A Heliocentric Satellite Constellation for Continuous Solar Coverage and Space Weather Monitoring

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Allan Shtofenmakher
ashtofen@mit.edu

Daniel Gochenaur
dgochena@mit.edu

Benjamin Waters
bwatr529@mit.edu

Robert Cato III
rcato3rd@alum.mit.edu

Duncan Miller
dmiller1@mit.edu

Luke de Castro
lukedc@mit.edu

Tai Zheng
taizheng@alum.mit.edu

Alexander Koenig
koe@alum.mit.edu

Katelyn Sweeney
ksweeney@g.harvard.edu

Joel Jurado Diaz
jurado@mit.edu

Alexis Lepe
alepe@mit.edu

Claire McLellan-Cassivi
cmclcas@mit.edu

Frederick Ajisafe
ajisafe@mit.edu

Akila Saravanan
akilasar@mit.edu

Joana Nikolova
jonik@mit.edu

Clara Ziran Ma
czm6@mit.edu

Leilani Trautman
trautman@mit.edu

Nadia Khan
nadiak@mit.edu

Olivier L. de Weck
deweck@mit.edu

Edward F. Crawley
crawley@mit.edu

Department of Aeronautics and Astronautics
Massachusetts Institute of Technology
Cambridge, MA

***Abstract*— While the Sun provides the Earth with the energy needed to sustain life, the volatility associated with this intense energy source generates solar weather, which can have devastating implications on Earth. Solar weather can result in data compromise, radio interference, premature satellite deorbit, and even failure of the power grid. To mitigate the negative effects of solar weather, constant observation of the entirety of the Sun's surface is essential. This complete picture of the Sun's ever-changing state will help scientists anticipate solar events that may negatively impact life on Earth. A heliocentric satellite constellation called the Solar Unobstructed Network-based First Long-term Outer-space Weather Effects Research (SUNFLOWER) Observatory is proposed to continuously monitor coronal mass ejections, sunspots, and coronal holes with a suite of science instruments capable of collecting data in various electromagnetic wavelengths. This report offers a holistic view of the mission and spacecraft architectures. The paper begins with a discussion of motivation, mission objectives, and influential past missions. Next, a high-level overview of the mission design flow, mission-level requirements, and cost and schedule estimation assumptions is explored. This is followed by an analysis of the stakeholders and associated value flows and identification of system boundaries. Next, high-level design decisions for critical components of the system architecture and project risks and risk mitigation strategies are discussed. Results for instrument selection, constellation design, and spacecraft design are presented along with the reasoning behind the recommended architectures and design decisions. The final result is an estimate of the overall mission cost and schedule—roughly $4B in FY2025 USD over an 18-year lifecycle beginning in FY2025. The conclusion summarizes the proposed constellation, composed of nine identical spacecraft—each containing a magnetograph, an extreme ultraviolet imager, and a coronagraph—in a Walker-Delta 54.7° configuration at one AU, with three spacecraft in each of three planes. This solution offers continuous 4π-steradian remote sensing coverage of the solar surface—including the poles—with daily communication of science and state-of-health data over Ka-band frequencies to Earth using 34-m ground stations within the Deep Space Network (DSN). To circumvent the significant burden that would be placed on DSN, a compelling and mutually beneficial case for investing in additional 34-m antennas is presented. The paper concludes with recommendations for future work on the SUNFLOWER Observatory.**

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I. INTRODUCTION

A. Motivation

The intense energy radiated by the Sun is a constantly evolving phenomenon which induces ever-changing magnetic fields. These changes cause variances in and expulsions of plasma and radiation that have serious implications for life on Earth. Solar radiation flares can cause radio blackout storms that degrade radio communication at high frequencies. Coronal mass ejections (CMEs), characterized by large ejections of plasma from the Sun's corona, induce geomagnetic storms in Earth's magnetic field. When these CMEs reach the Earth's surface, they can induce surplus current in power grids and cause blackouts [1], [2]. For example, a geomagnetic storm in 2003 caused an hour-long blackout across Scandinavia, as well as the rerouting of many aircraft around the world [3]. Geomagnetic storms can also impact radio signals and degrade GPS performance [1], [2]. These are only a few of the ways the Sun can negatively affect life on—and the environment around—Earth. The National Oceanic and Atmospheric Administration (NOAA) monitors the Sun and predicts space weather events by observing solar flares and

CMEs. This is accomplished via a host of sensors on the Earth's surface and in space [4]. While these observations help to predict and react to solar weather, a lack of data regarding solar and magnetic variability at the Sun's poles and on the far side of the (rotating) Sun curtails heliophysicists' understanding of this celestial body. For example, the solar magnetic field, which shapes and influences phenomena like CMEs, originates from the Sun's poles [5]. Likewise, views of the far side of the Sun allow improved tracking of solar weather indicators like sunspots [6].

B. Objectives

Continuous 4π-steradian (sr) coverage of the Sun's surface through a heliocentric constellation with polar and far-side observations will improve space weather predictions and overall understanding of the Sun. The Solar Unobstructed Network-based First Long-term Outer-space Weather Effects Research (SUNFLOWER) Observatory seeks to improve understanding of the magnetic variability of the Sun as it relates to space weather events and understanding of other stars by continuous observation of the Sun's entire surface, including the poles, using a constellation of heliocentric spacecraft. The overarching scientific objectives of this system are: 1) support global heliophysics research and 2) provide more accurate space weather forecasting and monitoring.

The first objective involves providing 4π sr of coverage of the solar surface, to include the solar poles, generating continuous high-cadence data of the entire solar surface, and providing a better understanding of the Sun's magnetic field. The second objective includes monitoring CMEs and sunspots, collecting high-resolution magnetic data, and producing an overall increased data return rate from past missions.

C. Past Missions

SUNFLOWER is unprecedented in its size as a heliocentric constellation. However, multiple standalone missions provide inspiration and valuable lessons to the proposed constellation.

Between 1994 and 1995, the Ulysses spacecraft, developed by the European Space Agency (ESA) in conjunction with the National Aeronautics and Space Administration (NASA), became the first spacecraft to observe the Sun's poles. The Ulysses spacecraft used a gravity assist around Jupiter to maneuver out of the ecliptic plane and reach over 80° latitude relative to the Sun. These polar passes only spanned a few months each, with years of relative idleness between [7]. SUNFLOWER improves upon Ulysses by providing continuous polar observations.

A more recent mission to pursue polar observations of the Sun is the ESA Solar Orbiter, which launched in 2020 and will eventually reach 33° inclination above the ecliptic plane [8], [9]. This spacecraft uses multiple gravity assists from Venus to reach its desired inclination over the course of several years [10]. Like Ulysses, with only one spacecraft, these observations are not continuous. Additionally, Solar Orbiter spends a very limited amount of time in view of the Sun's highest latitudes. SUNFLOWER improves upon this by observing all latitudes of the Sun continuously.

Another inspirational mission is NASA's Solar Dynamics Observatory (SDO). The launch of SDO represented another large step toward understanding the structure of the Sun's magnetic field [11]. However, since its launch in 2010, its instruments continue to undergo severe degradation. For example, the Extreme Ultraviolet Variability Experiment (EVE) has already experienced charge-coupled device degradation of 90% at the 105 nm wavelength [12]. This loss of performance contributes to the motivation for a constellation of spacecraft with newer instruments that degrade at slower rates than past missions, which is discussed in detail in Section IV.

Combining these points of improvement and novelty, SUNFLOWER seeks to improve upon past missions by continuously covering the poles—in contrast to time-limited flybys—which is accomplished using a multi-plane heliocentric constellation with 4π-sr coverage. It also seeks to account for instrument degradation in missions like SDO by using newer instruments that are inherently resilient to degradation. In addition, SUNFLOWER will utilize alternative trajectory methods to interplanetary flybys—namely, advanced solar electric propulsion (SEP, discussed further in Section IV).

The proposed concept of operations (CONOPS) for the SUNFLOWER Observatory, beginning with initial launch to low Earth orbit (LEO), is presented at a high level in Figure 1 and is also discussed further in Section IV.

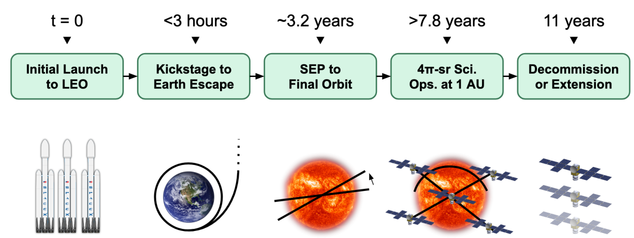


Figure 1. **SUNFLOWER CONOPS.**

II. METHODOLOGY

The respective methodologies for the systems design process, requirements derivation, and costing and scheduling process on this program follow.

A. Systems Process

To maximize the value of SUNFLOWER, the Systems Engineering (SE) team began the design of this mission with a detailed stakeholder analysis. Once the stakeholders were identified, key science objectives were determined to meet their needs. From this, a list of solar properties of interest, ideal spacecraft distance from the Sun and necessary technology payloads to collect the science data of interest were identified. By comparing metrics such as complexity, cost, solar coverage, and science value, a constellation architecture was down-selected and used to inform spacecraft trajectories, core constellation design, and types of propulsion. Given these specifications, a spacecraft bus design could be created with thermal, power, communications, attitude determination and control system (ADCS), material, and structural trade studies. Finally, given the spacecraft mass, the ideal launch vehicle was selected from a trade study including vehicle cost, risk, and capability. Figure 2 shows the methodology followed by the team.

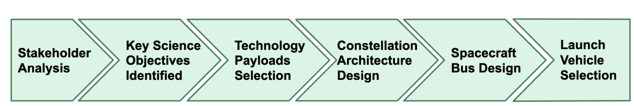


Figure 2. **Systems process flowchart.**

B. Requirements Derivation

The process for deriving high-level requirements for the SUNFLOWER program was largely driven by inputs provided by key stakeholders. In particular, many are derived from the overarching SUNFLOWER description and mission objectives, such as 4π-sr coverage of the Sun, space weather monitoring, and continuous communication of science data back to Earth. As an example, the requirement for mission lifetime was defined to align with the average duration of a solar cycle—roughly eleven years [13]. Other factors affecting the derivation of the requirements include cost, schedule, and practical considerations—such as compatibility with the NASA Deep Space Network (DSN)—to enable the mission as a whole.

Table 15 in Appendix A presents the resulting requirements matrix, which includes eleven requirements in all.

C. Costing and Scheduling Process

The overall program schedule, with the exception of instrument development, is primarily derived from the schedules of similar missions, such as Ulysses and Parker Solar Probe [14], [15]. Each payload instrument selected for use in the SUNFLOWER program has previously flown [16], is set to fly in the near future [17], or is space qualified and ready to launch [18]. This extensive heritage enables an accelerated timeline for instrument development relative to other heliophysics programs. This, in turn, enables the operating assumption that the payload development timeline does not drive the overall program schedule and can largely be enclosed within the timeframe for spacecraft/bus development.

The preliminary cost estimate for the SUNFLOWER program was derived using information from several sources [14], [19], [20], [21], [22], [23], [24].

In particular, cost estimating relationships (CERs) from [19], often called "SMAD" or "SME," were used to translate key metrics—such as subsystem-level dry mass estimates, science instrument power draws, and mission lifetime——into cost estimates for the program's work breakdown structure (WBS). The uncertainty surrounding these input parameters was used to derive low, medium, and high cost estimates for each WBS line item. Since the cost models in [19] assume FY2010 USD, the results were adjusted for inflation.

The high-level costing models in SME combine launch and orbital operation costs into one estimate [19]. However, for SUNFLOWER, the Constellation Design and Astrodynamics (CDA) team generated reasonable launch cost estimates as part of their launch vehicle trade study, and these figures were selected for the costing model due to their higher fidelity [20], [21]. To complete the set, mission operation and data analysis costs were determined using NASA's Mission Operations Costing Tool (MOCET) for a large Near-Earth Discovery Heliophysics program operating under the SUNFLOWER timeline [22].

A qualification or "qual" unit is also included in the cost estimate. Although not strictly necessary, a qual unit enables engineers to verify that the spacecraft will survive the launch vehicle and space environments while there is still time to adjust the final spacecraft design. For a program that is expected to fly multiple spacecraft far beyond Earth, the reduction in risk associated with the qualification unit is likely well worth the added cost. Further discussion of cost and schedule is presented in Section IV.

III. Analysis

Analyses are presented for the stakeholder investigation, the system boundary definition, and a high-level overview of instrument selection, constellation design, satellite bus architecture, and risks.

A. Stakeholder Investigation and Value Network

As a NASA project, this system is at the center of a complex stakeholder network that includes the U.S. government, NOAA, and the heliophysics community. Figure 3 shows the Stakeholder Value Network created from the SUNFLOWER observatory and its stakeholders. The most important value flows—in red—are critical to the program's survival. Yellow flows reflect values affecting the project performance, and green flows reflect excitements that are desirable but not strictly necessary for project success.

As a civil space agency, NASA's relationship and value exchanges with the government (i.e., the Executive Branch and Congress) are particularly important for the project's success. Specifically, Congress is the source of program policy, approval, and funding, all of which are vital for the system to exist. Other necessary value flows include materials that come from manufacturers in the commercial space industry and scientific results that are provided by the heliophysics program scientists using SUNFLOWER Observatory data.

Prioritizing stakeholders by value flows indicates that the U.S. government, commercial space industry, and the heliophysics community—in particular, the NASA Heliophysics Science Mission Directorate—are the most important for the SUNFLOWER program. The heliophysics community is considered essential for having shaped SUNFLOWER's scientific objectives. Specifically, discussions with stakeholders revealed that the core science needs driving the goals of this project are novel solar observation points outside of the Earth-Sun line and a better understanding of the solar magnetic field. Indeed, these two needs are connected, as continuous full-surface observations—especially of the Sun's poles—are required to determine the magnetic flux boundaries and the magnetosphere's role in shaping solar weather. Accordingly, SUNFLOWER will seek to offer the science community never-before-collected continuous full-surface magnetograms, extreme ultraviolet (EUV) imaging, and corona imaging, which will provide critical insights into the behavior and boundaries of the Sun's magnetic field. Full-surface magnetograms inclusive of the poles will allow the science community to determine the exact flux boundaries of the Sun's magnetic field, which will substantially increase heliophysicists' understanding of the creation of solar weather, as well as the behavior of the various layers of the Sun. EUV and coronal imaging will complement these magnetograms by providing detailed images of solar flares and sunspots.

SUNFLOWER's emphasis on space weather reveals notable value flows among itself, NOAA, the energy industry, and the U.S. government. A value loop is created when observations from SUNFLOWER go to NOAA, which uses them to make space weather predictions for the energy industry. The energy industry uses these predictions, along with the government's space weather mitigation policy, to provide reliable electricity to the general public. In turn, the public provides additional support for the SUNFLOWER program to Congress, encouraging potential program extension after the primary mission. This connection with the energy industry is a unique characteristic that sets the SUNFLOWER stakeholder landscape apart from that of most other space programs.

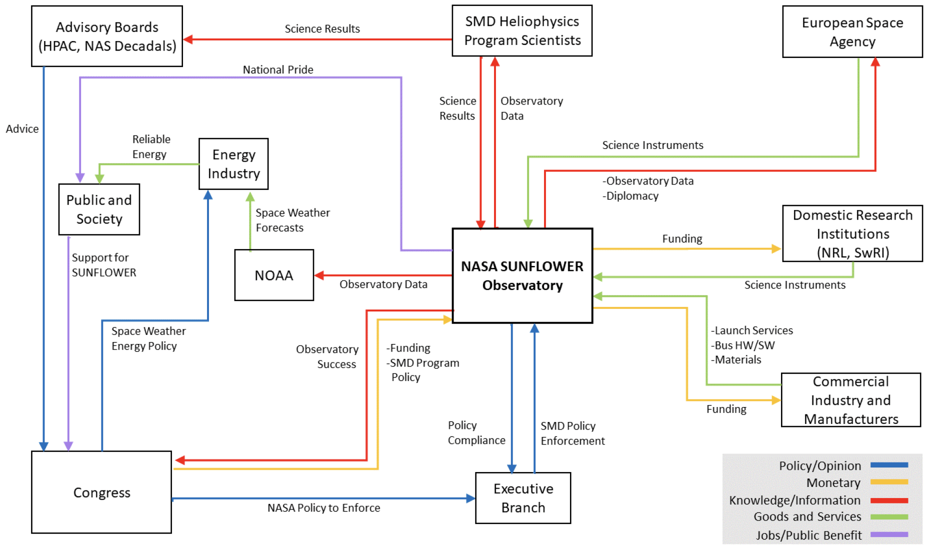


Figure 3. Stakeholder Value Network.

B. System Boundaries

Figure 4 defines the elements of the SUNFLOWER system and its context in a largely solution-neutral manner. The overall system consists of an arbitrary number (N) of spacecraft (SC) contained within an arbitrary number (Z) of constellation planes, which gather energy and collect science data from the Sun. Each SC interacts with its local space environment by exchanging heat, encountering charged particles, enduring radiation, and more. Likewise, each SC communicates with ground stations on Earth to transmit science and state-of-health data and to receive commands from operators.

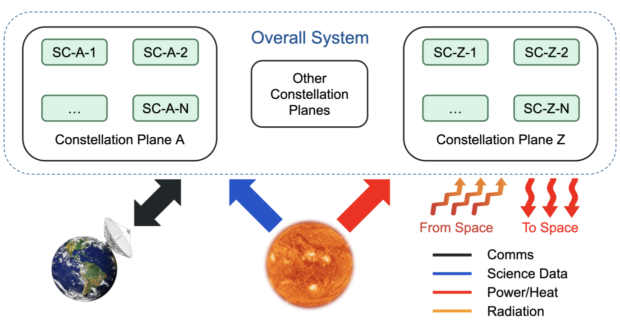


Figure 4. System boundary definition for the SUNFLOWER Observatory program.

C. High-Level Design: Instrument Selection

The preliminary scientific instrument package for SUNFLOWER was selected as a result of discussions between the Heliophysics and Solar Monitoring (HSM) team and various stakeholders that helped match overall scientific objectives to possible scientific payloads.

Starting from the core scientific objectives, the following goals were determined through stakeholder discussions: 1) determine the solar magnetic flux boundaries, and 2) determine the primary drivers of solar wind. Conversations were held with stakeholders in the heliophysics community to determine the observable solar phenomena needed to address scientific objectives. Observing CMEs, sunspots, and coronal holes continuously over the entire surface would fulfill these overarching goals. Specifically, the first goal can be fulfilled through observations of the magnetic flux around coronal holes (e.g., via omni-surface magnetograms), and the second goal can be fulfilled by measuring the magnetic flux, the structure of the corona, and the locations of sunspots (e.g., via full-disk photospheric magnetograms, continuous EUV imaging, and coronagraphic imaging).

Discussions with stakeholders determined the types of instruments that would be used to observe these phenomena. Instrument selection is discussed in detail in Section IV.

D. High-Level Design: Constellation & Astrodynamics

Balancing heliophysics science goals, spacecraft quantity, and bus propulsion needs drove the CDA team's design process for the constellation architecture. The architecture must consider a feasible bus design given current launcher and in-space propulsion technologies. The relevant trade spaces include the quantity and placement of final constellation orbits along with their corresponding deployment trajectories.

The SUNFLOWER Observatory prioritizes remote sensing over in-situ heliophysics measurements. Therefore, the final orbit trade space was narrowed to circular orbits at 1 astronomical unit (AU) to achieve continuous 4π-sr coverage while minimizing cost and fuel mass. Adjusting either the eccentricity or semimajor axis would raise Δv and require more stringent thermal and radiation considerations without substantially improving coverage or science value.

With the orbit shape selected, a variety of orbital plane inclinations and number of satellites per plane were analyzed. Each proposed constellation design was evaluated for the scientific value it would deliver, with specific focus on the worst-case viewing angles offered of each point on the Sun's surface. Section IV provides a detailed discussion on the selected orbital plane configuration.

The technical cost of each final orbit was determined from the Δv and time of flight needed to fly the full trajectory from Earth. The trade study included chemical propulsion, electric propulsion, and solar sails, all considered both with and without gravity assists.

From investigations into historic heliophysics missions and the degradation of their instruments over time, combined with recent advancements in radiation shielding technology, the reasonable expected lifetime of each spacecraft was estimated to be approximately 11 years. To maximize the time each spacecraft spent in its final orbital configuration, solar sails were eliminated from the study due to their high time of flight and relatively low Technology Readiness Level (TRL).

Likewise, gravity assists also required multi-year times of flight. A trajectory utilizing a Jupiter gravity assist (JGA) would permit a ballistic inclination increase with no additional Δv. Similar mission concepts have explored a JGA, including the Solar Polar Observing Constellation (SPOC) [25]. The analysis in [25] agrees with the CDA team's JGA investigation that the total time of flight would exceed six years. Given the additional complexity of deep space operations around Jupiter, the additional delay before useful science operations, and the availability of a feasible direct SEP injection, electric propulsion emerged as the selection from this trade study. The JGA remains as a backup trajectory in case of a substantial reduction to the baseline Δv budget.

After the Conceptual Design Review (CoDR), the CDA team focused on higher-fidelity modeling of the proposed Walker-Delta 54.7° constellation. This included investigations into launch vehicle capabilities, low-thrust trajectory optimization, and sensitivities to launch date and trajectory error. The results of these analyses are discussed in Section IV.

Trajectory design using low-thrust propulsion does not lend itself to optimal analytical solutions. The CDA team developed feasible hand-designed inclination and phasing maneuvers between 0.8 AU and 1.1 AU with the help of simulations from NASA's open-source General Mission Analysis Tool (GMAT) [26]. The results from the GMAT analysis are treated as baseline for all subsequent analyses. In addition, the team developed a preliminary nonlinearly optimized trajectory with `pykep`, a Python package for low-thrust optimization [27], which yielded improvements in both Δv and time of flight. Further trajectory optimization is recommended as future work.

E. High-Level Design: Spacecraft Bus

The goal of the SUNFLOWER bus team is to provide a resilient architecture to support the scientific objectives of the SUNFLOWER Observatory mission. Critical subsystems include command and data handling (C&DH), attitude determination and control, communications, electrical power, thermal management, and structures and mechanisms. The spacecraft bus architecture trade space—including radioisotope thermoelectric generators (RTGs) for power—is shown in Table 1. From these options, the spacecraft is designed to satisfy constraints and requirements determined by the SE, HSM, and CDA teams. These are shown in Table 2.

The communications system design depended on available infrastructure and technologies (both radio frequency and optical) as well as the need to manage blackout periods. NASA's DSN emerged as a viable option due to its proven heritage with deep space communications. The key variables in the trade space regarding radio frequency for data transmission were gain, antenna diameter, data rates, mass, and heritage.

The mechanical design of the spacecraft involved researching architectures from past missions. To minimize spacecraft cost and fuel consumption, the SUNFLOWER Observatory spacecraft architecture emphasized mass and volume minimization compared to previous heliophysics architectures. The core spacecraft bus design variables were relative launch loads, volume, mass, relative cost, and proven heritage.

The attitude determination system was compared across precision, power required, mass, cost, and proven heritage. The attitude control system was compared across torque performance, lifetime, power required, mass, cost, and proven heritage. The power system was compared across power generation efficiency, heat dissipation effectiveness, radiation degradation, mass, cost, and proven heritage.

The thermal systems were designed with constraints and requirements defined by the other subsystems within the SUNFLOWER vehicle. A trade space was analyzed for the thermal control of the spacecraft for the extreme hot case of 0.8 AU from the Sun and the extreme cold case of 1.1 AU from the Sun. The limited cooling capacity of passive radiators indicated that actively cooled systems would be necessary given the environment in which the SUNFLOWER Observatory mission will operate. The spacecraft will use a combination of passive radiators and pumped fluid loops to maintain the vehicle temperature. Pumped fluid loops have proven flight heritage and have flown on other, similar solar missions in the past [28]. For specific subcomponents that may require more granular thermal control, cryocoolers may be implemented on an as-needed basis.

Finally, the electrical power system (EPS) is the interstitial tissue that binds the other subsystems of the bus and allows them to function. Using the power input available at the limiting orbital distance of 1.1 AU and available mass and volume constraints, the SUNFLOWER EPS team calculated a necessary solar panel area of 25.4 m^2 to support 8.5 kW of power consumption during propulsive maneuvers and 0.6 kW when collecting heliophysics data. A summary of the power budget is shown in Table 3.

F. Risks

Sixteen risks were identified across four subteams: SE, HSM, bus design, and CDA. Table 4 enumerates these sixteen risks in the NASA standard risk matrix, as they were presented at the Preliminary Design Review (PDR) [29].

During the PDR phase, between one and five concurrent mitigation strategies were identified for each risk, with special attention given to the three highest risks presented at CoDR: 2.02 Payload Radiation Management, 3.01 Bus Radiation Shielding, and 3.06 Mass Budget Closure. Risks 2.02 and 3.01 were both reduced through further analysis leading to a more refined understanding of radiation effects on the payload (2.02) and a more radiation-robust design (3.01). An improved understanding of the relationship between launch vehicle payload mass and constellation deployment time released the dry mass constraint on the spacecraft bus, leading to the retirement of risk 3.06 at PDR. Risk 1.01 was discarded for being redundant with two others (2.02 and 3.01).

Risk statements are available in Appendix B, Table 16. The mitigation strategies that led to risk score changes are enumerated in Appendix B, Table 17, where the bolded mitigation strategies represent the actions that were taken en route to PDR to reduce risk scores.

IV. Results

This section presents design recommendations and analysis results for the scientific instruments, constellation, and bus, as well as a notional CONOPS and cost and schedule estimates.

Table 1. SUNFLOWER Bus Trade Space

Component	Option 1	Option 2	Option 3
Attitude Determination	Star Tracker	Sun Sensor	Gyroscopes
Attitude Control	Reaction Wheels	Thrusters	Sun Radiation Pressure Control
Power System	Identical Solar Arrays	Primary + Secondary Arrays	RTGs
Communication Band	S-band	X-band	Ka-band
Structure	Rack & Rail	"Box"	Unibody
Size	Small/Microsat	Medium Class	Large Class
Thermal Control	Cryocoolers	Fluid Loops	Radiators

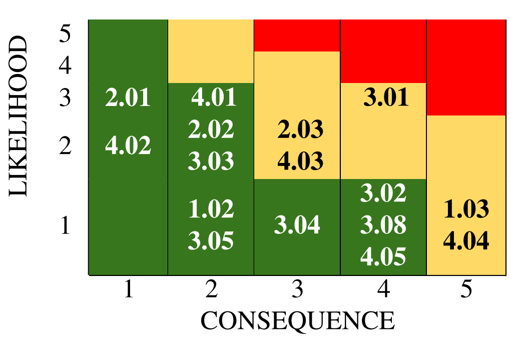
Table 2. SUNFLOWER Bus Design Constraints

Characteristic	Value
Mission Lifetime	11 years
Orbital Environment	0.8–1.1 AU
Communication Data Rate	500 MB/day min.
Onboard Data Storage	4 GB min.
Propulsion Power	7231 W
Instrument Power	21.5 W
Operational Temp. Constraint	300 K
S/C Pointing Accuracy	0.16°
S/C Pointing Knowledge	0.08°
Maximum Wet Mass	1000 kg

Table 3. SUNFLOWER Power Budget

Subsystem	Thrust Mode (W)	Sci. Ops. (W)
Payload	0	22
Propulsion	7231	0
Comms	328	328
Other Bus	194	194
Margin (10%)	775	55
Mode Total	**8528**	**599**

Table 4. NASA Risk Matrix at SUNFLOWER PDR



A. Recommended Architecture: Instrument Selection

The spacecraft payload bay will consist of three instruments that were selected to fulfill the overall science objectives of supporting global heliophysics research and facilitating more accurate space weather predictions. The Science Traceability Matrix (STM) in Appendix C offers a full mapping of scientific goals to chosen instruments. Some key properties of the preliminary instruments selected for SUNFLOWER—the white light Compact Coronagraph (CCOR-1) [30], [31], the Compact Doppler Magnetograph (CDM) [18], and the Sun Watcher using Active Pixel System Detector and Image Processing (SWAP) EUV imager [32], [33]—are shown in Table 5. Included in this table are each instrument's inner field of view (FOV) and outer FOV, both measured in arcminutes ('), as well as signal-to-noise ratio (SNR). In the case of SWAP, the SNR ranges from 18.3 (while observing a quiet Sun) to 183 (while observing an active region) [33]. The SNR for CCOR-1 could not be determined from limited publicly available information but is expected to meet mission needs.

Table 5. Final Instruments and Specifications

Instrument	Mass	Power	FOV	SNR
CCOR-1	21 kg	12.2 W	59'–299'	- - -
CDM	16 kg	6.7 W	0'–40'	4000
SWAP	11 kg	2.6 W	0'–54'	18.3–183

The CDM will collect full-surface magnetograms to observe CMEs, sunspots, and coronal holes. SWAP and CCOR-1 will provide overlapping solar disk images in different wavelengths (visible and ultraviolet) that will track the creation and journey of CMEs, solar wind, and the impacts of coronal holes on solar wind. Collectively, these payloads will observe CMEs, sunspots, and coronal holes continuously across the entire surface of the Sun, including the poles. These observations, in turn, will fulfill the specific scientific objectives of determining the Sun's magnetic flux boundaries and determining the primary drivers of solar wind.

Stakeholder conversations indicated that miniaturizing the instrument payloads would enable a lightweight spacecraft design without compromise to mission objectives. The final payload selections emphasized low-mass configurations of similar payload technologies. Potential risks due to low TRLs for these selections were mitigated on an instrument-specific basis. For example, the final chosen (miniaturized) magnetograph instrument is developed by the Southwest Research Institute and is at TRL 6 at time of writing [18]. The chosen white light coronagraph is one developed by the Naval Research Laboratory and is currently expected to be used on the NOAA GOES-U mission in 2024 [17]. The chosen EUV imager already has flight history on ESA's technology demonstration mission PROBA-2 [16].

Note that each of these instruments is remote sensing in nature, not in situ. This decision is a result of stakeholder conversations between the HSM team and heliophysics community that recommended prioritizing novel vantage points of remote sensing as a means of propelling the heliophysics field forward. As such, the mission is designed to prioritize continuous polar coverage via remote sensing. Moreover, these same stakeholder conversations indicated that a more direct zenith viewing angle, defined as ζ in Figure 5, of the solar surface—especially the poles—would generate higher-quality remote sensing data. Requirement 4.02, Solar Pole Viewing Angle, was created to capture this need and passed to the CDA team for constellation design consideration.

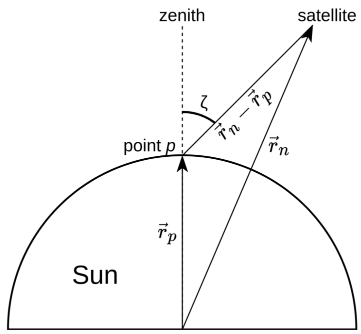


Figure 5. Definition of the zenith viewing angle ζ.

For each instrument, there will be two stages of calibration. Pre-mission calibration, which will occur on the ground in controlled laboratory environments, will be the first stage. The other stage of calibration will be performed semi-regularly during mission operations. These standard calibrations are performed for astronomy data—bias, dark, and flat corrections. Bias corrections (included in dark and flat corrections) account for image sensor noise, dark corrections account for thermal effects on the instrument, and flat corrections account for dust and vignetting. The focus will also be calibrated semi-regularly for each instrument to ensure that data remains high quality. Additionally, the CDM will require calibration to remove any small magnetic bias from the instrument [18]. All of these calibrations can be performed on the instrument, as the requisite operations, such as subtraction and division, are minimally intensive.

The limiting factor on cadence for the SUNFLOWER mission is the amount of data to be downlinked each day. Due to the large distance from Earth to many of the satellites, the data downlink budget will be smaller than the data budget for other heliophysics missions that are closer to the DSN. Additionally, due to the configuration of the satellites in the constellation and the layout of the DSN on Earth, there will be satellites below the ecliptic that will have less DSN contact time and a correspondingly lower amount of data downlinked per day. The maximum science data downlink per day in the worst-case scenario is 435.5 MB per day in the case of satellites below the ecliptic at a distance of 2 AU from Earth. For satellites at and above the ecliptic, this value rises to be 871.0 MB per day.

Each instrument on the satellite produces data files that will be in the Flexible Image Transport System (FITS) image format. The file sizes per instrument were determined from actual instrument data (like SWAP [34]), data from similar instruments (like CCOR-1 [35]), or from image dimensions (like CDM [18]). Additionally, on-board compression is needed, as specified by NASA's parameters on FITS compression [36]. The total and compressed file sizes are presented in Table 6.

Table 6. Instrument Data File Sizes

Instrument	File Size	Compressed Size
CCOR-1	2 MB	0.50 MB
CDM	7 MB	1.75 MB
SWAP	2 MB	0.50 MB

Using these file sizes, the cadence for each instrument was determined for satellites both above and below the ecliptic. The optimal cadences for below the ecliptic can be found in Table 7 and those at or above the ecliptic in Table 8.

Table 7. Optimal Instrument Cadences Below Ecliptic

Instrument	Cadence	MB/Day	Images/Day
CCOR-1	15.0 min	48	96
CDM	7.0 min	360	205
SWAP	30.0 min	24	48

Table 8. Optimal Instrument Cadence At/Above Ecliptic

Instrument	Cadence	MB/Day	Images/Day
CCOR-1	7.5 min	96	192
CDM	3.5 min	720	410
SWAP	15.0 min	48	96

After compression, the images will be sent to Earth, where calibration, post-processing, and data analysis will occur.

As mentioned in Section I, instruments with low annual sensor degradation rates were selected to meet mission lifetime requirements. Specifically, the imaging sensors of CCOR-1 and SWAP showcase longevity in imaging sensor degradation, degrading at only 0.45% and 0.35% total pixel failure rates every year, respectively [12], [37]. Figure 6 shows the expected degradation over the expected initial lifetime of 11 years of these imagers, which lose less than 5% of all pixels after one solar cycle [13]. For comparison, one of SDO's ultraviolet imaging sensors degraded over 90% at the 105 nm wavelength after just a few years [12].

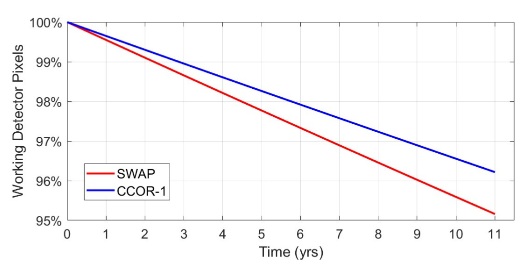


Figure 6. SUNFLOWER payload sensor degradation.

B. Recommended Architecture: Constellation Astrodynamics

A trade study for the constellation design was performed which resulted in the constellation shown in Figure 7, a classic 3-plane Walker-Delta 54.7° constellation [38]. In addition to a 120° difference in the longitude of the ascending nodes, each plane employs a ±80° true anomaly offset relative to the rear and forward planes, respectively, to optimize zenith viewing angles across the solar surface.

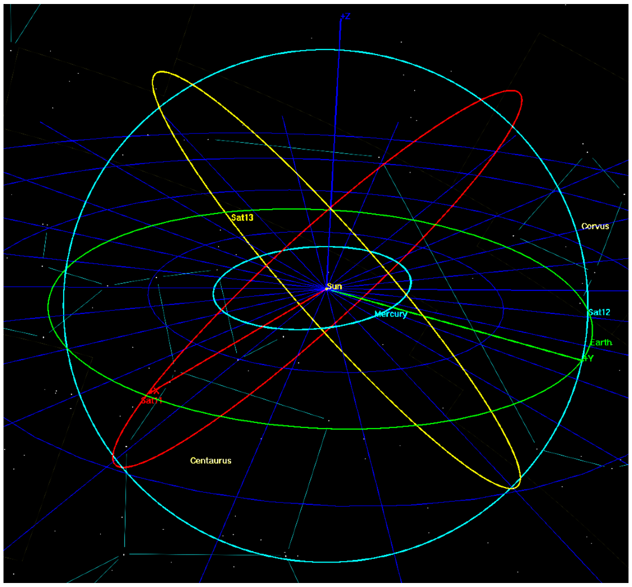


Figure 7. Orbits for a 3-plane Walker-Delta constellation with 54.7° inclination.

Evenly distributed across the planes, the nine satellites achieve constant 4π-sr coverage of the solar surface with a maximum zenith angle of 61° globally and just 49° at the solar poles, as shown in Figure 8. It is worth noting that this figure does not take into account differential solar rotation. The sinusoidal variation across longitude is due to the 7° tilt between the solar equator and the ecliptic, as depicted by Figure 9. The remaining undulating pattern results from the particular configuration of the orbital planes.

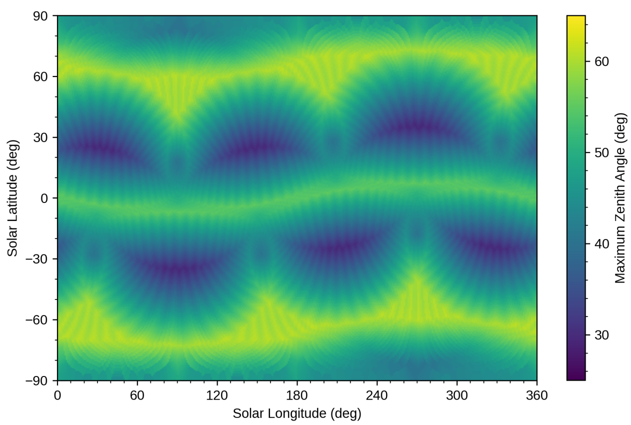


Figure 8. At the solar poles, the maximum zenith viewing angle ζ for the constellation is 49°.

The choice of direct injection to high-inclination orbits for improved observation of the solar poles necessitates unusually high Δv. While electric propulsion requires approximately 60% more Δv to reach the final orbit than chemical propulsion, the high specific impulse (I_{sp}) from the technology offers substantially higher Δv per unit propellant mass. As a result, the recommended propulsion technology that emerged from the trade study was NASA's Evolutionary Xenon Thruster-Commercial (NEXT-C) gridded ion engine [39]. Figure 10 shows that the proposed propulsion solution requires significantly less propellant mass to achieve the same Δv compared to other available technologies.

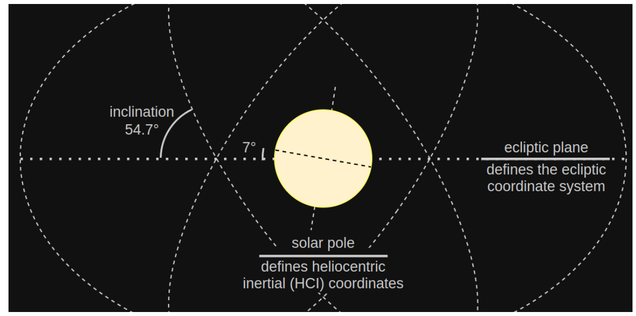


Figure 9. The constellation orbits are defined in the ecliptic plane, tilted 7° relative to the solar equator.

NASA recommended using the NEXT-C ion engine in its 2013 Planetary Science Decadal Survey [40]. As such, the engine's novel capability is a key enabler of this constellation.

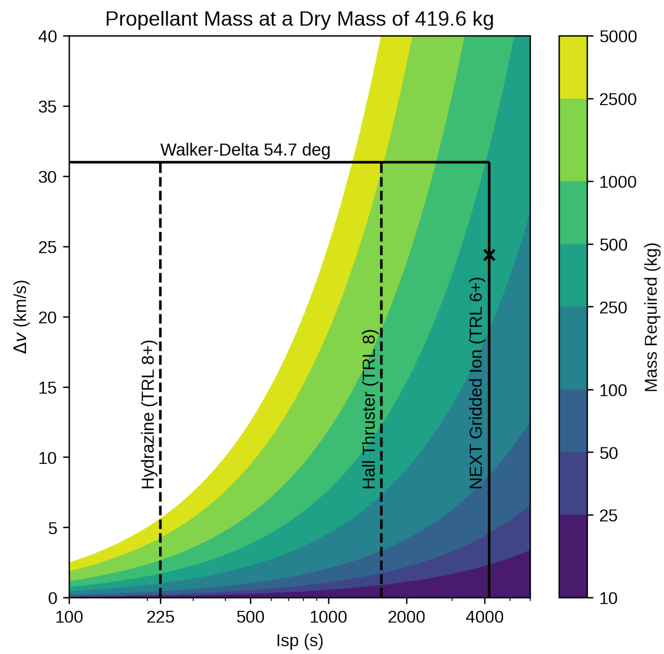


Figure 10. Propellant mass required to achieve a desired Δv for varying engines sorted by I_{sp}. Assumes an individual spacecraft dry mass of 419.6 kg, including 15.4 kg of hydrazine reserved for momentum desaturation.

The launch vehicle selected will also provide about 9.2 km/s of the approximately 38.2 km/s Δv required per spacecraft to reach the final orbits, with the rest from the NEXT-C gridded ion engine on each spacecraft. From a survey of the available launch vehicles, considering cost, risk, and capability, the CDA team selected the Falcon Heavy combined with a STAR 48BV upper stage to launch three SUNFLOWER Observatory spacecraft towards each Walker-Delta orbit (data provided by [41], [42]).

On this basis, a minimum of three launches—one per orbital plane—will be necessary. Although an option exists to increase the number of launches to accommodate one spacecraft per launch vehicle, thereby decreasing the wet mass and overall time of flight for each spacecraft, exercising this option will substantially increase the cost.

Furthermore, the solution is relatively insensitive to variations in the final spacecraft dry mass. An increase in mass increases time of flight rather than making the trajectory infeasible. As the bus design team settled on the final design, the CDA team refined the Δv and trajectory estimates at a spacecraft dry mass of 419.6 kg, including 15.4 kg of hydrazine reserved for momentum desaturation. With an analytical thrust rule for inclination change, and optimized phasing maneuvers, the resulting Δv budget is given in Table 9.

Table 9. **Δv Budget for a Time of Flight of 3.2 Years**

Maneuver	**Δv (km/s)**
Inclination Raising	24.4
Phasing	2.6
Station-Keeping	0.1
Margin	1.9
NEXT-C Total	**29.0**
Launch Vehicle V_∞	**9.2**

The total expected Δv from the NEXT-C xenon engine for the direct injection, including inclination and phasing, is 27.0 km/s. The heliocentric science operations only require about 0.1 km/s for station-keeping over the entire mission lifetime. This leaves about 1.9 km/s of Δv as margin for a total of 29.0 km/s of onboard Δv from the NEXT-C ion engine.

C. Recommended Architecture: Spacecraft Bus

Table 10 captures the recommended bus design decisions.

Table 10. **SUNFLOWER Bus Design Decisions**

Design Space	**Decision**
Attitude Determination	Star Trackers + Sun Sensors
Attitude Control	Reaction Wheels + Thrusters
Power System	Identical Solar Arrays
Communication	Ka-band + X-band Backup
Structure	Rack & Rail
Size	Small/Micro
Material	Machined Aluminum

Based on these design choices, Table 11 offers a mass budget broken down by subsystem. The resulting spacecraft has a dry mass of 404.2 kg (or 419.6 kg including hydrazine) and a wet mass of 849.6 kg.

A labeled depiction of the stowed vehicle architecture is presented in Figure 11. Aluminum brackets connect separate levels of the chassis, each holding different subsystems. During launch the vehicle is secured via hold down and release mechanisms (HDRMs) on the rear face of the vehicle opposite the parabolic antenna. This approach results in a first fundamental frequency of 40 Hz. Additionally, under random vibration loading as prescribed by the SpaceX Payload User Guide [43], the maximum deflection in this configuration is 2 mm, which can be mitigated by adding ribs and supports. The preliminary dynamics analysis performed for PDR showed consistency with these requirements.

Once the vehicle is deployed from the kickstage, its solar panels extend to begin powering the vehicle. The deployed configuration is shown in Figure 12.

Table 11. **Spacecraft Estimated Mass Budget**

Subsystem	**Mass (kg)**
Payload	48.0
Structure & Mechanisms	62.0
Thermal	26.0
Power	75.4
Communications	29.5
ADCS	30.7
Propulsion	102.1
Margin	30.5
Total Dry Mass	**404.2**
Xenon Propellant	430.0
Hydrazine Propellant	15.4
Total Wet Mass	**849.6**

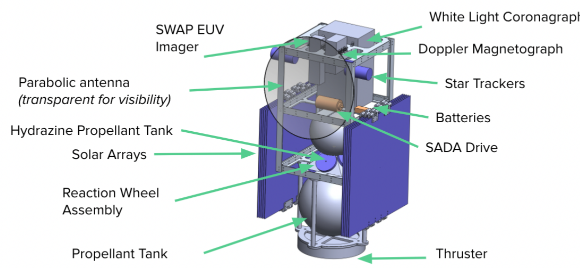


Figure 11. **The integrated SUNFLOWER vehicle in the stowed position.**

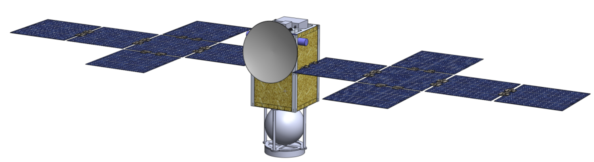


Figure 12. **The deployed SUNFLOWER spacecraft. The volume of the vehicle is 1 m x 1 m x 3 m stowed, with 24.5 m^2 of solar panel area when deployed.**

Given that the vehicle is in a heliocentric orbit that is virtually always in direct sunlight, cooling the vehicle is critical. Analysis of the trade space led to a combination of pumped fluid loops and passive radiators based on heritage from previous solar missions and performance [28]. This analysis showed that the system needs 15–20 kg of mass allocated for a pumped fluid loop system transferring heat load to the radiative surface. The goal is for the fluid loops to remove all excess heat load from the electronics and transfer it onto the radiative surface. With an operational temperature constraint of 300 K, 26.5 m^2 of radiator area will be necessary. The rear faces of the solar panels will supply most of this radiator area to limit the need for additional deployable radiators.

During the conceptual design phase, intersatellite links (ISLs) were investigated as a potential method for circumventing communication blackout periods and enabling near-real-time data transmission back to Earth. Two options were considered for ISLs: optical terminals and K-Band antennas. A free space optical satellite communications link budget was used to calculate the required transmission power for a 1.73-AU optical link [44]. Although the data rates and power requirements for the optical ISLs were appealing, the pointing requirements were orders of magnitude more challenging to accommodate than for the Ka-band link to DSN ($5.7 * 10^{-5}$ degrees budgeted pointing error compared to 0.16 degrees). Likewise, the link and data budget analyses for the K-band ISL revealed a data transmission rate of only 1302 bps at a distance of 1.73 AU. Moreover, Ka-band frequencies experience minimal interference from solar radiation [45], and the recommended constellation architecture enables all SUNFLOWER SC to have direct line of sight to Earth during nominal science operations, reducing the overall value provided by ISLs. In light of these facts, at PDR, ISLs were removed in favor of backup X-band antennas for contingency communication with DSN in case of Ka-band or ADCS anomaly, and risk 3.07 Solar Conjunction was retired.

D. Concept of Operations

Prior to launch, three SUNFLOWER Observatory spacecraft will be integrated into each of three Falcon Heavy fairings, as depicted in Figure 13 [43]. Operations begin with three Falcon Heavy launches from Earth, each approximately 60 days apart. Any error in launch window will directly affect the phase of the final science orbit with respect to Earth, but the launch can be delayed by up to three weeks before significant degradation ($>1\%$) of the zenith viewing angle risks compromising compliance to Requirement 4.02.

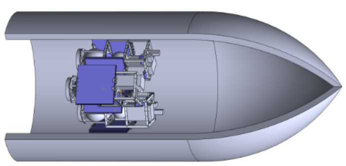


Figure 13. Three SUNFLOWER vehicles shown in a Falcon Heavy fairing on its side.

Each launch will first target a parking orbit of 160 km in altitude before firing the kickstage to escape Earth's orbit. A brief stay in the low parking orbit enables the rocket to take advantage of the high velocity to increase the escape V_∞.

After the upper stage completes its burn, each of the three spacecraft on a given launch will deploy from the rideshare. The ion propulsion system will then deliver each spacecraft to its final inclination and phase over a period of 3.2 years. During this transfer, the SC will change thrust direction twice per revolution to maximize the fuel efficiency of the inclination change maneuvers. Given the low-thrust maneuvering, there is ample time to turn off the engine and power on the instruments to collect data or calibrate instruments during the transfer orbit without affecting time of flight. This is a key benefit of the direct transfer, as the SC will always remain between 0.8 AU and 1.1 AU (with phasing) from the Sun.

At the end of the transfer orbit, the spacecraft will power down the ion engines and orient their science instruments toward the Sun to begin science operations. They will deliver continuous 4π-sr coverage of the Sun with nearly continuous communication to Earth until decommissioning.

The decommissioning procedure will be highly dependent on the context at the end of the eleven-year mission. If the spacecraft are functional after the eleven years, extended operational support can be requested by relevant stakeholders. Should the operators choose to decommission, the ground operations will command the spacecraft to disable battery charging and transmission capabilities of the spacecraft. This will ensure the spacecraft will not perform unplanned maneuvers or cause unnecessary radio interference as their batteries drain one final time. Further analysis regarding the stability of the heliocentric orbits over a long time horizon after decommissioning is recommended as future work.

E. Cost and Schedule

Figure 15 in Appendix D depicts a preliminary quarterly schedule estimate for the SUNFLOWER program, beginning in FY2025 and extending for roughly eighteen years through planned decommissioning. A notional launch date of Dec. 31, 2031, in Q1FY2032 kicks off the operations phase of the program, which continues until the notional decommissioning date of Dec. 31, 2042, at the end of Q1FY2043.

The NASA proposal for the FY2024 budget, at time of writing, features a 6.7% reduced allocation for heliophysics relative to the FY2023 budget [46]. Electing to begin the SUNFLOWER program in FY2025 allows for the release of the upcoming Decadal Survey for Solar and Space Physics (Heliophysics) 2024–2033, which—it is hoped—will identify an increased need for real-time space weather monitoring and prediction. This would encourage NASA and the US government to allocate additional funding to space programs dedicated to heliophysics like the SUNFLOWER Observatory, enabling the program to succeed.

The instrument development and delivery timeline leverages the heritage of the SUNFLOWER instruments—which have already launched [16], are manifested to launch in the near future [17], or are at least space qualified and being prepared for launch [18]—enabling an accelerated window of roughly four years from a NASA Announcement of Opportunity to instrument storage and preparation for delivery. Likewise, the spacecraft and bus development timeline of four years from kickoff through System Integration Review (SIR) is derived from the corresponding timeline for the Ulysses program [14]. The observatory integration phase following SIR through liftoff, based on the corresponding timeline for the Parker Solar Probe, is estimated to be roughly 1.5 years [15]. Finally, the operations phase is baselined as eleven years in accordance with the mission lifetime requirement.

Table 12 offers minimum, expected, and maximum cost estimates, each with 20% margin, for the program over its seven-year development timeline and eleven-year mission lifetime. Each line item depicted in Table 12 represents the summation of lower-level WBS line items omittedfor brevity. One qualification unit is included as part of design and development costs, and the operations phase includes both mission operation and data analysis costs.

In a departure from traditional program cost estimates, Table 12 allocates for the establishment of additional 34-m DSN dishes at Goldstone, Madrid, and Canberra, as well as their operation and maintenance during the SUNFLOWER mission. At time of writing, DSN supports about forty existing deep space missions, with more than thirty additional projects in development [47], [48]. With nine spacecraft in a constellation demanding nearly continuous communication with Earth, SUNFLOWER would place a significant burden on the limited number of 34-m dishes DSN is expected to have by 2031 [48]. Establishing additional stations at each DSN site reduces this burden while simultaneously reducing recurring engineering and mission operation costs for SUNFLOWER.

Table 12. Preliminary Cost Estimates (in Thousands of FY2025 USD)

	Min	Exp	Max
Design & Dev.	300,265	400,353	500,442
Build & Test	637,039	489,385	611,731
Launch & Kick	471,342	627,123	783,904
Operations	164,475	328,950	493,426
DSN Access	855,423	1,465,123	2,237,589
Margin (20%)	431,509	662,187	925,418
Total	2,589,053	3,973,121	5,552,509

The minimum, expected, and maximum cost estimates allocate for the construction of three, five, and seven additional 34-m dishes at each site, respectively [23]. A cost comparison shows that the establishment and maintenance of these dedicated stations is comparable to—if not less expensive than—the corresponding cost of nearly continuous access to DSN for nine spacecraft over eleven years [49].

The total cost for the program is distributed by fiscal year in Figure 14. Note that the annual costs never exceed the FY2024 NASA Heliophysics budget request [46].

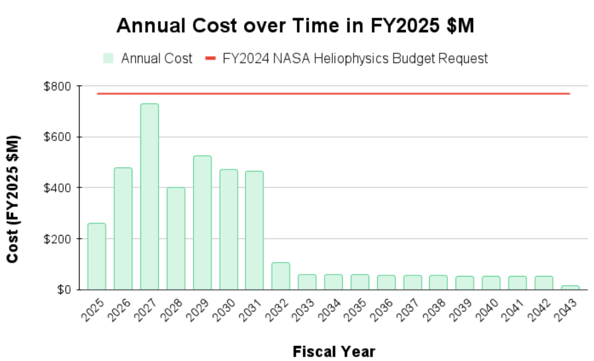


Figure 14. Total cost (in millions of FY2025 USD) distributed by fiscal year.

The average lifecycle cost is estimated to be $4.0 billion in FY2025 USD, which is substantial but ultimately consistent with the number of spacecraft and launches and the scope and lifetime of the program. Of this amount, roughly $2.1B FY2025 USD captures non-recurring engineering—including establishment of new DSN stations—while roughly $1.9B FY2025 USD captures recurring engineering—including DSN station operation and maintenance.

The average annual cost of SUNFLOWER during initial and extended operations was compared to the same costs of the GOES constellation [50], [51], the JPSS constellation [50], [51], and the GPS constellation [52], [53]. The results are given in Table 13. Overall, the average yearly cost of SUNFLOWER—as low as $200 million in FY2025 USD for mission extension—is comparable to that of NOAA Earth observation and solar observation constellations like GOES and JPSS.

The marginal value of SUNFLOWER forecasts for a "high performance" event (like the 2003 Halloween storms) and a "max performance" event (like the 1859 Carrington event) are given in Table 14 [54]. This data is taken from a 2018 study that examined the economic impacts of geomagnetic storms in Europe, modeling current space-based forecasts against improved forecasts beyond the Sun-Earth line [54]. Overall, the improved forecasting offers marginal expected average savings of $3.7B in global value yearly for a 2003-like solar event and $32.5B yearly for a catastrophic Carrington-level event [54]. Because this study assumed Europe to be the primary target, these cost estimates would likely increase for North America or Asia. Regardless, the exploration of marginal value shows that SUNFLOWER would provide value well beyond its costs in the event of intense solar weather events, in addition to the substantial scientific value provided by the constellation's continuous observation of the Sun's entire surface.

V. Conclusion

A. Summary

The objective of the SUNFLOWER Observatory is to provide continuous, comprehensive, long-term remote sensing across the entirety of the solar surface. With the selected science instruments, SUNFLOWER will provide heliophysicists with unprecedented observational capability, enabling improved understanding of the solar dynamo and more accurate space weather predictions, among other benefits.

The SUNFLOWER Observatory mission consists of nine total spacecraft arranged in a three-plane heliocentric Walker-Delta constellation. Each orbital plane contains three spacecraft in 1 AU circular orbits inclined at 54.7° relative to the ecliptic plane. Critically, this constellation geometry will allow for constant visual coverage of all solar latitudes with a viewing angle of no more than 61° measured from nadir. Spacecraft will be launched in groups of three onboard Falcon Heavy rockets with STAR 48BV kickstages. Upon separating from the kickstage, the spacecraft will use onboard electric propulsion to phase apart and continually raise orbital inclination until the target orbits are reached approximately 3.2 years after launch.

All nine spacecraft will be equipped with a full-disk Doppler magnetograph, EUV imager, and white light coronograph, allowing for observation of CMEs, sunspots, and coronal holes. To satisfy the identified deployment and station-keeping requirements, each spacecraft is equipped with a NEXT-C gridded ion engine with 430 kg of onboard xenon propellant as well as momentum-offloading thrusters with 15.4 kg of hydrazine propellant. Further, each spacecraft will house onboard star trackers, sun sensors, reaction wheels, twin solar arrays with secondary batteries, a machined aluminum rack and rail structure, and Ka-band nominal communication with Earth via the Deep Space Network. Such a design results in an estimated spacecraft dry mass of approximately 404.2 kg.

In the current schedule estimate, the mission's instrument and spacecraft bus development will start in FY2025 with launches in FY 2032 and decommissioning in FY2043. SUNFLOWER will provide at least 7.8 years of continuous solar observation, with the potential to extend the mission much longer. With the present spacecraft design, this schedule results in a total cost estimate of $4.0B in FY2025 dollars.

B. Future Work Beyond PDR

For future work, in the near term, the bus design team and CDA architectures would be matured to higher fidelity. For the spacecraft bus, specific vendors for spacecraft components and construction would be selected, and the thermal and

Table 13. **Lifecycle Cost Comparison (in Millions of FY2025 USD)**

	GOES	JPSS	GPS	SUNFLOWER
Lifecycle Cost	11,700	11,322	5,207	3,980
Average Cost per Year	334	808	158	200–362
Est. Value Added per Year	5,500	5,500	125,000	1,882–6,405

Table 14. **Marginal Value Scenarios (in Millions of FY2025 USD)**

	High Performance Scenario	Max Performance Scenario
Event Type	2003 Halloween Storm	Carrington Event
Total Global Marginal Benefit	$37,635	$960,793
Likelihood/Timespan	20 yrs	150 yrs
Average Marginal Value over Timespan	$1,882/yr	$6,405/yr

structural analyses would be performed with industry-grade software to increase confidence in the proposed solution. Likewise, the CDA team would work on nonlinear optimization of the spacecraft trajectories to their final science orbits and a stability analysis of the various proposed heliocentric orbits after decommissioning. Over a longer term, the project would seek approval and funding through a NASA proposal.

APPENDICES

A. REQUIREMENTS MATRIX

Table 15 lists the set of requirements for the SUNFLOWER program, including requirement identification number, title, description, rationale, category, and verification and validation (V&V) method at PDR.

B. RISK MATRIX

Table 16 provides the title of each risk presented at PDR, along with the corresponding risk realization in the form of an if-then statement. Table 17 enumerates the potential mitigation strategies for reducing the total risk score for each risk through either likelihood or consequence reduction. Mitigation strategies that were implemented during the preliminary design phase are presented in bold.

C. SCIENCE TRACEABILITY MATRIX

The STM (Table 18) was developed to streamline the SUNFLOWER instrumentation suite. The STM also helps to guide cadence determination and instrument suite calibration. The scientific questions are derived from current key problems within both the space weather and heliophysics spheres. Instrument requirements were derived from expert assessments of relevant phenomena to ensure the collection of the correct data [5] and from the performance specifications of comparable instruments to ensure that the SUNFLOWER suite would be competitive [55], [56]. Projected instrument performance is derived from [30] for CCOR-1, from [18] for CDM, and from [57] for SWAP.

D. GANTT CHART

Figure 15 offers a quarterly Gantt chart of the SUNFLOWER program. The schedule begins in FY2025 and is truncated after FY2035 for easier viewing. Activities beyond the cutoff are continued primary operations and decommissioning in the 73rd quarter, after the eleven-year primary mission.

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BIOGRAPHY

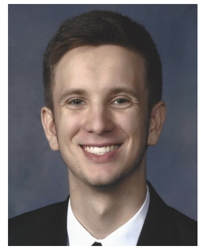
Allan Shtofenmakher *is the chief engineer for the SUNFLOWER Observatory program, as well as a PhD candidate at the Massachusetts Institute of Technology (MIT). His research interests broadly reside at the intersection of space situational awareness (SSA) and control of multi-agent systems, with a focus on tracking orbital debris using in-space satellite sensors. He holds an MS degree in Aeronautics and Astronautics from Stanford University and a BS degree in Aerospace Engineering from University of California, Irvine.*

Daniel Gochenaur *is a Constellation Design and Astrodynamics engineer for SUNFLOWER. He is a PhD student and NDSEG fellow in MIT's Department of Aeronautics and Astronautics. Daniel holds an MPhil in Supersonic Aerodynamics from the University of Cambridge and a BS in Aeronautics and Astronautics from Purdue University. His present research interests center on atmospheric entry and multidisciplinary design optimization.*

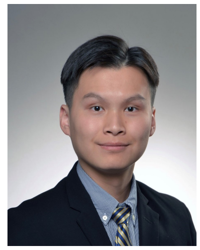
Ben Waters *is the systems engineer on the SUNFLOWER Observatory mission primarily responsible for integration with the Heliophysics and Solar Monitoring subteam. He is an SM candidate at MIT with interests in space systems focused on orbital debris remediation and risk management. He holds a BS degree in Astronautical Engineering from the U.S. Air Force Academy.*

Robert Cato III *is a systems engineer on the SUNFLOWER Observatory mission working closely with the spacecraft bus design subteam. He graduated from MIT in June 2023 with an SB in Aerospace Engineering. He is a flight systems engineer at NASA JPL as of August 2023. His interests include systems engineering, robotics, and autonomy.*

Duncan Miller *is the systems engineer responsible for integration related to the Technology Roadmapping subteam. He brings 10 years of experience as a naval aviator leading complex missions involving remote sensing and reconnaissance. Duncan earned his BS in Physics from the U.S. Naval Academy in 2012 and is pursuing his SM in Engineering and Management from MIT.*



Luke de Castro *is a systems engineer for the SUNFLOWER Observatory who works closely with the Constellation Design and Astrodynamics subteam. After graduating with an SB from MIT in June 2023, he is pursing his SM in Aerospace Engineering at MIT, studying optimal trajectory design and control for agile unmanned aerial vehicles.*



Tai Zheng *is a member of the Systems Engineering subteam for the SUNFLOWER Observatory mission. He graduated from MIT in June 2023 with an SB in Aerospace Engineering. His career interests include space exploration and systems engineering, and he is a satellite systems engineer at Boeing as of September 2023.*

Alexander Koenig *is the Constellation Design and Astrodynamics lead for SUNFLOWER. He is a graduate of MIT (dual SB in Physics and Aerospace Engineering 2022, SM in Aeronautics and Astronautics 2023) and presently works as a Starlink GNC engineer at SpaceX as of June 2023. Alongside constellation design and operation, his professional interests include observational astrophysics, multi-messenger astronomy, and SSA.*

Katelyn Sweeney *is the engineering lead of the bus design subteam for the SUNFLOWER Observatory program. She specializes in dynamic and thermal analyses of complex spacecraft systems. She received her SB in Mechanical Engineering from MIT in 2018 as well as her MS in Engineering Sciences and her MBA from Harvard University in 2023.*

Joel Jurado Diaz *is the lead telecommunications engineer on the bus design subteam for the SUNFLOWER Observatory mission, as well as a visiting student in the Department of Aeronautics and Astronautics at MIT. He graduated from Universitat Politècnica de Catalunya in July 2023 with a BS in Telecommunications Engineering. His interests include space communications, the architecture of connectivity systems, and optical communications.*

Alexis Lepe *is the Heliophysics and Solar Monitoring subteam lead. They graduated with an SB in Aerospace Engineering from MIT in June 2023. Their interests include optical astronomy, spacecraft controls, and embedded systems.*

Claire McLellan-Cassivi *is a member of the Heliophysics and Solar Monitoring subteam for the SUNFLOWER Observatory mission. They are working on a dual SB in Earth, Atmospheric, and Planetary Sciences and Engineering, with plans to graduate in February 2024. Their interests include optical and x-ray astronomy, communication systems, and remote sensing systems.*

Frederick Ajisafe *is a member of the SUNFLOWER Heliophysics and Solar Monitoring subteam. He graduated from MIT in June 2023 with an SB in Aerospace Engineering, with plans to complete an SM in Aeronautics and Astronautics. His interests include orbital dynamics, data analysis, and applications of machine learning to systems engineering and natural phenomena.*

Akila Saravanan *is a member of the SUNFLOWER testbed subteam. She graduated in June 2023 from MIT with a dual SB in Aerospace Engineering and Computer Science. She is continuing at MIT in pursuit of an SM in Aeronautics and Astronautics, working on optimization problems to dynamically allocate and schedule UAVs for search-and-rescue operations. Her interests include autonomy and perception.*

Joana Nikolova *is the primary ADCS engineer on the bus design subteam for the SUNFLOWER Observatory mission. She graduated from MIT with an SB in Aerospace Engineering in the spring of 2023. She is currently working on her SM in Aeronautics and Astronautics at MIT, investigating machine learning capabilities for scheduling ground based observation of orbiting objects. Her interests include control, dynamics, planning, and autonomy.*

Clara Ziran Ma *is a member of the Technology Roadmapping subteam for SUNFLOWER. She is an SM student in the Technology and Policy Program and Department of Aeronautics and Astronautics at MIT. In her research, she forecasts the evolution of the launch industry in the coming decades and uses atmospheric chemistry simulations to model rocket emissions. Clara has a BS in Science, Technology, and International Affairs from Georgetown University.*

Leilani Trautman *is a graduate teaching assistant for the MIT space systems engineering capstone course and serves as an advisor to the SUNFLOWER team. She is currently an MEng student in the department of Electrical Engineering and Computer Science at MIT, and her research is focused on mission automation for the Mars Perseverance rover through a collaboration with NASA JPL. She earned her SB in Electrical Engineering and Computer Science from MIT in 2021.*

Nadia Khan *is a graduate researcher in the Engineering Systems Lab in MIT's Department of Aeronautics and Astronautics, pursuing an SM in Technology and Policy at MIT's Institute for Data Systems and Society. She served as a graduate teaching assistant for the MIT spacecraft systems engineering capstone course. Nadia also led the heliophysics case study for the NASA-MIT Advanced Space Technology Roadmap Architecture (ASTRA) project. Nadia is currently working as a business development trainee in ESA's Human and Robotic Exploration directorate.*

Olivier de Weck *is the Apollo Program Professor of Astronautics at the Massachusetts Institute of Technology where he is the director of the Engineering Systems Laboratory. His research is in systems engineering with a focus on how complex technological systems are designed and how they evolve over time. He is a Fellow of INCOSE and a Fellow of AIAA and serves as editor-in-chief of the Journal of Spacecraft and Rockets.*

Edward Crawley *is the Ford Professor of Engineering at MIT. His research has focused on architecture, design, and decision support and optimization in complex technical systems subject to economic and stakeholder constraints. From 2011 to 2016, he served as the founding president of the Skolkovo Institute of Science and Technology, Moscow. Prior to that, he served as the director of the Bernard M. Gordon-MIT Engineering Leadership Program. From 2003 to 2006, he served as the executive director of the Cambridge-MIT Institute.*

Table 15. Requirements Matrix

ID	Requirement Title	Requirement Description	Rationale	Category	V&V
1.01	Spacecraft Launch Vehicle Compatibility	Each spacecraft shall be physically compatible with the SpaceX Falcon Heavy rocket and STAR 48BV motor	Outputs of launch vehicle trade study analysis, offering optimal performance weighed against cost and risk	Interface	Inspection
1.02	Spacecraft Communication Network Compatibility	Each spacecraft shall be compatible with the NASA Deep Space Network (DSN) for communication and ranging	Pre-existing DSN infrastructure optimal for deep space communication	Interface	Inspection
1.03	Spacecraft Lifetime	The suite of scientific instruments and spacecraft hardware shall be designed for survival over 11 years in the environment of the heliosphere	Average duration of a solar cycle is 11 years	Environment/ Reliability	Similarity/ Analysis
2.01	Scientific Instruments	The suite of scientific instruments on each spacecraft shall be capable of collecting information to support global heliophysics research and provide more accurate space weather forecasting	Flow-down from SUNFLOWER Observatory mission description	Functional	Inspection
3.01	Spacecraft Attitude Control Capability	Each spacecraft shall support a pointing performance at least as accurate as 0.16° during communication, science, and thrust operations	Required to support Ka-band communication with DSN at 2 AU, which is limiting case for pointing performance	Performance	Analysis
3.02	Spacecraft Attitude Determination Capability	Each spacecraft shall support a pointing knowledge at least as accurate as 0.08° during communication, science, and thrust operations	Required to support Ka-band communication with DSN at 2 AU, which is limiting case for pointing performance	Performance	Analysis
3.03	Spacecraft Data Communication	Each spacecraft shall support transmission of at least 500 MB/day of state-of-health and science data to Earth during nominal cruise operations	Flow-down from key science objectives, based on bus and science instrument data generation analysis	Performance	Analysis
3.04	Spacecraft Onboard Data Storage	Each spacecraft shall be capable of storing at least 4 GB of pertinent state-of-health and science data in the event of a communication blackout	Flow-down from key stakeholders; enables storage of 1 week of pertinent data for transmission at a later date	Performance	Analysis
4.01	Solar Surface Coverage	The constellation of spacecraft shall have 4π-sr continuous observation of the Sun during nominal science operations	Flow-down from SUNFLOWER Observatory mission description	Performance	Analysis
4.02	Solar Pole Viewing Angle	The constellation of spacecraft shall have a zenith viewing angle ζ of the Sun's poles not to exceed 60° during nominal science operations	Flow-down from key stakeholders	Performance	Analysis
4.03	Δv for Constellation Placement	Each spacecraft shall be capable of providing at least 27.1 km/s of Δv to support orbital inclination changing, phasing, and station-keeping maneuvers after launch vehicle separation	Corollary of Requirement 1.01, enabling greater flexibility for meeting overall mission objectives	Performance	Analysis

Table 16. **SUNFLOWER Risk Statements**

Risk ID	Risk Title	Realization
~~1.01~~	Radiation Failure (REMOVED)	IF the spacecraft modules are not sufficiently resistant to radiation failures, THEN the modules could degrade in performance or fail completely
1.02	Fuel Requirements Prohibitive	IF the amount of fuel required for each spacecraft is prohibitively high, THEN the constellation architecture solution may not close
1.03	High Launch Vehicle Costs	IF specialized launch vehicle (LV) performance or multiple launches are required, THEN launch costs may become prohibitively expensive
2.01	Payload Thermal Management	IF payload temperatures exceed maximum non-operational survival limits, THEN payload functionality may degrade or cease
2.02	Payload Radiation Management	IF payload sensors, lenses, etc., are not properly protected for the heliosphere radiation environment, THEN instruments may degrade and data quality may decrease
2.03	Payload Pointing Degradation	IF attitude control is degraded or lost due to hardware degradation or a C&DH error, THEN payload observation capability may diminish or cease
3.01	Bus Radiation Shielding	IF the SC does not have sufficient radiation shielding or shielded components, THEN data will be corrupted and components will fail or lose accuracy
3.02	Bus Thermal Management	IF the SC has insufficient thermal offload or balancing, THEN component functionality will break down due to unsafe thermal ranges
3.03	Bus Power Supply Failure	IF the battery pack fails, THEN the spacecraft may need to operate on photovoltaic power alone
3.04	Actuator Degradation/Failure	IF the actuators become incapacitated either through a procedural error or due to long-term use, THEN the spacecraft may suffer degraded attitude control capability
3.05	Link Budget Closure	IF the spacecraft link budget can only support a limited data rate, THEN the spacecraft may not be able to downlink all science data in near real time
~~3.06~~	Mass Budget Closure (REMOVED)	IF the spacecraft wet mass exceeds 666 kg, THEN three spacecraft cannot fit into a single LV, resulting in the need for more launches and thus increased cost
~~3.07~~	Solar Conjunction (REMOVED)	IF the spacecraft attempts to transmit data past the Sun to Earth, THEN the data will lose coherency due to the noise produced by the Sun
3.08	Spacecraft Center of Mass and Thrust Vector Alignment	IF the thrust vector is excessively misaligned with respect to the center of mass of the spacecraft, THEN there will not be enough hydrazine to desaturate the reaction wheels over the mission lifetime
4.01	Missed Launch Window	IF the SC or LV experiences delays that result in a missed launch window, THEN the program will experience substantial schedule delays
4.02	Suboptimal Mission Design Tools	IF analysis and orbit optimization tools fail to offer full coverage of the orbit tradespace or take too long to run, THEN the constellation design may be suboptimal and could cause program delays
4.03	Excessive Time to Final Orbit	IF the selected trajectory takes a long time to reach the final desired science orbits, THEN the mission will have a reduced timeline for generation of useful science data
4.04	Excessive Fuel Demands	IF the total mass and Δv requirements for the constellation exceed the capabilities of current launchers, THEN the mission concept could be declared infeasible due to inability to launch under existing LV mass limits
4.05	Low TRL of NEXT-C Ion Thruster	IF the NEXT-C ion thruster is not available to achieve the required Δv or high duty cycle, THEN the direct injection architecture may be infeasible

Table 17. **Risk Mitigation Strategies**

Risk ID	Mitigation Strategies
~~1.01~~	1) Select radiation-hardened modules for use in spacecraft; 2) Build in additional aluminum shielding to protect from ionizing radiation; 3) Design architecture to be fully hardware redundant
1.02	**1) Explore alternative propulsion options, such as electric;** **2) Explore more efficient trajectories and planetary flybys;** 3) Reduce redundancy within the spacecraft to save mass
1.03	**1) Conduct thorough trade study on launch options to inform decision;** **2) Consider leaner constellation architecture to minimize quantity of LVs needed;** **3) Reduce SC mass to fit multiple on one LV**
2.01	1) Design thermal protection system with heritage techniques to meet payload temperature limits with margin; **2) Choose orbital distance from the Sun that reduces risk of exceeding non-operational temperatures**
2.02	**1) Radiation harden all relevant payload components;** 2) Encase payload electronics in protective casing
2.03	**1) Design the payload and science mission to require minimal maneuvering of each SC during science operations;** **2) Design all attitude modes to allow scientific observation;** 3) Consider phased array antenna to limit attitude adjustments needed for communications
3.01	**1) Research and implement radiation hardening techniques and encoding requirements on non-robust components;** 2) Include redundant components for critical modules
3.02	1) Research previous thermal balancing methods taken by similar solar observers and associated data; 2) Encode strict requirements on thermal management; **3) Add sufficient heaters and radiators to improve thermal control**
3.03	1) Perform a failure modes and effects analysis (FMEA) to determine what could cause battery failure; **2) Ensure there are at least two battery modules**
3.04	1) Remove friction-based control mechanisms (e.g., reaction wheels); 2) Include sufficient Δv margin for contingency attitude control maneuvers; **3) Include redundant attitude control systems in the design**
3.05	**1) Use larger, higher-gain antennas on spacecraft to increase data throughput;** 2) Ensure sufficient power is available for high-power RF output; 3) Add additional satellites to the constellation to decrease relay distance and increase data throughput; 4) Reduce amount of scientific data communicated back to Earth; **5) Confirm there are no Ka-band conjunctions**
~~3.06~~	1) Allocate less mass to scientific payload and scale the rest of the spacecraft accordingly
~~3.07~~	Early analysis showed this risk would not be realized either through ISLs or through a solar-noise-resilient antenna with appropriate constellation architecture designed to enable line-of-sight communication directly to Earth
3.08	**1) Store additional hydrazine;** **2) Add a gimbal to the electric propulsion thruster to dynamically change the thrust vector**
4.01	**1) Build substantial margin into the schedule to reduce the likelihood of risk realization;** **2) Design a constellation that allows for regular launch windows (e.g., no flybys)**
4.02	**1) Build substantial margin into the schedule to reduce the likelihood of risk realization;** **2) Maintain a rigorous schedule;** **3) Rely on first principles as a backup to novel optimization methods**
4.03	**1) Use gravity assists sparingly;** **2) Investigate trades between trajectory time and propulsion technology selection;** **3) Design a constellation that can generate useful science data on approach to final trajectories/orbits**
4.04	**1) Maintain awareness of current LV capabilities when designing the constellation;** **2) Investigate trades between SC mass and LV performance;** 3) Explore possibility of launching smaller but more numerous spacecraft with distributed capabilities
4.05	1) Develop spare constellation architecture and trajectory with possibly worse performance but lower Δv requirements; **2) Track the NEXT-C engine development in current projects**

Table 18. **Science Traceability Matrix**

Science Goals	Science Objectives	Scientific Meas. Requirements		Instr.	Instr. Performance		
		Physical Parameters	**Observables**		**Param.**	**Goal**	**Projection**
What are the flux boundaries of the Sun's magnetic sphere?	Examine coronal holes to determine structure and location of flux boundaries	Magnetic flux around coronal holes, esp. around poles	Magnetogram at poles	CDM	AR (arcsec)	1.0	1.2
	Determine occurrence of active regions	Location and number of sunspots	Image of photosphere	SWAP	AR (arcsec)	1.00	3.17
					max FOV (SR)	3.0	2.5
What are the primary drivers of solar wind?	Observe nature of interaction between CMEs and coronal holes in Sun's magnetic field to determine structure of outgoing solar wind	Magnetic flux and structure of corona	Coronagraphic imaging	CCOR-1	AR (arcsec)	6–30	33
					FOV (SR)	3.0–15.0	3.7–18.7
			Continuous-FOV EUV imaging	SWAP	spectral range (nm)	17.1–19.3	16.6–19.5
			LOS full-disk photospheric magnetograms	CDM	AR (arcsec)	7.0	1.2

Instr. = instrument, Param. = parameter, FOV = field of view, LOS = line-of-sight, AR = angular resolution, SR = solar radii

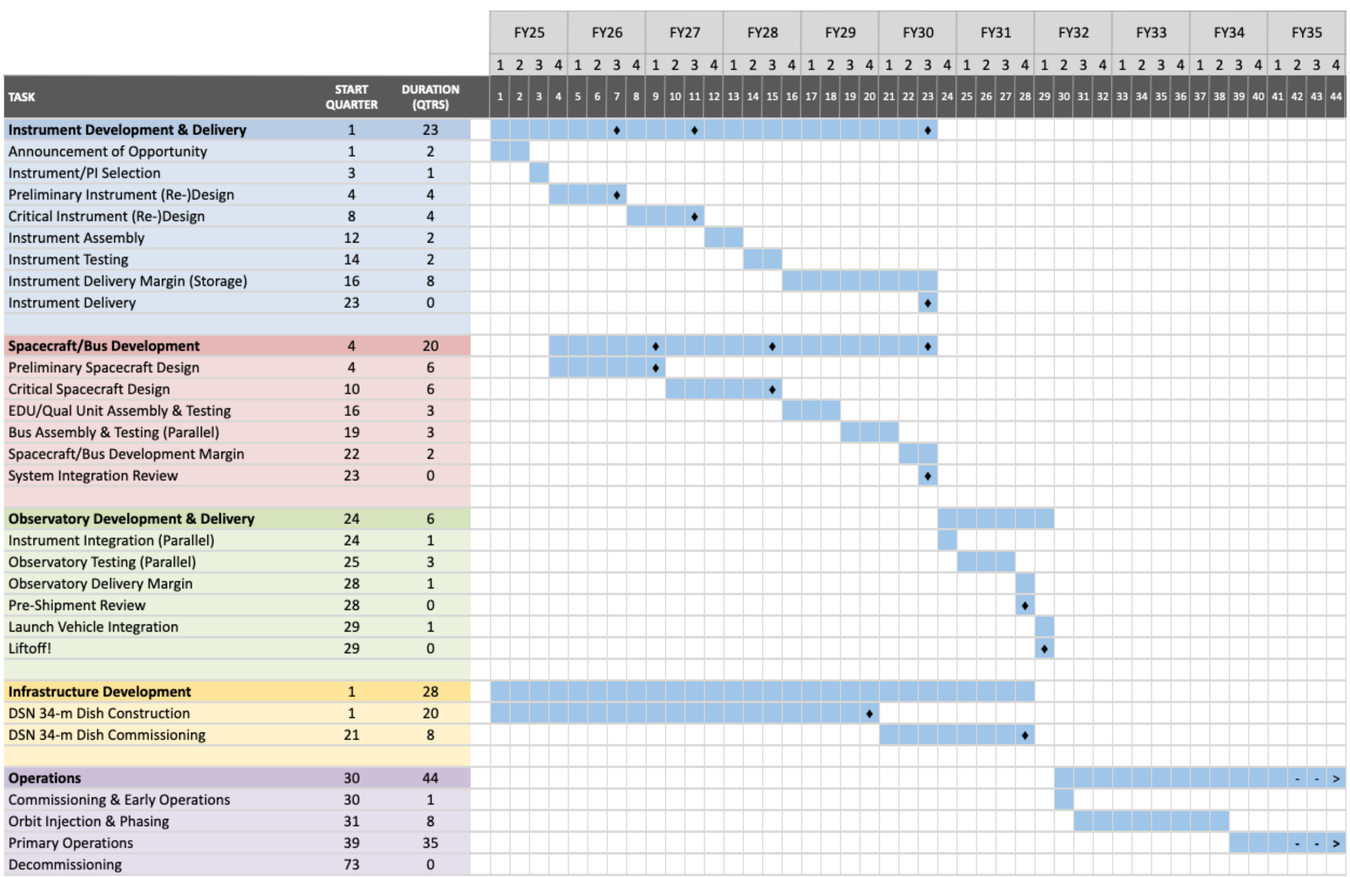


Figure 15. **Quarterly Gantt chart for the SUNFLOWER program.**