mission Planning"]
    end

    G <-->|"RF Link (UHF)"| H
    H -.->|"Commands"| A

    style SPACECRAFT fill:#e8f4f8,stroke:#0077b6,stroke-width:2px
    style GROUND fill:#f5f5f5,stroke:#555,stroke-width:2px
    style D fill:#ffdd57,stroke:#b8860b
    style E fill:#98c379,stroke:#3d5a3d
```

LOOP: Steps 2-7 repeat continuously during science operations

Operational Steps

Step 1: Power On and Initialization Upon deployment from the launch vehicle dispenser, the spacecraft initializes automatically:

- Deployment switches detect separation
- 30-minute timer delays antenna deployment (per range safety)
- Flight computer boots and performs self-test
- AI payload remains off until commissioning command

Step 2: Sensor Sampling The sensor package continuously acquires environmental data:

- RADFETs sampled at 1 Hz for dose rate calculation
- Particle telescope provides real-time flux measurement
- Housekeeping sensors sampled at 0.1 Hz
- Camera captures images at 0.1 Hz during illumination

Step 3: AI Inference The AI payload processes sensor and image data:

- Star-field images analyzed for attitude determination
- Radiation sensor data processed by classifier neural network
- Inference results passed to mode decision logic
- All processing results logged with timestamps

Step 4: Mode Decision The autonomous mode controller evaluates system state:

- Compares radiation environment to threshold values
- Checks SEU counter and system health flags
- Determines if mode transition is required
- Executes transition with hysteresis enforcement

Step 5: Data Storage Processed data and raw telemetry are stored locally:

- Science data written to primary storage
- Mirror copy written to redundant storage
- Storage utilization monitored (auto-purge of oldest data at 90%)

Step 6: Communications Queue Data is prioritized for downlink:

- Priority 1: Anomaly reports and critical health data
- Priority 2: Mode transition events and triggers
- Priority 3: Science data (AI metrics, radiation correlations)
- Priority 4: Housekeeping telemetry
- Priority 5: Stored imagery

Step 7: Downlink to Ground During ground station passes:

- Beacon provides acquisition signal
- Ground station initiates link establishment
- Queued data transmitted in priority order

- Commands received and acknowledged
- Pass statistics logged

Mission Phases

Phase 1: Launch and Early Operations (Days 1-7)

- Deployment from dispenser
- Antenna and solar array deployment
- Initial ground contact establishment
- Spacecraft health verification

Phase 2: Commissioning (Weeks 2-4)

- Subsystem activation and checkout
- Sensor calibration
- AI payload activation and baseline establishment
- Communications link characterization
- Initial mode transition testing

Phase 3: Characterization (Months 2-4)

- Normal science operations initiated
- SAA passage documentation
- Mitigation strategy validation
- Baseline performance metrics established
- Algorithm tuning based on flight data

Phase 4: Science Operations (Months 5-12)

- Continuous autonomous operation
- Periodic algorithm updates via uplink
- Long-term degradation tracking
- Comparative analysis across radiation zones
- Data publication preparation

Phase 5: Extended Operations (Months 13+, if applicable)

- Continued data collection
- Degradation trend extrapolation
- End-of-life characterization
- Lessons learned documentation

---

10. System Block Definitions Diagram

The system architecture is defined through three complementary block diagrams: external system context, internal subsystem connections, and functional data flow.

10.1 External System Diagram

The external system diagram illustrates the mission environment hierarchy and interfaces external to the spacecraft.

External System Diagram:

```
flowchart TB
    subgraph ENV["🌌 SPACE ENVIRONMENT"]
        E1["☀ Solar radiation (1367 W/m²)"]
        E2["⚡ Trapped radiation (Van Allen belts, SAA)"]
        E3["🌠 Galactic cosmic rays"]
        E4["🌡 Thermal environment (eclipse cycling)"]
    end

    subgraph SC["🛰 RAD-AI SPACECRAFT"]
        direction LR
        S1["AI Payload
System"]
        S2["Spacecraft
Bus"]
        S3["Power
System"]
    end

    subgraph GND["📡 GROUND SEGMENT"]
        direction LR
        G1["Primary
Ground Station"]
        G2["SatNOGS
Network (Backup)"]
        G3["Mission
Operations Center"]
    end

    ENV -->|"Radiation & Thermal Environment"| SC
    SC <-->|"RF Link (UHF)"| GND

    style ENV fill:#1a1a2e,stroke:#4a4a6a,color:#fff
    style SC fill:#e8f4f8,stroke:#0077b6,stroke-width:2px
    style GND fill:#f5f5f5,stroke:#555,stroke-width:2px
```

10.2 Internal System Diagram

The internal system diagram shows electrical and data connections between spacecraft subsystems.

Internal System Diagram:

```
flowchart TB
    subgraph POWER["⚡ POWER BUS (12-17V)"]
        direction LR
        BAT["🔋 Battery Pack
```

```
(60 Wh)"]
        SOL["☀ Solar Arrays
(45 W)"]
        PDU["PDU"]
    end

    subgraph SUBSYSTEMS["📦 SUBSYSTEMS"]
        direction LR
        AI["AI Payload"]
        CDH["C&DH"]
        ADCS["ADCS"]
        COMM["Comms"]
    end

    subgraph AIPAYLOAD["🧠 AI PAYLOAD DETAIL"]
        direction TB
        RISCV["RISC-V + FPGA"]
        SENS["Sensors
(RAD/CAM)"]
    end

    subgraph DATABUS["💾 DATA BUS (I2C/SPI/UART)"]
        direction LR
        BUS[" "]
    end

    SOL --> BAT
    BAT --> PDU
    PDU ==>|"Power"| AI
    PDU ==>|"Power"| CDH
    PDU ==>|"Power"| ADCS
    PDU ==>|"Power"| COMM

    AI --- RISCV
    RISCV --- SENS

    AI <-->|"Data"| BUS
    CDH <-->|"Data"| BUS
    ADCS <-->|"Data"| BUS
    COMM <-->|"Data"| BUS
    SENS <-->|"Data"| BUS

    style POWER fill:#fff3cd,stroke:#ffc107,stroke-width:2px
    style SUBSYSTEMS fill:#e8f4f8,stroke:#0077b6,stroke-width:2px
    style AIPAYLOAD fill:#d4edda,stroke:#28a745,stroke-width:2px
    style DATABUS fill:#f8d7da,stroke:#dc3545,stroke-width:2px
```

Legend: Double lines (==) = Power connections | Single lines (--) = Data connections

10.3 Functional System Diagram

The functional diagram illustrates data flow and control relationships during science operations.

Functional System Diagram:

```
flowchart LR
    subgraph SENSING["📷 SENSING"]
        CAM["Cameras (×2)"]
        RAD["RADFETs (×4)"]
        PT["Particle Telescope"]
    end

    subgraph PROCESSING["🧠 PROCESSING"]
        AI["AI Inference"]
        MODE{"Mode Decision
Logic"}
    end

    subgraph STORAGE["💾 STORAGE"]
        DATA[("Data Storage
(64GB)")]
    end

    subgraph COMMS["📡 COMMUNICATION"]
        TX["Radio TX"]
        RX["Radio RX"]
    end

    subgraph GROUND["🌍 GROUND"]
        GS["Ground Station"]
        MOC["MOC (Ground)"]
        CMD["Command Handler"]
    end

    CAM -->|"Images"| AI
    RAD -->|"Dose Data"| MODE
    PT -->|"Particle Flux"| MODE
    AI --> MODE
    AI --> DATA
    MODE --> DATA
    DATA --> TX
    TX <-->|"RF Downlink"| GS
    GS --> MOC
    MOC -->|"Commands"| CMD
    CMD -->|"RF Uplink"| RX
    RX -->|"Command Flow"| MODE

    style SENSING fill:#e3f2fd,stroke:#1976d2,stroke-width:2px
    style PROCESSING fill:#fff3e0,stroke:#f57c00,stroke-width:2px
    style STORAGE fill:#e8f5e9,stroke:#388e3c,stroke-width:2px
    style COMMS fill:#fce4ec,stroke:#c2185b,stroke-width:2px
    style GROUND fill:#f5f5f5,stroke:#616161,stroke-width:2px
```

Data Flow: Solid arrows (→) | Command Flow: Dashed path from Ground

11. Expected Flight Results

As RAD-AI is a design-phase mission, this section presents expected flight results based on analysis, simulation, and heritage mission data. These predictions establish baseline expectations against which actual flight performance will be evaluated.

11.1 Orbital Environment Predictions

Based on SPENVIS modeling for a 500 km, 51.6° inclination orbit [84]:

Table 11.1: Predicted Radiation Environment

Parameter	Annual Value	Daily Average
Total Ionizing Dose (unshielded)	50-100 krad	140-275 rad
Total Ionizing Dose (2mm Ta shielded)	5-10 krad	14-28 rad
Proton fluence (>10 MeV)	10^{10} p/cm²	2.7×10^{7} p/cm²/day
SAA passes	~2,500/year	6-8/day
SAA residence time	~300 hr/year	~50 min/day

11.2 AI Performance Predictions

Inference Rate:

Based on FPGA implementation benchmarks and RISC-V performance modeling:

- Normal mode: 10.2 ± 0.5 Hz sustained inference rate
- Protected mode: 3.1 ± 0.2 Hz with enhanced TMR
- Processing latency: <100 ms per frame

Accuracy Degradation Model:

Literature data from Phi-Sat-1 and ground testing suggests AI inference accuracy degrades logarithmically with accumulated dose [85]:

```
Accuracy(D) = A₀ − k × ln(1 + D/D₀)

Where:
  A₀ = Initial accuracy (expected 95% for star-field tracking)
  D  = Accumulated dose (krad)
  D₀ = Reference dose (1 krad)
  k  = Degradation coefficient (expected 0.5–2% per decade)
```

Table 11.2: Predicted AI Accuracy Over Mission

Mission Month	Accumulated TID (krad)	Predicted Accuracy

Mission Month	Accumulated TID (krad)	Predicted Accuracy
0 (BOL)	0	95.0%
3	1.5	94.2%
6	3.0	93.5%
9	4.5	92.9%
12 (EOL)	6.0	92.4%

11.3 Mode Transition Predictions

Expected Mode Distribution:

Based on orbital mechanics and SAA geometry:

- Normal mode: 85-90% of mission time
- Protected mode: 8-12% of mission time (SAA passages)
- Safe mode: <2% of mission time (anomalies, testing)

Expected Transition Frequency:

- Normal → Protected: 6-8 transitions/day (SAA entry)
- Protected → Normal: 6-8 transitions/day (SAA exit)
- Any → Safe: <1 transition/week (predicted)

11.4 SEU Rate Predictions

Based on SPENVIS proton flux and literature SEU cross-sections for 28nm CMOS [86]:

Table 11.3: Predicted SEU Rates

Component	Cross-section (cm^2/bit)	Bits	Upsets/day (unmitigated)	Upsets/day (with TMR)
RISC-V registers	10^{-14}	10^4	0.3	<0.01
DDR4 memory	10^{-13}	3×10^{10}	800	8 (with EDAC)
FPGA config	10^{-14}	4×10^7	1.2	0.1 (with scrubbing)

The predicted SEU rates are manageable with implemented mitigation techniques. TMR reduces effective processor upset rate by >95%, while EDAC reduces memory error rate by >99% [87].

11.5 Data Products

Expected Data Volume:

Table 11.4: Daily Data Generation

Data Type	Size/Day	Priority

Data Type	Size/Day	Priority
Housekeeping telemetry	5 MB	Low
Radiation sensor data	15 MB	High
AI performance metrics	10 MB	High
Mode transition logs	1 MB	Critical
Imagery (compressed)	50 MB	Medium
Total	**81 MB**	

Downlink Capacity:

At 9600 bps with 4-6 passes per day averaging 8 minutes:

- Per pass: ~0.5 MB
- Daily capacity: 2-3 MB (primary station only)
- With SatNOGS: 15-20 MB/day

Data prioritization ensures critical mode transition and radiation data are downlinked; imagery stored onboard for selective retrieval.

11.6 Success Criteria Verification

Table 11.5: Success Criteria Assessment

Criterion	Requirement	Prediction	Confidence
Minimum mission (30 days)	Valid telemetry	Achievable	High (>95%)
Baseline mission (6 months)	Continuous operation	Achievable	High (>90%)
Full mission (12 months)	SAA adaptive behavior	Achievable	Medium (>75%)
TID tolerance	>20 krad functional	Expected >30 krad	High
TMR effectiveness	>99% error masking	Expected 99.5%	Medium
Mode transitions	Autonomous execution	Expected 6-8/day	High

12. Conclusion

The RAD-AI mission addresses a validated and urgent need for radiation-tolerant autonomous computing capability in the space industry. As NASA prepares for Artemis lunar surface operations and Mars Sample Return, the ability to perform real-time AI inference in radiation environments becomes increasingly critical. The 6-44 minute communication delays inherent to deep-space missions preclude ground-based decision making for time-critical functions such as hazard avoidance and precision landing [88].

Design Achievements

The RAD-AI design successfully balances performance, radiation tolerance, and cost constraints within the 6U CubeSat form factor:

1. **Novel Radiation-Aware Architecture**: The combination of RISC-V processor, FPGA AI accelerator, TMR software mitigation, selective tantalum shielding, and autonomous mode switching represents a unique approach to radiation-tolerant computing not previously demonstrated in flight [89].

2. **Timely Technology Demonstration**: By targeting the 2025-2027 window before NASA HPSC widespread availability, RAD-AI provides empirical data exactly when early adopters need integration guidance for near-term missions.

3. **Cost-Effective Approach**: The $100,000-$120,000 development cost with free CSLI launch (valued at $250,000) demonstrates that meaningful space AI research is achievable within university and small program budgets.

4. **Strong NASA Alignment**: The mission directly supports NASA Technology Area 4 objectives, Artemis program technology needs, and Mars Sample Return autonomy requirements, maximizing the strategic value of the flight data.

Contribution to the Field

RAD-AI will generate the following contributions to radiation-tolerant space computing:

- **Empirical Performance Data**: First comprehensive dataset on RISC-V processor and FPGA AI accelerator performance degradation in LEO radiation environment over 12-month duration.

- **Mitigation Effectiveness Validation**: Quantitative assessment of TMR, EDAC, and selective shielding effectiveness against actual space radiation, enabling improved design trades for future missions.

- **Autonomous Adaptation Demonstration**: First demonstration of AI-driven radiation detection with autonomous processing mode adjustment, establishing a foundation for deep-space autonomous systems.

- **Design Reference**: Complete design documentation available for adaptation by follow-on university and commercial missions.

Lessons Learned from Design Process

Several insights emerged from the RAD-AI design process:

1. **Heritage Matters**: Building on proven architectures (TRISAT-R for RISC-V, OPTOS for COTS mitigation) significantly reduced technical risk compared to fully novel approaches [90].

2. **Selective Shielding Enables Performance**: The decision to shield only the most sensitive components (processor, FPGA, critical memory) achieves meaningful dose reduction without prohibitive mass penalty.

3. **Software Mitigation is Essential**: Hardware shielding alone is insufficient; TMR, EDAC, and watchdog timers provide defense-in-depth against radiation effects that cannot be completely shielded.

4. **Test Like You Fly**: The comprehensive test program, including proton beam radiation testing, ensures that flight performance matches design predictions.

Recommendations for Future Work

Based on the RAD-AI design experience, the following recommendations are offered for future radiation-tolerant AI missions:

1. **HPSC Integration Studies**: As HPSC processors become available, flight data from RAD-AI should inform integration strategies for hybrid HPSC/COTS architectures.

2. **Higher-Altitude Missions**: Extend radiation-tolerant AI demonstrations to MEO and GEO environments where radiation challenges are more severe.

3. **Deep-Space Demonstrations**: Adapt RAD-AI architecture for lunar orbit or cislunar missions supporting Artemis technology development.

4. **Commercial Applications**: Transition lessons learned to commercial LEO constellation computing requirements (Earth observation, communications).

Final Assessment

RAD-AI represents a technically sound, strategically relevant, and financially feasible approach to advancing radiation-tolerant autonomous computing for space applications. The mission addresses validated NASA technology needs, builds on proven heritage systems, and generates high-value flight data during a critical technology transition period. Upon successful completion, RAD-AI will directly enable the autonomous capabilities required for humanity's expansion to the Moon and Mars.

---

13. Sponsor Interactions

The RAD-AI project, as a graduate-level design study, engaged with potential sponsors and industry partners to validate technical approaches and explore future collaboration opportunities. While the design phase did not involve formal sponsorship agreements, these interactions informed requirements and established relationships for potential implementation.

Industry Engagement

SiFive, Inc.

Initial contact was established with SiFive regarding educational licensing for RISC-V development tools and evaluation hardware. SiFive offers academic programs providing:

- Reduced-cost development boards (~$10,000 value)
- Access to core documentation and design collateral
- Technical support for student projects

SiFive expressed interest in flight demonstrations of RISC-V processors, noting that successful CubeSat missions provide valuable heritage data for commercial space customers. A formal educational license application is planned for the implementation phase [91].

Lattice Semiconductor

Lattice Semiconductor was contacted regarding the CrossLink-NX FPGA selected for AI acceleration. Lattice provides:

- University program with free design software licenses
- Evaluation boards for prototyping
- Application engineering support for edge AI implementations

Lattice confirmed that CrossLink-NX radiation characterization data is limited, making RAD-AI flight data potentially valuable for their space-qualified product roadmap [92].

Cosmic Shielding Corporation

Following their successful 2024 demonstration of radiation-shielded Nvidia GPUs on the Aethero CubeSat, Cosmic Shielding was contacted to discuss potential collaboration [93]. Areas of mutual interest include:

- Comparative analysis of shielding approaches (selective tantalum vs. comprehensive enclosure)
- Data sharing on COTS AI hardware performance in LEO
- Joint publication of radiation effects findings

Cosmic Shielding indicated interest in partnerships that expand the empirical database of COTS electronics performance in space radiation environments.

Blue Canyon Technologies

Blue Canyon Technologies, manufacturer of the XACT CubeSat bus platform, was contacted regarding 6U bus procurement and educational pricing. Blue Canyon offers:

- 10-20% educational discount on bus components
- Technical integration support
- Flight heritage documentation for CSLI proposals

A preliminary budgetary quote was obtained confirming the ~$80,000 bus cost estimate used in project planning [94].

Government and Academic Engagement

NASA Goddard Space Flight Center

Informal discussions with NASA GSFC personnel validated the mission concept alignment with NASA technology needs. Key feedback included:

- Strong interest in COTS AI mitigation data for future missions
- Confirmation that RAD-AI addresses Technology Area 4 gaps
- Encouragement to pursue CSLI as primary launch path
- Offer to review CSLI proposal materials prior to submission

Air Force Research Laboratory (AFRL)

AFRL's Space Vehicles Directorate was contacted regarding potential interest in radiation-tolerant AI research. AFRL confirmed:

- Active programs in COTS radiation mitigation
- University Nanosat Program as potential funding source ($50,000-$100,000)
- Interest in autonomous satellite computing for DoD applications

An application to the University Nanosat Program is under consideration as supplemental funding source [95].

University of Colorado Colorado Springs

Internal university engagement established:

- Faculty advisor: Dr. David Lee (SPCE 5400 instructor)
- Department support: Access to laboratory facilities and test equipment
- College of Engineering: Potential seed funding ($10,000-$30,000)
- Ground station: University UHF station available for mission operations

Sponsor Interaction Summary

Table 13.1: Sponsor Engagement Status

Organization	Type	Status	Potential Contribution
SiFive	Industry	Initial contact	Hardware, licensing (~$10k value)
Lattice Semiconductor	Industry	Initial contact	Software, dev boards (~$5k value)
Cosmic Shielding	Industry	Initial contact	Technical collaboration
Blue Canyon Technologies	Industry	Quote received	Bus procurement (~$80k)
NASA GSFC	Government	Informal discussion	Technical guidance, CSLI support
AFRL	Government	Initial contact	Potential funding ($50-100k)
UCCS	Academic	Active support	Facilities, advising, ground station

Future Sponsor Development

For implementation phase, the following sponsor development activities are planned:

1. **Formal CSLI Proposal**: Submit to NASA CSLI during next solicitation cycle (typically March-April)
2. **Industry Partnerships**: Formalize agreements with SiFive, Lattice for educational programs
3. **Grant Applications**: Submit proposals to NASA STRG, AFRL University Nanosat Program
4. **University Funding**: Request College of Engineering seed funding for prototype development

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14. Team Interactions

The RAD-AI project was conducted as an individual graduate student design study for SPCE 5400 (Small Satellite Engineering & Operations) at the University of Colorado Colorado Springs. While not a multi-person team project, significant collaboration occurred with faculty, industry contacts, and the broader CubeSat community.

Project Structure

Principal Investigator / Designer:

- Jordan Clayton, Graduate Student, Mechanical Engineering
- Responsibilities: All aspects of mission design, analysis, and documentation

Faculty Advisor:

- Dr. David Lee, SPCE 5400 Instructor
- Responsibilities: Technical guidance, design review, grading
- Interaction frequency: Weekly during course, as-needed outside course

Collaboration Model

As an individual project, RAD-AI employed a self-directed learning model with external validation:

Course Framework:

- SPCE 5400 provided structured milestones (outline, draft, final paper)
- Course materials (CubeSat 101, NASA handbooks) established baseline knowledge
- Instructor feedback guided design refinement

External Consultation:

- Industry contacts provided technical validation of component selections
- Online CubeSat community (Reddit r/cubesat, CubeSat Developers forum) answered specific questions
- Published literature and mission reports informed design decisions

Documentation Approach:

- All design decisions documented with rationale and sources
- Trade studies capture alternatives considered and selection criteria
- This approach ensures design knowledge is transferable to future team if project proceeds to implementation

Communication Methods

Table 14.1: Communication Channels

Channel	Purpose	Frequency
Course meetings	Progress review, Q&A	Weekly
Email	Specific technical questions	As needed
Office hours	In-depth design discussions	Bi-weekly
Industry emails	Sponsor development	Monthly
Online forums	Community knowledge	As needed

Lessons Learned: Individual vs. Team Projects

The individual project format offered several advantages and challenges relevant to future CubeSat efforts:

Advantages:

- Complete design ownership enables deep understanding of all subsystems
- No coordination overhead or scheduling conflicts
- Consistent documentation style throughout
- Full accountability for design decisions

Challenges:

- Limited bandwidth for parallel development activities
- No peer review during design process
- Single point of failure for specialized knowledge
- Reduced diversity of perspectives on trade studies

Recommendation for Implementation: If RAD-AI proceeds to hardware development, a 3-5 person team is recommended with the following roles:

- Project Manager / Systems Engineer (1)
- AI Payload Lead (1)
- Spacecraft Bus Lead (1)
- Software/Ground Systems Lead (1)
- Integration & Test Lead (1, can be shared role)

Acknowledgments

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- **Dr. David Lee** (UCCS) for course instruction, technical guidance, and design feedback
- **SPCE 5400 course materials** including NASA CubeSat 101 and mission handbooks
- **Industry contacts** at SiFive, Lattice, Cosmic Shielding, and Blue Canyon Technologies for technical discussions
- **The CubeSat community** for openly sharing mission experiences and lessons learned
- **NASA** for publicly available documentation supporting CubeSat development

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16. Appendices

Appendix A – 3D CAD Drawings and Spacecraft Configuration

This appendix provides detailed drawings and configuration diagrams for the RAD-AI spacecraft structure and component layout.

A.1 Spacecraft External Views

Figure A.1: Isometric View - Stowed Configuration

```
                    +Z (Zenith)
                        ↑
                        │
              ┌─────────┴───────────┐
             /│                    /│
            / │    Solar Arrays   / │
           /  │     (Stowed)     /  │
          ┌───┼─────────────────┘   │
          │   │                │    │
          │   │    ┌────────┐ │     │
          │   │    │ Camera │ │     │
          │   │    │   ×2   │ │     │ +Y
          │   │    └────────┘ │     ↗
          │   │                │    /
          │   └────────────────┼───┘
          │  /                 │  /
          │ /    UHF Antennas  │ /
          │/       (Stowed)    │/
          └────────────────────┘
                        → +X (Ram)

  Dimensions: 100mm × 226.3mm × 340.5mm (6U CDS compliant)
```

Figure A.2: Isometric View - Deployed Configuration

```
                                    Solar Array Wing (+Y)
                                   /
                    +Z            /
                    ↑        ┌────┴────┐
                    │        │█████████│
              ┌─────┴────────┤█ Cells █├──────┐
             /│              │█████████│      │
            / │              └────┬────┘      │
           /  │                   │           │
          ┌───┼───────────────────┼───────────┤
```

```
         │   │                     │           │
         │   │      Spacecraft     │           │
         │   │         Body        │           │
         │   │                     │           │
         │   └─────────────────────┼───────────┘
         │  /                      │          /│
         │ /     \                 │         / │
         │/       \                │        /  │
         └─────────\───────────────┴───────┘   │
                    \  UHF Antenna (×4)        │
                     \   Turnstile            /
                      \                      /
                       \____________________/
                            Solar Array Wing (−Y)
```

A.2 Internal Component Layout

Figure A.3: Exploded View - Internal Layers

```
Layer 4 (Top): Antenna Deployment & GPS
┌──────────────────────────────────┐
│                                 │
│  ┌─────┐ ┌─────┐ ┌─────┐         │
│  │Ant 1│ │Ant 2│ │Ant 3│ ┌───┐   │
│  └─────┘ └─────┘ └─────┘ │GPS│   │
│                          └───┘   │
│                                 │
└──────────────────────────────────┘
              ↓
Layer 3: C&DH, Communications, ADCS
┌──────────────────────────────────┐
│                                 │
│ ┌──────────┐ ┌───────┐ ┌─────┐   │
│ │  C&DH    │ │ UHF   │ │Star │   │
│ │  Board   │ │ Radio │ │Trkr │   │
│ └──────────┘ └───────┘ └─────┘   │
│                                 │
│ ┌────────────────────────────┐   │
│ │   Magnetorquer Assembly    │   │
│ └────────────────────────────┘   │
│                                 │
└──────────────────────────────────┘
              ↓
Layer 2: AI Payload (Shielded)
┌──────────────────────────────────┐
│                                 │
│ ╔════════════════════════════╗   │
│ ║                            ║   │
│ ║  ┌────────┐ ┌──────────┐   ║   │
│ ║  │ RISC-V │ │  FPGA    │   ║   │
│ ║  │  U74   │ │CrossLink │   ║   │
│ ║  └────────┘ └──────────┘   ║   │
│ ║                            ║   │
│ ║  ┌────────┐ ┌──────────┐   ║   │
│ ║  │  DDR4  │ │  eMMC    │   ║   │
│ ║  │  4GB   │ │  32GB    │   ║   │
│ ║  └────────┘ └──────────┘   ║   │
│ ║                            ║   │
│ ╚════════════════════════════╝   │
│   (Tantalum Shielding Box)       │
```

```
 └──────────────────────────────────┘
                    ↓
Layer 1 (Bottom): Power System
 ┌──────────────────────────────────┐
 │                                  │
 │  ┌────────────────────────────┐  │
 │  │    Battery Pack (4S2P)     │  │
 │  │    ┌──┐┌──┐┌──┐┌──┐        │  │
 │  │    │18││65││00││  │×8      │  │
 │  │    └──┘└──┘└──┘└──┘        │  │
 │  └────────────────────────────┘  │
 │                                  │
 │  ┌────────────┐  ┌──────────────┐│
 │  │Power Dist. │  │Solar Array   ││
 │  │   Unit     │  │ Interface    ││
 │  └────────────┘  └──────────────┘│
 │                                  │
 └──────────────────────────────────┘
```

A.3 Shielding Enclosure Detail

Figure A.4: AI Payload Shielding Configuration

```
              Top View (2mm Tantalum)
        ┌──────────────────────────────┐
        │ ╔════════════════════════════╗ │
        │ ║         Lid (2mm Ta)       ║ │
        │ ╠════════════════════════════╣ │
        │ ║  ┌────────┐   ┌──────────┐ ║ │
        │ ║  │RISC-V  │   │   FPGA   │ ║ │
        │ ║  │(2mm)   │   │ (2mm)    │ ║ │
        │ ║  └────────┘   └──────────┘ ║ │
        │ ║                            ║ │
        │ ║  ┌────────┐   ┌──────────┐ ║ │
        │ ║  │Memory  │   │ Storage  │ ║ │
        │ ║  │(1mm)   │   │ (struct) │ ║ │
        │ ║  └────────┘   └──────────┘ ║ │
        │ ╚════════════════════════════╝ │
        └──────────────────────────────┘

       Wall thickness: 2mm Tantalum (16.6 g/cm³)
       Total shielding mass: 480g
       Dose reduction factor: 10× at 500km altitude
```

A.4 Deployable Mechanisms

Figure A.5: Solar Array Deployment Sequence

```
State 1: Stowed            State 2: Released         State 3: Deployed
 ┌───────────────┐          ┌───────────────┐         ┌───────────────┐
 │   ┌───────┐   │          │   ┌───────┐   │         │               │
 │   │███████│   │  Burn    │   │███████│   │ Spring  │   ┌───────┐   │
```

```
│  ││██████████││◄───┼─ Wire ──►│  │██████████│  │───┼─Hinge─►│  ││██████████││  │
│  ││██████████││    │ Release   │  │          │  │   │        │  ││██████████││  │
│  │└──────────┘│    │           │  └────┬─────┘  │   │        │  ││██████████││  │
│              │    │           │       │        │   │        │  └─────┬─────┘   │
│              │    │           │       │        │   │        │        │         │
│  Spacecraft  │    │           │       │        │   │        │        │         │
│    Body      │    │           │       ▼        │   │        │        │  90°    │
└──────────────┘    │           └────────────────┘   │        └────────┴─────────┘

Deployment time: <5 seconds
Redundancy: Dual burn wires per hinge
```

Appendix B – Complete System Circuit

This appendix provides the electrical system architecture and circuit interconnections for the RAD-AI spacecraft.

B.1 Power System Schematic

Figure B.1: Power Distribution Architecture

```
flowchart TB
    subgraph SOLAR["☀ SOLAR ARRAYS"]
        direction LR
        SY["+Y Wing
22.5W"]
        SN["-Y Wing
22.5W"]
    end

    subgraph CHARGE["🔌 CHARGING"]
        MPPT["MPPT
Charger"]
        BAT["🔋 4S2P Li-Ion
60 Wh
12-16.8V"]
    end

    subgraph PDU["⚡ POWER DISTRIBUTION UNIT"]
        direction LR
        OCP["OCP"]
        OVP["OVP"]
    end

    subgraph DCDC["DC-DC CONVERTERS"]
        direction LR
        DC1["DC-DC 5V
(AI Payload)"]
        DC2["DC-DC 5V
(C&DH)"]
        DC3["DC-DC 3.3V
```

```
(Sensors)"]
        DC4["DC-DC 12V
(Heaters)"]
    end

    subgraph LOADS["📦 LOADS"]
        direction LR
        L1["AI Payload
15W max"]
        L2["Flight Computer
1.5W"]
        L3["Sensor Package
0.5W"]
        L4["Thermal Control
10W max"]
    end

    SY --> MPPT
    SN --> MPPT
    MPPT <--> BAT
    BAT -->|"Unregulated Bus (12-17V)"| PDU
    PDU --> DC1 & DC2 & DC3 & DC4
    DC1 --> L1
    DC2 --> L2
    DC3 --> L3
    DC4 --> L4

    style SOLAR fill:#fff3cd,stroke:#ffc107,stroke-width:2px
    style CHARGE fill:#d4edda,stroke:#28a745,stroke-width:2px
    style PDU fill:#cce5ff,stroke:#004085,stroke-width:2px
    style DCDC fill:#e2e3e5,stroke:#383d41,stroke-width:2px
    style LOADS fill:#f8d7da,stroke:#721c24,stroke-width:2px
```

B.2 AI Payload Circuit

Figure B.2: AI Payload Block Diagram

```
flowchart TB
    subgraph POWER["⚡ 5V REGULATED"]
        PWR[" "]
    end

    subgraph COMPUTE["🧠 COMPUTE"]
        direction LR
        RISCV["SiFive U74
RISC-V"]
        FPGA["Lattice CrossLink-NX
FPGA"]
        MEM["DDR4 4GB
+ EDAC"]
    end
```

```
    subgraph PERIPHERALS["📷 PERIPHERALS"]
        direction LR
        WDT["Watchdog
Timer"]
        CAM["Camera
×2 CSI"]
        RAD["RADFET
×4 ADC"]
    end

    subgraph INTERFACE["🔗 DATA INTERFACE TO C&DH"]
        ISO["Isolated - Optocoupler barrier"]
    end

    PWR -->|"Power"| RISCV & FPGA & MEM
    RISCV <-->|"SPI"| FPGA
    FPGA <-->|"I2C"| MEM
    RISCV -->|"UART"| WDT
    FPGA -->|"GPIO"| CAM
    FPGA -->|"ADC"| RAD
    RISCV -->|"UART"| ISO

    style POWER fill:#fff3cd,stroke:#ffc107,stroke-width:2px
    style COMPUTE fill:#d4edda,stroke:#28a745,stroke-width:2px
    style PERIPHERALS fill:#e3f2fd,stroke:#1976d2,stroke-width:2px
    style INTERFACE fill:#f3e5f5,stroke:#7b1fa2,stroke-width:2px
```

B.3 Communications System Circuit

Figure B.3: UHF Communications Block Diagram

```
flowchart LR
    subgraph SPACECRAFT["🛰 SPACECRAFT"]
        CDH["C&DH"]
        subgraph RADIO["📻 UHF TRANSCEIVER (EnduroSat)"]
            TRX["Transceiver"]
            CTRL["Control Logic
• PA Enable
• TX/RX Switch
• PTT Control"]
        end
        ANT["📡 Antenna
Turnstile ×4"]
    end

    SPACE(("🌌 SPACE"))

    CDH <-->|"UART"| TRX
    TRX --> CTRL
    TRX <-->|"RF"| ANT
```

```
    ANT <-->|"UHF Link"| SPACE

    style SPACECRAFT fill:#e8f4f8,stroke:#0077b6,stroke-width:2px
    style RADIO fill:#fff3cd,stroke:#ffc107,stroke-width:2px
    style SPACE fill:#1a1a2e,stroke:#4a4a6a,color:#fff
```

Specifications:

- TX Power: 0.5-8W selectable
- Frequency: 435-438 MHz
- Data Rate: 1200-9600 bps
- Modulation: GMSK
- Protocol: AX.25

Appendix C – Requirements and Verification Matrix

This appendix provides the complete requirements traceability matrix with verification methods and status.

C.1 Communications Requirements Verification

Table C.1: Communications Requirements Verification Matrix

ID	Requirement	Verification Method	Success Criteria	Status
COM-1	Uplink capability ≥1200 bps	Demonstration	Successful command reception	Planned
COM-2	Downlink capability ≥9600 bps	Demonstration	Sustained data transfer	Planned
COM-3	UHF amateur band operation	Inspection	Frequency within 430-440 MHz	Design Complete
COM-4	TX power ≤8 W	Test	Measured output ≤8 W	Planned
COM-5	72-hour data buffering	Demonstration	Data recovery after 72-hr gap	Planned
COM-6	Antenna deployment	Test	Full deployment in 10 cycles	Planned
COM-7	AX.25 protocol compliance	Demonstration	Interoperability with ground	Planned
COM-8	Beacon transmission	Demonstration	60-second interval verified	Planned

C.2 Power Requirements Verification

Table C.2: Power Requirements Verification Matrix

ID	Requirement	Verification Method	Success Criteria	Status
POW-1	Solar generation ≥45 W BOL	Test	Measured output ≥45 W	Planned
POW-2	Energy storage ≥60 Wh	Test	Capacity test ≥60 Wh	Planned
POW-3	5V regulation ±5%	Test	Voltage within 4.75-5.25 V	Planned
POW-4	3.3V regulation ±5%	Test	Voltage within 3.14-3.47 V	Planned
POW-5	Overcharge protection	Demonstration	Charge terminates at 4.2 V/cell	Planned
POW-6	Undervoltage load shed	Demonstration	Loads shed at 3.0 V/cell	Planned
POW-7	Autonomous load shedding	Demonstration	Priority sequence verified	Planned
POW-8	Overcurrent protection	Test	Faults isolated correctly	Planned

C.3 Telemetry and Control Requirements Verification

Table C.3: Telemetry and Control Requirements Verification Matrix

ID	Requirement	Verification Method	Success Criteria	Status
TEL-1	Radiation sensor sampling ≥1 Hz	Demonstration	1-second data intervals	Planned
TEL-2	Housekeeping sampling ≥0.1 Hz	Demonstration	10-second data intervals	Planned
TEL-3	Timestamp accuracy ±1 s	Test	GPS-synchronized time	Planned
TEL-4	7-day onboard storage	Inspection	Storage capacity verified	Design Complete
CON-1	Autonomous mode transitions	Demonstration	All modes exercised	Planned
CON-2	Mode transition ≤60 s	Test	Latency measured <60 s	Planned

ID	Requirement	Verification Method	Success Criteria	Status
CON-3	Watchdog timer reset	Demonstration	Recovery from induced fault	Planned
CON-4	State persistence across reset	Demonstration	No state loss on reset	Planned
CON-5	C&DH isolation from AI	Inspection	Electrical isolation verified	Design Complete

C.4 Mechanical Requirements Verification

Table C.4: Mechanical Requirements Verification Matrix

ID	Requirement	Verification Method	Success Criteria	Status
MEC-1	6U envelope compliance	Inspection	Dimensions within CDS limits	Design Complete
MEC-2	Mass ≤14.0 kg	Test	Measured mass ≤14.0 kg	Planned
MEC-3	CoM within 2.0 cm	Test	CoM location verified	Planned
MEC-4	Quasi-static load 7.5 g	Analysis	Positive margin of safety	Design Complete
MEC-5	Random vibration 14.1 Grms	Test	No damage, <5% freq shift	Planned
MEC-6	Deployable retention	Inspection	Positive restraint verified	Design Complete
MEC-7	Rail surface finish	Inspection	Ra ≤1.6 µm verified	Planned
MEC-8	Shielding mounting	Inspection	Shielding secured	Design Complete
MEC-9	Sensor field of view	Inspection	Unobstructed apertures	Design Complete

C.5 Thermal Requirements Verification

Table C.5: Thermal Requirements Verification Matrix

ID	Requirement	Verification Method	Success Criteria	Status

ID	Requirement	Verification Method	Success Criteria	Status
THE-1	Battery temp 0-45°C	Analysis, Test	Temps within range	Planned
THE-2	AI processor -20 to +70°C	Analysis, Test	Temps within range	Planned
THE-3	FPGA -40 to +85°C	Analysis, Test	Temps within range	Planned
THE-4	Processor ΔT ≤5°C/min	Analysis	Rate within limit	Design Complete
THE-5	30 W dissipation capacity	Analysis	Positive thermal margin	Design Complete
THE-6	10 W heater capacity	Test	Heater output verified	Planned
THE-7	Survival -40 to +60°C	Test	Functional after exposure	Planned
THE-8	TMR thermal accommodation	Analysis	Temps within limits	Design Complete

Appendix D – Management

This appendix provides project management documentation including schedule, budget breakdown, and risk register.

D.1 Detailed Project Schedule

Figure D.1: RAD-AI Project Gantt Chart

```
Task                          | 2025  | 2026           | 2027           |
2028
                              |Q3|Q4|Q1|Q2|Q3|Q4|Q1|Q2|Q3|Q4|Q1|Q2|Q3|Q4|
──────────────────────────────┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┤
PHASE 1: DESIGN               |██|██|██|██|██|██|  |  |  |  |  |  |  |  |
  Concept Development         |██|██|  |  |  |  |  |  |  |  |  |  |  |  |
  Requirements Definition     |  |██|██|  |  |  |  |  |  |  |  |  |  |  |
  Preliminary Design          |  |  |██|██|  |  |  |  |  |  |  |  |  |  |
  ▲ PDR                       |  |  |  |▲ |  |  |  |  |  |  |  |  |  |  |
  Detailed Design             |  |  |  |██|██|██|  |  |  |  |  |  |  |  |
  ▲ CDR                       |  |  |  |  |  |▲ |  |  |  |  |  |  |  |  |
──────────────────────────────┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┼──┤
PHASE 2: BUILD & TEST         |  |  |  |  |  |██|██|██|██|██|██|  |  |  |
  Component Procurement       |  |  |  |  |██|██|██|  |  |  |  |  |  |  |
  ETU Integration             |  |  |  |  |  |██|██|  |  |  |  |  |  |  |
  FlatSat Testing             |  |  |  |  |  |  |██|██|  |  |  |  |  |  |
```

```
    Flight Unit Build          │   │   │   │   │   │   │██ │██ │██ │   │   │   │   │   │
    Environmental Testing      │   │   │   │   │   │   │   │██ │██ │██ │   │   │   │   │
    Radiation Testing          │   │   │   │   │   │   │   │   │██ │   │   │   │   │   │
    ▲ FRR                      │   │   │   │   │   │   │   │   │   │▲  │   │   │   │   │
    Delivery to Integrator     │   │   │   │   │   │   │   │   │   │██ │   │   │   │   │
  ─────────────────────────────┼───┼───┼───┼───┼───┼───┼───┼───┼───┼───┼───┼───┼───┼───┤
  PHASE 3: LAUNCH & OPS        │   │   │   │   │   │   │   │   │   │   │██ │██ │██ │██ │
    Launch Campaign            │   │   │   │   │   │   │   │   │   │   │██ │   │   │   │
    ★ LAUNCH                   │   │   │   │   │   │   │   │   │   │   │★  │   │   │   │
    Commissioning              │   │   │   │   │   │   │   │   │   │   │██ │   │   │   │
    Science Operations         │   │   │   │   │   │   │   │   │   │   │   │██ │██ │██ │
    Data Analysis              │   │   │   │   │   │   │   │   │   │   │   │   │██ │██ │
  ─────────────────────────────┴───┴───┴───┴───┴───┴───┴───┴───┴───┴───┴───┴───┴───┴───┘

  Legend: ██ = Activity duration   ▲ = Review milestone   ★ = Launch
```

D.2 Detailed Budget Breakdown

Table D.1: Detailed Cost Breakdown

Category	Item	Unit Cost	Qty	Total
AI Payload				**$37,500**
	SiFive U74 Dev Kit	$1,500	2	$3,000
	Lattice CrossLink-NX Kit	$800	3	$2,400
	DDR4 Memory Modules	$200	6	$1,200
	eMMC Storage	$100	4	$400
	Camera Modules	$150	6	$900
	RADFETs	$500	8	$4,000
	Particle Telescope Components	$2,000	1	$2,000
	PCB Fabrication	$3,000	3	$9,000
	Tantalum Shielding Material	$5,000	1	$5,000
	Connectors, Passives, Misc	$2,000	1	$2,000
	AI Payload Labor (in-kind)	—	—	$7,600
Spacecraft Bus				**$45,000**
	Bus Platform (COTS)	$40,000	1	$40,000
	Integration Hardware	$3,000	1	$3,000
	Software Licenses	$2,000	1	$2,000
Testing				**$12,500**

Category	Item	Unit Cost	Qty	Total
	Vibration Testing	$3,000	1	$3,000
	Thermal Vacuum Testing	$4,000	1	$4,000
	Radiation Testing (LBNL)	$2,500	2	$5,000
	EMI/EMC Testing	$500	1	$500
Ground Segment				**$6,500**
	Ground Station Upgrades	$3,000	1	$3,000
	Cloud Services (3 years)	$1,500	1	$1,500
	Software Development	$2,000	1	$2,000
Program Support				**$4,500**
	Travel (Reviews, Delivery)	$3,000	1	$3,000
	Documentation, Supplies	$1,500	1	$1,500
Subtotal				**$106,000**
Reserve (20%)				**$21,200**
TOTAL DEVELOPMENT				**$127,200**
Launch Services (CSLI)				**~$250,000**
	(Provided by NASA)			(No cost)

D.3 Risk Register

Table D.2: Project Risk Register

ID	Risk Description	Likelihood	Impact	Risk Score	Mitigation Strategy	Contingency
R1	CSLI selection not achieved	Medium	High	High	Strong NASA alignment; compelling proposal	Commercial launch ($50-100k)
R2	RISC-V radiation performance worse than expected	Medium	High	High	Early radiation testing; conservative design margins	Fallback to ARM processor
R3	FPGA configuration upsets exceed predictions	Medium	Medium	Medium	Configuration scrubbing; TMR in logic	Reduce AI complexity

ID	Risk Description	Likelihood	Impact	Risk Score	Mitigation Strategy	Contingency
R4	Power budget exceeded	Low	High	Medium	18% EOL margin; detailed power analysis	Reduce duty cycle
R5	Thermal design inadequate	Low	Medium	Low	Conservative thermal model; margin in all cases	Add heater capacity
R6	Schedule slip delays launch	Medium	Medium	Medium	Schedule reserve; parallel paths	Accept later launch opportunity
R7	Component obsolescence	Medium	Low	Low	Early procurement; identify alternates	Redesign with available parts
R8	Ground station communication gaps	Low	Medium	Low	SatNOGS backup; onboard data storage	Increase storage capacity
R9	Budget overrun	Medium	Medium	Medium	20% reserve; phased procurement	Reduce scope; seek additional funding
R10	Key personnel unavailable	Low	Medium	Low	Documentation; knowledge transfer	Recruit replacement

Risk Matrix:

```
            │ Low Impact │ Med Impact │ High Impact │
────────────┼────────────┼────────────┼─────────────┤
High        │            │            │             │
Likelihood  │            │            │             │
────────────┼────────────┼────────────┼─────────────┤
Medium      │    R7      │   R6, R9   │    R1, R2   │
Likelihood  │            │            │             │
────────────┼────────────┼────────────┼─────────────┤
Low         │    R10     │   R5, R8   │     R4      │
Likelihood  │            │     R3     │             │
────────────┴────────────┴────────────┴─────────────┘
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D.4 Work Breakdown Structure

Table D.3: Work Breakdown Structure

WBS	Task	Duration	Dependencies
1.0	**Project Management**	42 months	—
1.1	Planning and Control	Continuous	—
1.2	Reviews (PDR, CDR, FRR)	Milestones	2.3, 3.3, 4.4
1.3	Documentation	Continuous	—
2.0	**Systems Engineering**	18 months	—
2.1	Requirements Development	3 months	—
2.2	System Architecture	3 months	2.1
2.3	Interface Definition	2 months	2.2
2.4	Verification Planning	2 months	2.1
3.0	**AI Payload Development**	18 months	—
3.1	Processor Board Design	4 months	2.3
3.2	FPGA Development	6 months	2.3
3.3	Software Development	8 months	3.1, 3.2
3.4	Payload Integration	3 months	3.1, 3.2, 3.3
4.0	**Spacecraft Integration**	12 months	—
4.1	Bus Procurement	6 months	2.3
4.2	Payload-Bus Integration	3 months	3.4, 4.1
4.3	System Testing	4 months	4.2
4.4	Environmental Testing	3 months	4.3
5.0	**Ground Segment**	12 months	—
5.1	Station Upgrades	3 months	2.3
5.2	Software Development	6 months	2.3
5.3	Operational Procedures	3 months	5.2
6.0	**Mission Operations**	12 months	—
6.1	Launch Support	1 month	4.4
6.2	Commissioning	1 month	6.1
6.3	Science Operations	10 months	6.2

End of Document

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