2. DESCRIPTION AND COMPARISON OF ALTERNATIVES

The purpose of the Mars 2020 mission is to continue the National Aeronautics and Space Administration's (NASA's) in-depth exploration of Mars. Specifically, the mission would consist of a science-focused, highly mobile rover designed to explore and investigate in detail a site on Mars that was likely once habitable. The mission concept includes new *in situ* scientific instrumentation designed to seek signs of past life. This instrumentation would be used to select a suite of samples, which would be stored in a returnable cache. The mission would also demonstrate technology for future exploration of Mars (both robotic and human missions).

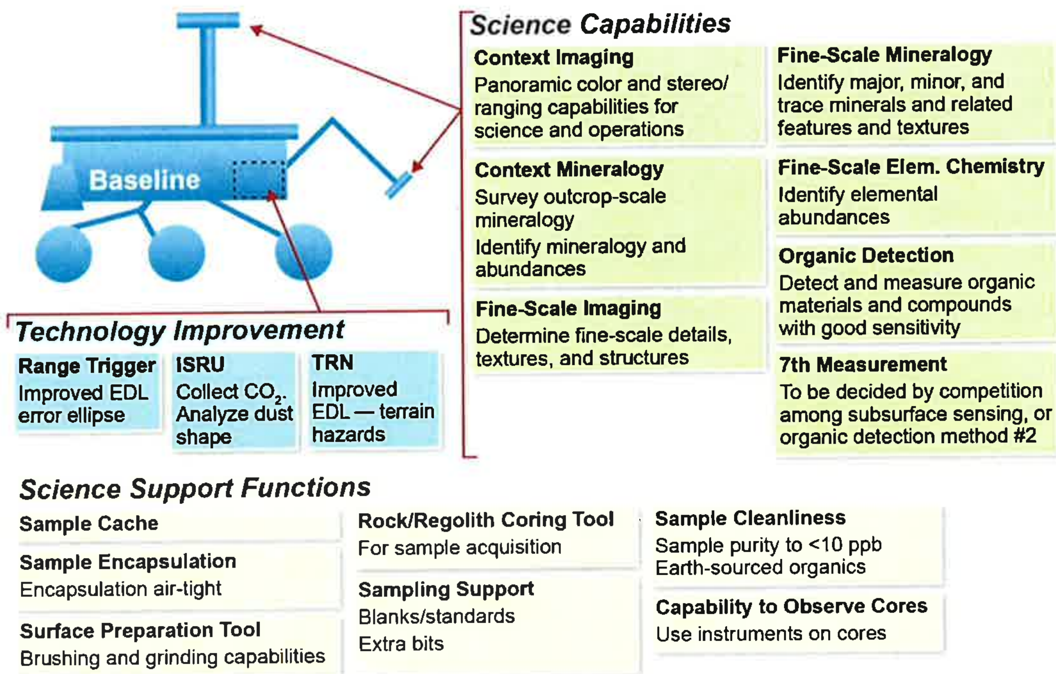
This chapter of the Draft Environmental Impact Statement (DEIS) for the Mars 2020 mission describes and compares the following alternatives:

- **Proposed Action (Alternative 1, NASA's Preferred Alternative) —** NASA proposes to continue preparations for and implement the Mars 2020 mission to the surface of Mars. The proposed Mars 2020 spacecraft would be launched on board an expendable launch vehicle from Kennedy Space Center (KSC) or Cape Canaveral Air Force Station (CCAFS), Brevard County, Florida, during a 20-day launch opportunity that runs from July through August 2020, and would be inserted into a trajectory toward Mars. Should the mission be delayed, the proposed Mars 2020 mission would be launched during the next available launch opportunity in August through September 2022. The rover proposed for the Mars 2020 mission would utilize a radioisotope power system to continually provide heat and electrical power to the rover's battery so that the rover could operate and conduct science on the surface of Mars. A description of the Proposed Action is presented in Section 2.1.
- **Alternative 2 —** In this Alternative, NASA would discontinue preparation for the Proposed Action (Alternative 1) and implement an alternative configuration for the Mars 2020 mission to Mars. The Mars 2020 rover would utilize solar power as its source of electrical power to operate and conduct science on the surface of Mars. The alternative Mars 2020 spacecraft would still be launched on board an expendable launch vehicle from KSC or CCAFS, Brevard County, Florida, during a 20-day launch opportunity that runs from July through August 2020, and would be inserted into a trajectory toward Mars. Like alternative 1, should the mission be delayed, the proposed Mars 2020 mission would be launched during the next available launch opportunity in August through September 2022. A description of Alternative 2 is presented in Section 2.2.
- **Alternative 3 —** In this Alternative, NASA would discontinue preparations for the Proposed Action (Alternative 1) and implement an alternative configuration for the Mars 2020 mission to Mars. The Mars 2020 rover would utilize solar power as its source of electrical power to operate and conduct science on the surface of Mars. The rover thermal environment would be augmented by the thermal output from Light-Weight Radioisotope Heater Units (LWRHUs) to help keep the rover's onboard systems at proper operating temperatures. The Mars 2020 spacecraft would still be launched on board an expendable launch vehicle from KSC or CCAFS, Brevard County, Florida, during a 20-day launch opportunity that runs

from July through August 2020, and would be inserted into a trajectory toward Mars. Should the mission be delayed, the proposed Mars 2020 mission would be launched during the next available launch opportunity in August through September 2022. A description of Alternative 3 is presented in Section 2.3.

- **No Action Alternative** — NASA would discontinue preparations for any Mars 2020 mission and the spacecraft would not be launched. A description of the No Action Alternative is presented in Section 2.4.

The Mars 2020 Science Definition Team (SDT) report (Mars 2020 SDT 2013) suggested baseline[3] operational capabilities for the Mars 2020 mission. These capabilities were part of the basis for capability requirements that NASA provided both in an Announcement of Opportunity for Mars 2020 Investigations (NASA 2013d) and for the landing site selection process (NASA 2014b). The capability requirements for the proposed Mars 2020 mission are summarized in Figure 2-1 and Table 2-1. Achieving these baseline capabilities would maximize the potential for the mission to be most responsive to real-time discoveries and fulfill its comprehensive science objectives.



Source: Mars 2020 SDT 2013

Figure 2-1. Baseline Science and Technology Capabilities for Mars 2020 Mission

Table 2-1. Baseline Operational Capabilities for the Mars 2020 Mission

[3] Baseline is defined as measurements or capabilities necessary to achieve the science objectives of the mission and a point of departure from where implementation begins. The SDT report defined a threshold level as a measurement or capability level below which a mission may not be worth the investment.

Launch Related Capability
Be ready for launch during the 2020 Mars opportunity
Be compatible with an intermediate/heavy class expendable launch vehicle
Arrival and Landing-Site Related Capability
Provide data communication throughout critical events at a rate sufficient to determine the state of the spacecraft in support of fault reconstruction
Be capable of landing on the surface of Mars within a 25 km x 20 km (16 mi x 12 mi) elliptical target area. Improved ability to avoid terrain hazards within the targeted landing area.
Be capable of landing between 30° north and 30° south latitudes
Be capable of landing and operating at an elevation of up to +0.5 km (about 0.3 mi) as defined by the survey by the Mars Orbiter Laser Altimeter
Functional Capability
Be designed to operate for at least one Mars year (687 Earth days)
Be capable of adequate mobility to ensure representative measurement of diverse sites at distances of at least 20 km (12 mi)
Science Capability
Accommodate the NASA-selected science payload capable of definitively analyzing the mineralogy, chemistry, texture, and structure of surface and near-surface materials; and be capable of detecting organic material. Instrumentation suite would include the capability for: context imaging, context mineralogy, fine-scale imaging, fine-scale mineralogy, fine-scale elementary chemistry, and organic detection.
Be able to select, acquire, process, distribute, analyze and cache at least 38 samples of rock, rock fragments, and soil of high scientific interest.
Technology Capability
Demonstrate a technology enabling future human missions to Mars

2.1 DESCRIPTION OF THE PROPOSED ACTION (ALTERNATIVE 1)

The mission and spacecraft for the Proposed Action (Alternative 1) would be designed and developed to meet the baseline operational capabilities. The descriptions presented in this section are based on the information available at the time this DEIS was prepared. Should NASA make changes in the Proposed Action (Alternative 1) that are relevant to environmental concerns, NASA would evaluate the need for additional environmental analysis and documentation.

2.1.1 Mission Description

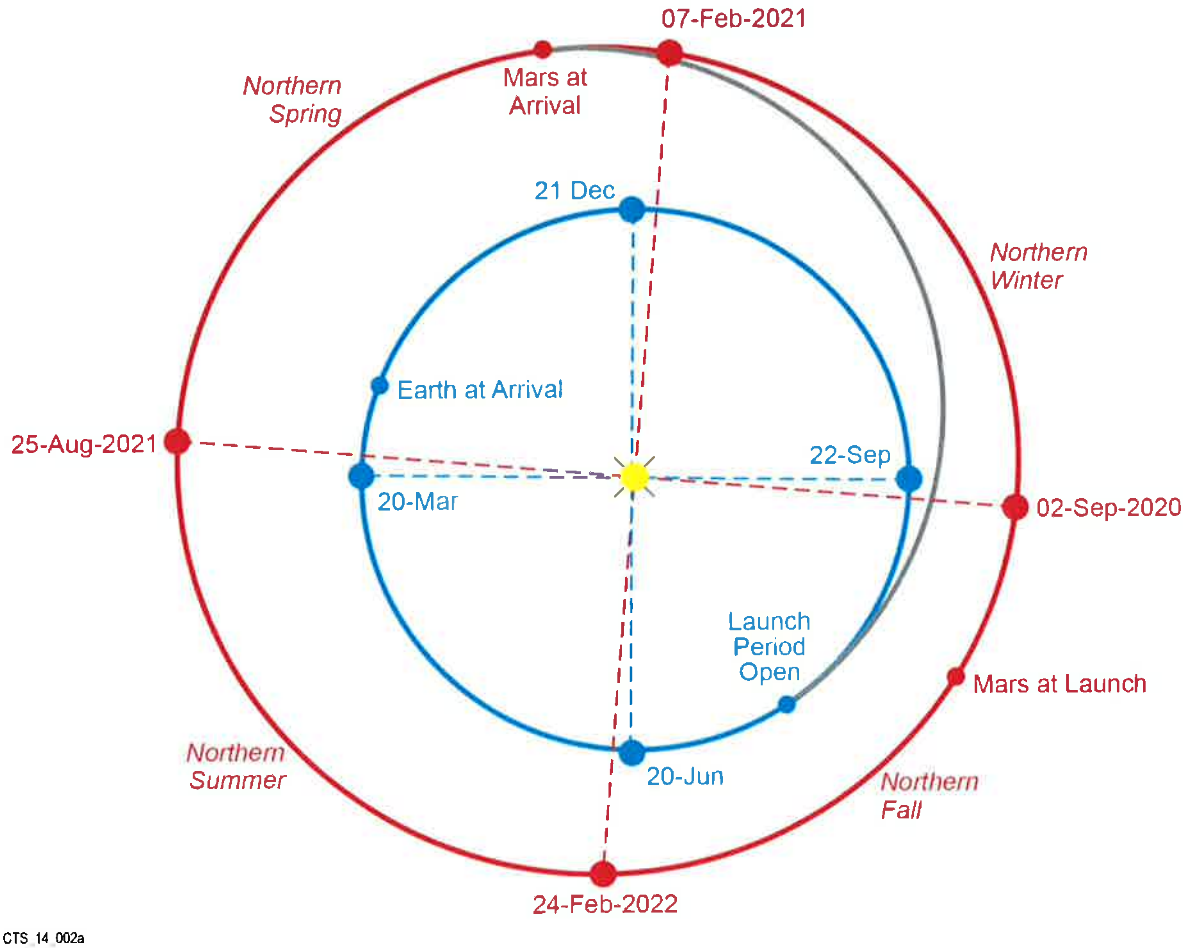
The Mars 2020 spacecraft (described in Section 2.1.2) would be launched from KSC or CCAFS onboard an Atlas V, Delta IV, or Falcon Heavy class of expendable launch vehicles. The launch would occur within an approximate 20-day launch period opening in July of 2020 and closing in August of 2020. Should the Mars 2020 mission not launch during this launch period, it would launch during the next available launch opportunity—August through September 2022. The mission cruise phase would begin when the spacecraft separates from the launch vehicle and would end prior to atmospheric entry at Mars. The cruise phase would last approximately 7 months depending on the exact launch date, trajectory, and selected landing site.

The spacecraft's trajectory from Earth would be designed for a direct entry into the Martian atmosphere, without the spacecraft first entering into orbit around Mars. A final trajectory correction maneuver would be performed prior to separation of the cruise stage from the entry vehicle. Cruise stage separation would occur from 20 to 40 minutes before atmospheric entry. The cruise stage would enter the Martian atmosphere and would break apart and burn up from friction and heating.

The arrival date at Mars would range from January 2021 to March 2021. The arrival date at Mars is constrained by many factors including the need for real-time data transmission from the spacecraft during the critical entry, descent, and landing operations so that fault reconstruction could be developed should a failure occur. This capability would be implemented most efficiently during the Mars 2020 mission via high data rate communication. A high-rate communication link would allow real-time transmission of all critical engineering data (e.g., spacecraft position and orientation, and confirmation of deployment sequences).

For the Mars 2020 mission, this could only be achieved by using a pre-positioned Mars orbiting spacecraft to relay transmissions from the Mars 2020 flight system to Earth. Currently available orbiting spacecraft for entry, descent, and landing (EDL) communications and surface operations relay include the Mars Reconnaissance Orbiter (MRO), which entered Mars orbit in March 2006 and Mars Odyssey which entered orbit in October 2001. In addition, two planned future missions would provide an opportunity for additional Mars-orbiting spacecraft before the Mars 2020 mission arrives at Mars. These missions—MAVEN, which launched in November of 2013 with a planned arrival at Mars in 2014 and the 2016 ExoMars Trace Gas Orbiter (part of a European Space Agency mission with NASA support) with a planned arrival at Mars in 2016—would insert spacecraft with communications capabilities able to support the Mars 2020 mission. NASA would coordinate among these four missions to identify which would provide the optimal high data rate communication relay spacecraft for the Mars 2020 arrival event and for subsequent rover surface operations. The constraints on launch dates and arrival conditions during the 20-day launch period, including mutual visibility at arrival among the orbiting spacecraft and the Mars 2020 spacecraft, would limit arrival to specific dates between January 2021 and March 2021.

Figure 2-2 shows the positions of Earth and Mars as they orbit the sun and the seasons for Mars. The range of Mars 2020 proposed arrival dates would coincide with the transition from winter to spring in the northern hemisphere of Mars.



Source: JPL 2013

Note: Earth and its orbit are in blue; Mars and its orbit in red

Figure 2-2. Arrival Dates for the Proposed Mars 2020 Mission

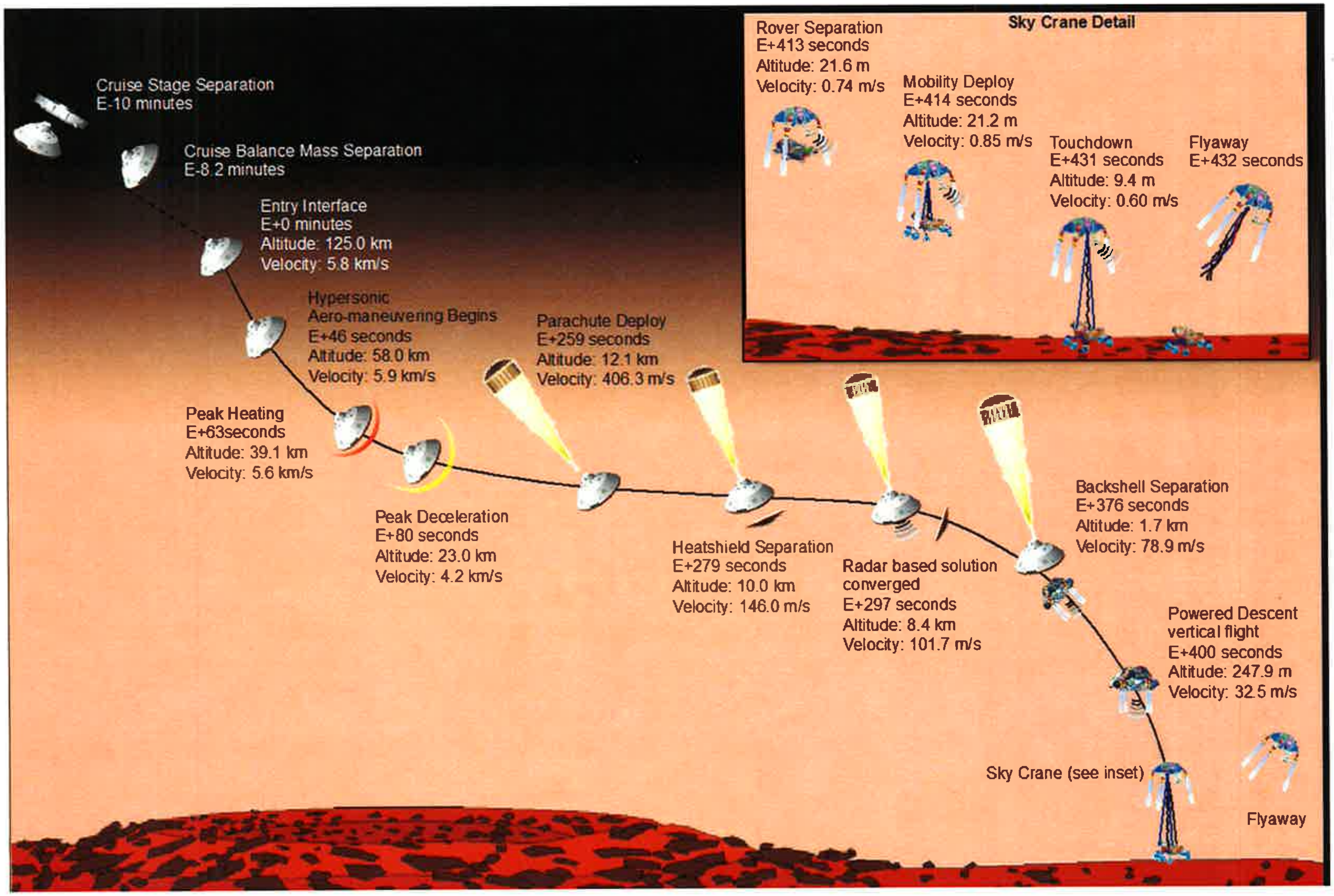
The exact landing site for the proposed Mars 2020 mission has not yet been selected. The location of the landing site would be restricted to between 30° north and 30° south latitudes as indicated in Table 2-1. It is anticipated that the landing site would be selected far enough in advance of the planned launch to allow sufficient time to determine the final details of the mission design (e.g., the specific launch trajectory). The site selection process would include a consensus recommendation by mission scientists, utilizing very detailed, high resolution images expected from the MRO mission and other available science data, on the most scientifically worthy location to land the rover. The selection process would also include NASA's engineering assessment of the rover's capabilities at the proposed site. NASA would then approve the selected site. The selected landing site would then factor into determination of the optimum launch and arrival dates for the mission, given the other constraints discussed above.

The EDL phase of the mission (Figure 2-3) would begin when the entry vehicle reaches an altitude of approximately 125 km (78 mi) above the surface of Mars, and would end with a soft touchdown of the rover on the Martian surface. The spacecraft would enter the Mars atmosphere directly from its interplanetary trajectory after a final trajectory correction maneuver and without entering orbit. The entry vehicle would maneuver during the early portion of atmospheric flight in order to reduce the landing site targeting errors that could result from pressure and density variations in the atmosphere.

Following parachute deployment at an altitude of about 12 km (7.5 mi), the heat shield would be released, the rover's mobility system deployed, and the landing radar initiated. The descent stage and rover would be released from the backshell about 1700 meters (m) (5,580 feet (ft)) above the surface and the terminal descent engines would be fired to slow the descending vehicle. At just over 20 m (66 ft) above the landing site, the rover would be lowered from the descent stage on tether/umbilical lines for a wheels-down soft landing on the Martian surface, called the "skycrane" phase of the landing sequence. The exact landing site is expected to be within a 25 km x 20 km (16 mi x 12 mi) elliptical area, although an improved EDL stage that would reduce the size of the landing area to an 18 km x 14 km (11 mi x 8.7 mi) elliptical area is being considered for the Mars 2020 mission. The tether/umbilical lines connecting the descent stage and the rover would be released, and the descent stage with the tether/umbilical lines attached would perform a fly-away maneuver to a hard landing a safe distance from the rover.

After landing on Mars, primary surface operations would commence and last for approximately one Martian year, which is 669 sols[4] or 687 Earth days. Under nominal initialization procedures, initial rover health checks would include calibration/checkout of the high gain antenna gimbal and the rover mast azimuth/elevation mechanism, removal of any engineering camera covers, and checkout of arm and mobility actuators. The rover would check the status of all major subsystems. Initial landed engineering camera and science instrument payload health checks would also occur during surface operations phase initialization, as well as a transition to the surface flight software load (i.e., a replacement of the onboard interplanetary cruise flight software with a flight software load tailored for the operation of a rover on the surface of a planet). A second phase of rover commissioning would include further checkout of mobility and arm functionality before the rover would be ready to start nominal science operations. In addition, first-time activities during nominal surface operations would require additional scrutiny. For example, first-time activities on the Mars Science Laboratory (MSL) Curiosity rover system included the first use of sample processing hardware and the first use of the corer. Mars 2020 would have comparable first-time activities to implement upon landing.

[4] 1 sol = 1 Martian day = 24 hours, 37 minutes = 1.026 Earth days.



Source: Mars 2020 Proposal Information Package (JPL)

Figure 2-3. Entry, Descent, and Landing Phase

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Surface operations are characterized by a long primary mission driven by an inherently interactive geological exploration and surveying process. The rover would have limited resources (power, mass storage, bandwidth, CPU, etc.) that require both flight- and ground-based management. The operations would be driven by a small set of repeating science scenarios. The general features of a typical surface operational scenario timeline can be divided into five main types of activities. This division is intended as an aid to understanding the Mars 2020 surface activities and scenario-dependent resource allocations, and is not intended to exclude any type of investigation that would be proposed through the Mars 2020 AO process. These scenarios are built of sol templates. Five different sol templates describe the expected building blocks of the surface mission operations plan: (1) traverse & approach, (2) site reconnaissance (remote sensing science), (3) arm manipulation & contact science, (4) coring/caching & contact science, and (5) recharge /telecom. The sol templates are used to help define resource usage for Mars 2020 planned activities to meet the mission and science objectives. The operations concept for the Mars 2020 mission, including team structures, uplink and downlink planning scenarios, daily operations timeline, and planned changes in operations approach over the course of the mission is derived from the experience and plans for Mars Exploration Rover (MER) and MSL flight operations.

Surface operations involve making decisions about how much time would be spent driving, how much time would be spent conducting fieldwork, and how much time would be spent collecting and caching samples. The amount of driving that might be required would depend greatly on where the rover has landed and where the highest-priority science targets might be located. *Fieldwork* is a term used here to encompass all of the effort expended to characterize the geology, assess habitability and preservation potential, identify possible biosignatures, and prepare any potential cores for caching. In particular for the Mars 2020 mission, fieldwork would include:

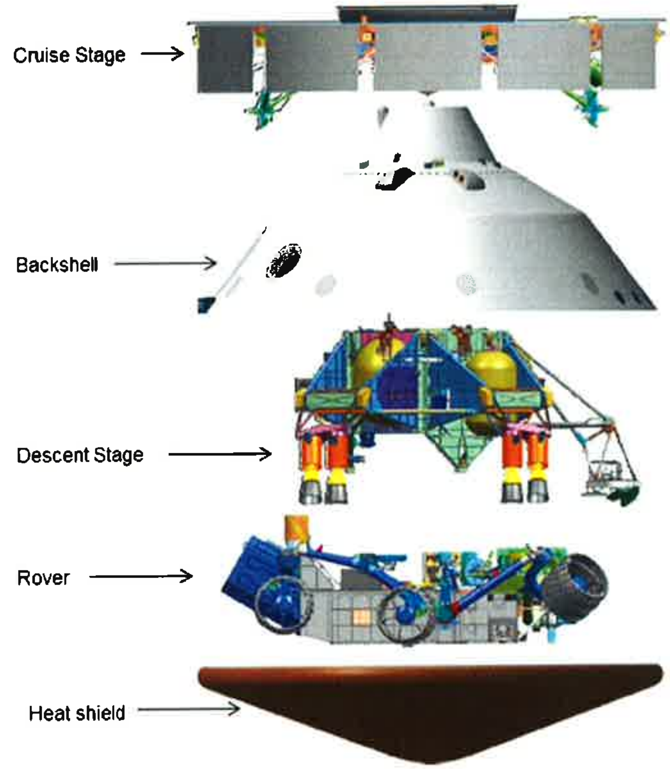
- acquisition and analysis of contextual imaging and mineralogy measurements,
- targeted contextual and fine-scale imaging and mineralogy observations,
- close-up elemental and organic detection measurements,
- preparation of rock surfaces by brushing and/or abrasion, and
- conduct of experiments in support of human exploration.

Fieldwork measurements would set the stage for selection of what to core, and which cores to cache for possible return to Earth. This effort would include the engineering interrogation of materials for their suitability to be cored. Decisions about the time spent on each of these activities would be governed by the strategic science objectives.

Scenarios for the rover's surface science operations are still being planned and evaluated by Mars 2020 mission scientists and engineers. The final details of the scenarios would depend upon factors such as the actual capabilities of the rover, when finally assembled and tested, and the selected landing site. Surface operations would also be adaptable to actual conditions on the surface of Mars and discoveries made during the course of the rover's mission. Best available information derived from the Mars 2020 AO documentation, from SDT mission objectives and SDT desired landing sites, are consistent with the mission operations scenarios of driving and fieldwork that have been used to estimate resource usage in order to accomplish surface mission objectives for the Mars 2020 mission. Resource usage models to accomplish these objectives are based upon a high-heritage flight system implementation, as discussed in Section 2.1.2, using representative instrumentation "stand-ins" and payload elements that would accomplish the desired measurements as sought by the currently underway Mars 2020 competitive procurement process.

2.1.2 Spacecraft Description

The Mars 2020 spacecraft flight system is based upon the successful MSL design and would consist of a cruise stage, an entry vehicle, a descent stage, and the science rover. The flight system, illustrated in Figure 2-4, is currently estimated to weigh up to 4,050 kilograms (kg) (8,930 pounds (lb)).



Source: JPL 2013

Figure 2-4. Illustration of the Proposed Mars 2020 Flight System

The cruise stage, approximately 4.4 m (14.4 ft) in diameter, would provide the services necessary to support the trip to Mars. These services would include communications with Earth and provision of electrical power to the entry vehicle via a 6.8 square meter (73.2 square feet) solar array. Attitude control and trajectory correction maneuvers would be performed via a spin-stabilized hydrazine propellant system. Two titanium propellant tanks would contain approximately 70 kg (154 lb) of hydrazine.

The entry vehicle, approximately 4.5 m (14.8 ft) in diameter, would contain the systems that would safely enter the Martian atmosphere and deliver the rover to its designated landing site. The entry vehicle would include a heat shield and backshell, a supersonic parachute deployed by a mortar, and the stowed descent stage and rover.

The descent stage, illustrated in Figure 2-5, would provide the systems needed to guide, decelerate, hover, and lower the rover onto its designated landing site. The descent stage would contain five propulsion system tanks; three hydrazine tanks made of titanium and two helium pressure vessels made of composite material. The total propellant load for the descent stage would be about 390 kg (860 lb) of hydrazine.

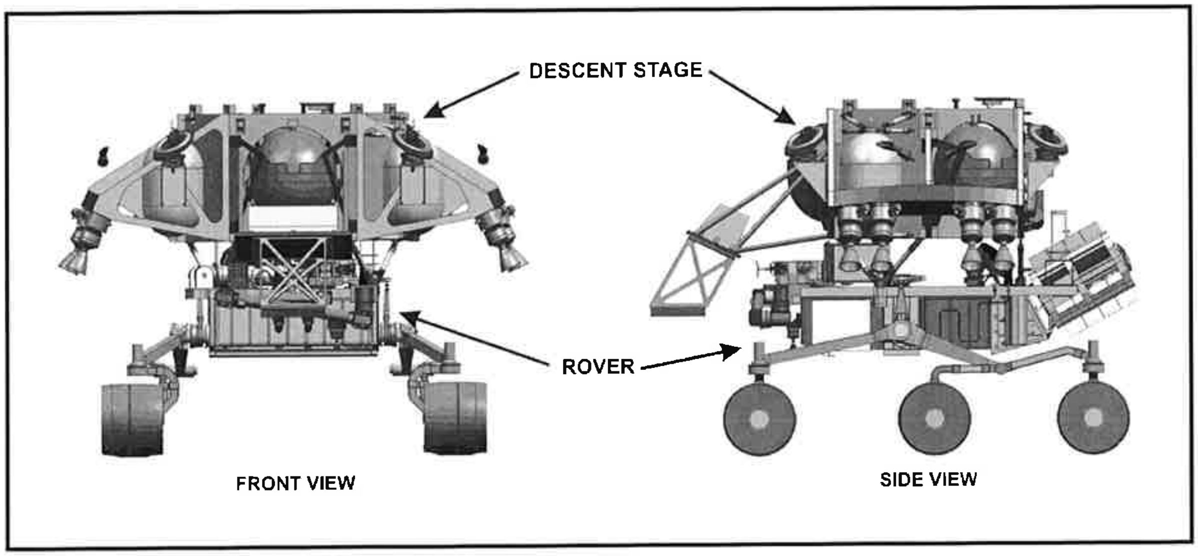


Figure 2-5. The Mars 2020 Descent Stage and Proposed Mars 2020 Rover

The preferred alternative rover, illustrated in Figure 2-6, would be made from an all-aluminum primary structure with machined panels. The thermal subsystem would include a heat exchange radiator system that allows use of the waste heat from the Multi-Mission Radioisotope Thermoelectric Generator (MMRTG) to keep the avionics and communication systems within thermal limits throughout Mars' daily and seasonal temperature variations. The mobility system would connect to the rover chassis. The rover would be designed to accommodate a payload module that would contain the body-mounted instruments and payload element, as well as the robotic arm. The rover would also support a remote sensing mast that would provide an elevated platform for critical engineering and scientific assets such as navigation imaging cameras, science imaging cameras, remote sensing instruments, and possibly meteorology instruments.

The payload instrumentation planned for the Mars 2020 mission would be selected by NASA through a competitive process (NASA 2013d) to meet the science objectives summarized in Chapter 1. The instrumentation solicited includes the science instrumentation used for investigating the surface of Mars (objectives A and B: to explore an astrobiologically relevant environment and to seek signs of life) and technology capabilities (objectives C and D: to make technical progress towards sample return and further preparation for human and robotic exploration). The selection of the instruments to be included on the Mars 2020 rover is the subject of a NASA Announcement of Opportunity published on September 24, 2013 to solicit proposals for the Mars 2020 surface-science investigations and exploration technology investigations. Following receipt and review of the proposals, NASA plans to select the suite of instruments in 2014. Pending the selection of the instruments for the Mars 2020 mission, the following discussion is based on the Mars 2020 mission SDT's assessment of the needs for the 2020 mission. The final selection of instruments would be based on the determination of the instruments that are best able to meet the goals of the Mars 2020 mission.

Source: JPL 2013

Figure 2-6. The Proposed Mars 2020 Rover

The SDT report identified two levels of scientific measurement for the Mars 2020 mission: a threshold level and a baseline level. The baseline level includes all of the measurements identified for the threshold level plus additional measurement capabilities and represents the capability to which the Mars 2020 science instrumentation would be designed. The types of measurements needed to meet the baseline science objectives for the Mars 2020 mission are summarized in Table 2-2, and the possible locations (the turret, mast, and internal rover volume) for the instruments on the rover are shown in Figure 2-7.

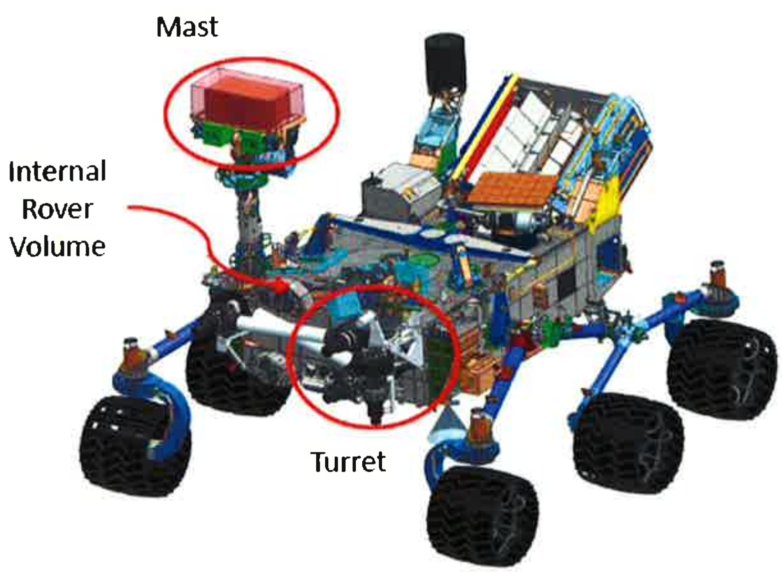
For objectives A, B, and C, five measurement types are threshold requirements to effectively and efficiently characterize the geology of a site, assess habitability, select samples, and document sample context.

Context Imaging. This measurement would image the terrain at a sufficient level of detail for navigational purposes (enabling the rover to travel at the required minimum distances per day), to characterize the geological context, to select (at a distance) locations for further in-depth analyses by close-up instruments and sampling, and to identify terrain that could support the assessment of past habitable environments and the potential for preservation of signs of life.

Table 2-2. Mars 2020 Science Measurements

Measurement	Objective A	Objective B	Objective C	Objective D
Context Imaging	√	√	√	
Fine-Scale Imaging	√	√	√	
Context Mineralogy	√	√	√	
Fine-Scale Elementary Chemistry	√	√	√	
Fine-Scale Mineralogy	√	√	√	
Subsurface Sensing	†			
Reduced/Organic Matter Detection		√		
Organic Matter Detection	†	†	†	
In Situ Resource Utilization				†
Entry, Descent, and Landing Data				†
Entry Descent, and Landing Precision				†
Surface Weather Monitoring				†
Biohazards to Astronauts				†

√ - Threshold
† - Baseline
Note: The total mass allocation for the science instruments is currently 28kg (62 lb) (NASA 2013d)



Source: JPL 2013

Figure 2-7. The Science Instrument Locations on the Proposed Mars 2020 Rover

Context Mineralogy. This measurement would serve a dual role in supplying reconnaissance information for possible drive targets and provide context for fine-scale measurements. Context mineralogy would identify, from afar, the presence of key mineral phases in surface targets to support the selection of specific outcrops, rocks, and soils to investigate in detail with other rover instrumentation, especially with respect to identifying potential areas that show signs of past habitable environments and the preservation of signs of life.

Fine-scale Imaging. The objectives of this measurement would be to characterize grain form and structure and the textural fabric of rocks and soils at a microscopic scale. Data from this investigation would: 1) contribute to the characterization of the rover site's geological environment; 2) illuminate details of local geologic history, such as crystallization of igneous rocks, deposition and conversion of sediment to rock, and weathering and erosion; and 3) assist in the search for structural signs of life, if preserved, in the rock record.

Fine-scale Mineralogy. The objectives of this investigation would be to detect and to measure the spatial distribution, at sub-millimeter scale, of the signatures of key minerals in outcrops, rocks, and soils. For objective B, a key purpose of the mineralogical measurement would be to detect potential biominerals and determine the mineral composition of other potential biosignatures and associated materials.

Fine-scale Elemental Chemistry. The objective of this investigation would be to measure the abundances of major and selected minor elements most indicative of igneous, alteration, and sedimentary processes. Among the science goals of these measurements would be to determine the fine-scale elemental chemistry of sedimentary, igneous and alteration features, and (for objective B) to detect potential chemical signs of life, determine the elemental composition of potential signs of life, and search for historical evidence of the activity of liquid water.

In addition to the five threshold investigations described above, baseline investigations would include organic detection investigation; both to provide contextual information on habitability and potential signs of life and to select, if possible, samples with preserved organic chemistry.

Organic Matter Detection. Organic matter detection would provide observations for assessing the processes that influence preservation of information about ancient environments. Detection of organic matter, via the identification of reduced carbon compounds in near-surface materials, could be used to help characterize meteoritic inputs, hydrothermal processes, atmospheric processes, and other potential processes that might form organic matter. Lastly, in order to identify the most desirable samples for possible return to Earth, detecting organic matter at a site would be valuable. The vast majority of spaceflight-compatible methods for detecting organic matter that might include potential organic signs of life can be categorized as types of mass spectrometry, chromatography, spectrophotometry, and binding assays or metabolic assays.

Subsurface Sensing. Techniques that sense subsurface structural continuity could provide contextual information complementary to that obtained by the envisaged threshold payload for surface exposures. Ground-penetrating radar and electromagnetic sounding are examples of relevant techniques that could provide information to better understand local stratigraphy.

Five demonstration payloads have been identified that meet the needs for Objective D; but these may not be the full set of demonstration payloads ultimately considered for the Mars 2020 mission. The first demonstration payload would be the demonstration of carbon dioxide (CO_2) capture and dust size characterization for atmospheric In Situ Resource Utilization (ISRU). This payload addresses two-high priority items:

demonstrating atmospheric ISRU and measuring dust properties. It would be an architecture enabling technology for human missions to Mars, which will likely depend on ISRU for producing the propellants needed for the return trip to Earth; ISRU can greatly reduce mass transported to the Martian surface. ISRU would demonstrate dust filtration and non-intrusive measurement during Mars CO_2 capture and subsequent CO_2 collection.

The second demonstration payload would be a flight of an enhanced EDL instrumentation payload to acquire temperature and pressure measurements on the heat shield and other parts of the spacecraft. The temperature and pressure measurements during atmospheric entry would be used to validate analytical models for designing future EDL systems. EDL systems capable of landing large payloads on Mars are an architecture enabling technology for human missions.

Another possible EDL technology demonstration would include technologies to improve EDL precision (reduce the size of the potential landing area or better ensure landing survival). Potential technologies include: a Range Trigger, improved technology for deployment of the parachute based on range to the landing site; Terrain Relative Navigation, navigation by matching visual images of the landing site taken during descent to images taken from orbit; and terminal hazard avoidance systems, a combination of landing site hazard identification and terminal guidance technologies.

The inclusion of a Surface Weather Station on the Mars 2020 payload would provide density for EDL and ascent profiles, plus validation data for global atmosphere models that would enable validation of global model extrapolations of surface pressure. It would also provide local-surface and near-surface validation data to validate regional and local model atmospheric conditions. Parameters monitored could include pressure, temperature, winds, humidity, and vertical temperature profiles. Additionally, total atmospheric aerosol content and aerosol profiles could be monitored. This set of instrumentation, plus the characterization of the dust properties provided as part of the ISRU demonstration, would address a number of climatological science questions and objectives.

A biohazards to astronauts technology would consist of a "biomarker detector" system which could not only examine the potential for contaminants to impact astronauts (and other species should the contaminant be returned to Earth), but could also assess the impact of terrestrial contaminants on Mars. Such a system could be used for extraterrestrial life detection by targeting universal biomarkers such as amino acids, polymers, polysaccharides, whole cells, and microbial spores; and also for planetary protection to monitor forward contamination during robotic/human operations in an extraterrestrial environment.

While the science instruments for the Mars 2020 rover are yet to be selected, it is reasonable to assume that at least some of them may contain small radioactive sources. These sources are typically used for calibration of the science instrument, or they could be a necessary part of the instruments investigative process. For example, the Mars Science Laboratory rover and the Mars Exploration Rovers (MERs) contained science instruments that contained radioactive sources used for instrument calibration or science experiments. The isotope and quantity of each source is listed below.

Mission	Instrument	Radioisotope	Activity, curies
MER	APXS	Curium-244	0.03
MER	Mossbauer Spectrometer	Cobalt-57	<0.35
MSL	APXS	Curium-244 Cadmium-109	0.06 0.105
MSL	DAN	Tritium (hydrogen-3)	2

Definitions: APXS – Alpha Particle X-ray Spectrometer; DAN – Dynamic Albedo of Neutrons

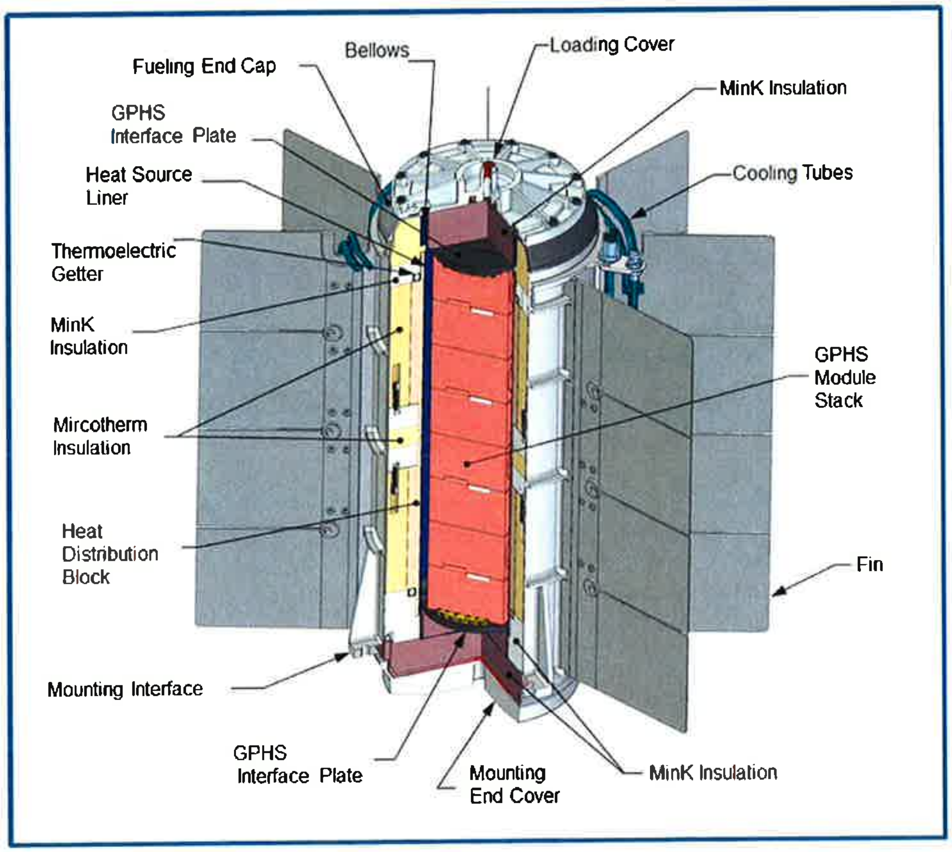
2.1.3 Rover Electrical Power

The proposed Mars 2020 rover would use a Multi-Mission Radioisotope Thermoelectric Generator (MMRTG), provided to NASA by the U.S. Department of Energy (DOE), as the source of electrical power for its engineering subsystems and science payload. This is the same power supply used by the MSL. The MMRTG would be the only radioisotope thermoelectric generator available for the Mars 2020 mission. NASA has pursued the development of both the MMRTG and an Advanced Stirling Radioisotope Generator (ASRG) (NASA 2006b). However, NASA recently announced that it has decided to end procurement of and discontinue work on the development of ASRG flight hardware. Therefore, an ASRG would not be available for the Mars 2020 mission.

An MMRTG (Figure 2-8) converts heat from the natural radioactive decay of plutonium (in a ceramic form called plutonium dioxide consisting mostly of plutonium-238) into usable electrical power. RTGs have been successfully used on 27 previously-flown United States space missions (Table 2-3), including six Apollo flights, and the Pioneer, Viking, Voyager, Galileo, Ulysses, Cassini, and New Horizons missions. The evolutionary development of radioisotope power systems has resulted in several RTG configurations, evolving from the Systems for Nuclear Auxiliary Power (SNAP)-3 RTG through the Multi-Hundred Watt (MHW)-RTG to the General Purpose Heat Source (GPHS)-RTG used for the New Horizons mission to Pluto. The MMRTG is designed for applications both in the vacuum of deep space and on the surface of bodies with an atmosphere, such as Mars.

Development of the MMRTG has been documented in NASA's Final Programmatic Environmental Impact Statement for the Development of Advanced Radioisotope Power Systems (NASA 2006b).

The heat source assembly of the MMRTG consists of eight GPHS modules, an isolation liner, and end components. Each GPHS module (Figure 2-9) has dimensions of approximately 9.3 by 10.0 by 5.8 centimeters (cm) (3.7 by 3.9 by 2.3 inches (in)), a mass of about 1.6 kg (3.5 lb), and would contain about 0.6 kg (1.3 lb) of plutonium dioxide (SNL 2014). A GPHS module consists of a graphite aeroshell, two carbon-bonded carbon fiber insulator sleeves, two graphite impact shells (GIS), and four iridium clads, with each clad containing a ceramic pellet of plutonium dioxide.



Source: SNL 2014

Figure 2-8. Components of a Multi-Mission Radioisotope Thermoelectric Generator

An MMRTG contains about 4.8 kg (10.6 lb) of plutonium dioxide with a total radiological activity of about 60,000 curies (Ci). Plutonium can exist in a number of different radioactive isotopic forms. The principal plutonium isotope in the fuel, in terms of mass and total activity, is Pu-238. Table 2-4 provides representative characteristics and the isotopic composition of the plutonium dioxide in the MMRTG (SNL 2014). Plutonium dioxide has a density of 9.6 grams per cubic centimeter (5.5 ounces per cubic inch), melts at 2,400 degrees Celsius (°C) (4,352 degrees Fahrenheit (°F)), and boils at 3,870°C (6,998°F).

The DOE designed the MMRTG to provide for containment of the plutonium dioxide fuel to the extent feasible during all mission phases, including ground handling, launch, and unplanned events such as reentry, impact, and post-impact situations including fires. Under normal, accident, and post-accident conditions the safety-related design features of the MMRTG to be used for the Mars 2020 mission are intended to:

- Prevent, to the extent possible, the release of plutonium dioxide from the iridium clad and GPHS
- minimize the release and dispersion of the plutonium dioxide fuel, especially small, respirable particles that could be hazardous to human health

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Table 2-3. U.S. Space Missions Using Radioisotope Power Sources (RPSs)

Power Source (number of RPSs)	Spacecraft	Mission Type	Launch Date	Status	Activity at Launch (curies)
SNAP-3B7 (1)	TRANSIT 4A	Navigational	Jun 29, 1961	Currently in Earth orbit	1,500 – 1,600
SNAP-3B8 (1)	TRANSIT 4B	Navigational	Nov 15, 1961	Currently in Earth orbit	1,500 – 1,600
SNAP-9A (1)	TRANSIT 5BN-1	Navigational	Sep 28, 1963	Currently in Earth orbit	17,000
SNAP-9A (1)	TRANSIT 5BN-2	Navigational	Dec 5, 1963	Currently in Earth orbit	17,000
SNAP-9A (1)	TRANSIT 5BN-3	Navigational	Apr 21, 1964	Mission aborted; RPS burned up on reentry as designed	17,000
SNAP-19B2 (2)	NIMBUS-B-1	Meteorological	May 18, 1968	Mission aborted; RPS retrieved intact	34,400
SNAP-19B2 (2)	NIMBUS III	Meteorological	Apr 14, 1969	Currently in Earth orbit	37,000
SNAP-27 (1)	APOLLO 12	Lunar	Nov 14, 1969	ALSEP [a] shut down and remains on lunar surface	44,500
SNAP-27 (1)	APOLLO 13	Lunar	Apr 11, 1970	Mission aborted on way to moon; ALSEP (in Lunar Module) was successfully targeted to the Tonga Trench in the Pacific Ocean for safe disposal	44,500
SNAP-27 (1)	APOLLO 14	Lunar	Jan 31, 1971	ALSEP shut down and remains on lunar surface	44,500
SNAP-27 (1)	APOLLO 15	Lunar	Jul 26, 1971	ALSEP shut down and remains on lunar surface	44,500
SNAP-19 (4)	PIONEER 10	Planetary	Mar 2, 1972	Successfully operated to Jupiter and beyond	80,000
SNAP-27 (1)	APOLLO 16	Lunar	Apr 16, 1972	ALSEP shut down and remains on lunar surface	44,500
TRANSIT-RTG (1)	TRIAD-01-1X	Navigational	Sep 2, 1972	Currently in Earth orbit	24,000
SNAP-27 (1)	APOLLO 17	Lunar	Dec 7, 1972	ALSEP shut down and remains on lunar surface	44,500
SNAP-19 (4)	PIONEER 11	Planetary	Apr 5, 1973	Successfully operated to Jupiter, Saturn and beyond	80,000
SNAP-19 (2)	VIKING 1	Planetary	Aug 20, 1975	Lander shut down and remains on surface of Mars	41,000
SNAP-19 (2)	VIKING 2	Planetary	Sep 9, 1975	Lander shut down and remains on surface of Mars	41,000
MHW-RTG (2)	LES 8	Communications	Mar 14, 1976	Successfully operating in Earth orbit	159,400
MHW-RTG (2)	LES 9	Communications	Mar 14, 1976	Successfully operating in Earth orbit	159,400
MHW-RTG (3)	VOYAGER 2	Planetary	Aug 20, 1977	Successfully operated to Neptune and beyond	240,000
MHW-RTG (3)	VOYAGER 1	Planetary	Sep 5, 1977	Successfully operated to Saturn and beyond	240,000
GPHS-RTG (2)	GALILEO	Planetary	Oct 18, 1989	Successfully operated in Jupiter orbit; after 8 years, spacecraft purposefully entered Jupiter's atmosphere	269,000 [b]
GPHS-RTG (1)	ULYSSES	Planetary	Oct 6, 1990	Successfully operating in heliocentric orbit	132,500
GPHS-RTG (3)	CASSINI	Planetary	Oct 15, 1997	Successfully operating in Saturn orbit	404,000 [b]
GPHS-RTG (1)	NEW HORIZONS	Planetary	Jan 19, 2006	Successfully operating in flight to Pluto	121,000
MMRTG (1)	MSL	Planetary	Nov 26, 2011	Successfully operating on the surface of Mars	58,700

(a) Apollo Lunar Surface Experiments Package.

(b) Includes inventory from Radioisotope Heater Units.

Note: The proposed Mars 2020 mission would use one MMRTG with approximately 60,000 curies.

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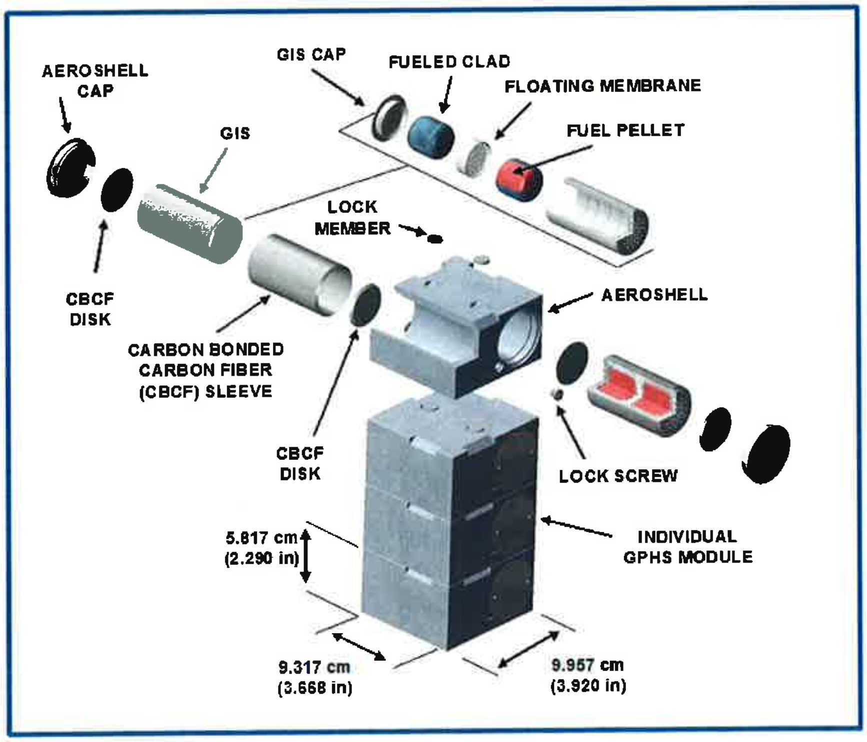


Source: SNL 2014

Figure 2-9. A General Purpose Heat Source Module

- reduce the likelihood that small respirable sized particles could be generated during an accident
- minimize any land, ocean and atmosphere contamination, particularly in populated areas; and,
- maximize the long-term immobilization of the plutonium dioxide fuel following postulated accidents so that it does not spread further and could be more effectively recovered.

The layered approach to the safety design features of the MMRTG and their response to potential accidents include the following elements.

- **Thermoelectric Converter/GPHS Design:** The MMRTG is designed to release the individual GPHS modules in case of inadvertent reentry into Earth's atmosphere after launch in order to minimize the terminal velocity of the modules and the potential for fuel release on Earth impact. The converter housing is made of aluminum alloy to ensure melting and breakup of the converter upon reentry, resulting in release of the modules.
- **GPHS Module, GIS, and related graphite components:** The GPHS module and its graphite components are designed to provide reentry and surface impact protection to the iridium fueled clad in case of accidental sub-orbital or orbital reentry. The aeroshell and GIS are composed of a rugged carbon-carbon Fine Weave Pierced Fabric, developed originally for reentry nose cone material. The

Table 2-4. Typical Isotopic Composition of an MMRTG

Fuel Component	Weight Percent	Half-Life, years	Specific Activity, curies/gram	Total Activity [a], curies
Plutonium (Pu)	85.99			
Pu–236	6×10^{-8}	2.851	531.3	0.0016
Pu–238	72.33	87.7	17.12	59,440
Pu–239	11.83	24,131	0.0620	17.6
Pu–240	1.70	6,569	0.2267	18.5
Pu–241	0.09	14.1	103.0	445
Pu–242	0.04	375,800	0.00393	0.0080
Actinide Impurities	0.97	NA	NA	24
Other Impurities	1.14	NA	NA	NA
Oxygen	11.9	NA	NA	NA
Total	**100.00**	**NA**	**NA**	**59,936**

Source: SNL 2014

(a) Based on 4.8 kg (10.6 lb) of PuO_2.
NA = Not Applicable

existing GPHS module is an evolution of a design that has worked with extreme reliability for the past three decades; to provide even greater protection, the broad face of the module and the face between the two shells are 20 percent thicker than the modules used in the GPHS-RTG in order to increase the module's strength and enhance its performance under impact and reentry conditions (SNL 2014).

- **Iridium Clads:** The iridium that encases each plutonium dioxide pellet is a strong, ductile metal that resists corrosion and does not react chemically with the radioisotope fuel. In the unlikely event of an accident involving an impact, the iridium cladding is designed to deform yet contain the fuel. Iridium is chemically compatible with the graphite components of the GPHS module and the plutonium dioxide fuel over the operating temperature range of the MMRTG, given its high melting temperature (2,443°C (4,430°F)) and excellent impact response.
- **Ceramic Form of Plutonium Dioxide:** The nuclear fuel used in an MMRTG is manufactured in a ceramic form. This form has material properties similar to a coffee cup: it tends to fracture in large, non-inhalable chunks and it is highly insoluble; this means that it does not easily mix or become easily transportable in water, nor does it react easily with other chemicals. Plutonium dioxide has a high melting temperature (2,400°C (4,352°F)).

DOE has over 30 years of experience in the engineering, fabrication, safety testing, and evaluation of GPHS modules, building on the experience gained from previous heat source development programs and an information base that has grown since the 1960s.

The GPHS modules were designed to prevent the release of fuel under a wide variety of accident scenarios, including high-speed impacts, projectiles, fires, and Earth re-entry.

Previous generations of heat source designs have survived two accidents: the heat sources on the Nimbus-B spacecraft (1968) protected the fuel from release during an early launch abort (with the fuel subsequently being re-used on a future mission), and the Apollo 13 lunar module (1970) carried a lunar surface science experiment package heat source that was similarly protected during its re-entry and ocean impact.

The MMRTG and enhanced GPHS module were successfully flown on the MSL mission that launched in November 2011 and is now operating as designed on Mars. Even though formal safety testing is ongoing, much insight has been gained by examining the safety testing performed on the earlier GPHS-RTG and its components. The GPHS-RTG with 18 GPHS modules has been used on the Galileo, Ulysses, Cassini, and New Horizons missions. Formal safety testing of both the MMRTG and GPHS-RTG components has established a database that allows prediction of responses in accident environments. These safety tests have covered responses to the following environments:

- impact from fragments,
- other mechanical impacts,
- thermal energy,
- explosive overpressure, and
- reentry conditions (i.e., aerodynamic loads and aerodynamic heating).

2.1.4 Operational Considerations

An MMRTG supplies sufficient power for the rover to perform operations at all times and at all possible landing sites between 30° north and 30° south latitudes. At no time would the rover be required to operate at less than 100% capability (constrained capacity), nor would it have to hibernate (cease all operations but maintain the rover temperature within limits needed to assure rover survival).

2.1.5 Spacecraft Processing

The Mars 2020 spacecraft would be designed, fabricated, integrated and tested at facilities of the spacecraft provider, the Jet Propulsion Laboratory (JPL), which is managed for NASA by the California Institute of Technology in Pasadena, CA. These facilities have been used extensively in the past for a broad variety of spacecraft, and no new facilities would be required for the Mars 2020 spacecraft. JPL would deliver the spacecraft to NASA's Kennedy Space Center (KSC) in Florida for further testing and integration with the MMRTG and with the launch vehicle.

The spacecraft would be received at the KSC Payload Hazardous Servicing Facility (PHSF). The spacecraft would be inspected and comprehensive tests would be performed, including flight and mission simulations. The DOE would deliver the MMRTG to a KSC storage facility. Once the spacecraft tests are completed, the MMRTG would be moved to the PHSF where it would be fitted to the rover for a pre-flight systems check. After completing these checks, the MMRTG would be returned to storage. The spacecraft would then be fueled with a total of about 460 kg (1,014 lb) of hydrazine (SNL 2013), the currently estimated propellant load capability for the cruise stage and descent stage.

A systems check and other tests would then be performed, after which the spacecraft would be enclosed within the launch vehicle payload fairing (PLF). The PLF, containing the spacecraft, would then be transported from the PHSF to the launch complex at KSC or CCAFS and would be attached to the vehicle's second stage. The aft end of the PLF would be sealed with a barrier and connected to an environmental control system to prevent contamination during transit.

After the Mars 2020 spacecraft and its launch vehicle have been integrated at KSC or CCAFS, the MMRTG would be transported to the launch complex where it would be installed on the rover through special access panels on both the launch vehicle PLF and the entry vehicle aeroshell (Lytal 2010). MMRTG handling at KSC and CCAFS would be performed under stringent conditions following all requirements governing the use of radioactive materials. Transportation of the MMRTG between KSC and CCAFS would be in accordance with applicable U.S. Department of Transportation and other federal, state, and local regulations (NASA 2001).

2.1.6 Representative Launch Vehicle Configurations for the Mars 2020 Mission

Early in the development process for the proposed Mars 2020 mission, NASA plans to issue a Request for Launch Service Proposal to all NASA Launch Service (NLS)-approved contractors. The Request for Launch Service Proposal would contain a statement of work and request that proposals be submitted to NASA for the Mars 2020 mission. Once the proposals are received from the NLS contractors, NASA's Launch Service Task Order (LSTO) board would evaluate them in accordance with LSTO procedures and previously determined technical evaluation criteria. Upon completion of the evaluation, NASA would identify the proposed configuration of the launch vehicle that would meet all the specified mission requirements and would present the best value to the government.

The evaluations of potential environmental consequences for this DEIS, summarized in Section 2.5 and presented in more detail in Chapter 4, were prepared before NASA selected the launch vehicle for the proposed Mars 2020 mission. These evaluations were based upon representative configurations of the Atlas V and Delta IV class vehicles (the Delta IV class vehicle representing the liquid fueled Delta IV and Falcon Heavy launch vehicles) that would have the performance capabilities necessary for the mission. The representative launch vehicle configurations are described in the following sections.

2.1.6.1. Description of the Atlas V Launch Vehicle

The Atlas family of launch vehicles, provided by United Launch Alliance (ULA) a joint venture of Lockheed Martin Corporation and The Boeing Company (a NLS-approved contractor), has evolved through various government and commercial programs from the first research and development flight in 1957 through the Atlas II, III, and V configurations. Versions of Atlas vehicles have been built specifically for both robotic and human space missions. The most recent version, the Atlas V, is currently available in 400 and 500 series configurations.

The Atlas V configurations being considered for the proposed Mars 2020 mission are the Atlas V 541 and 551, each of which would consist of a liquid propellant first stage with strap-on solid rocket boosters (SRBs), a liquid propellant Centaur second stage, the Mars 2020 spacecraft, and the PLF. The "541" designation denotes a 5-m PLF, four SRBs, and a single-engine Centaur second stage; the "551" has five SRBs. The SRBs are attached to the first stage and the Centaur is mounted on top of the first stage. The Mars 2020 spacecraft would be mounted atop the Centaur. The PLF encloses and protects the spacecraft. The Atlas V, depicted in Figure 2-10, is approximately 62.4 m (205 ft) in height (ULA 2010).

2.1.6.1.1. First Stage

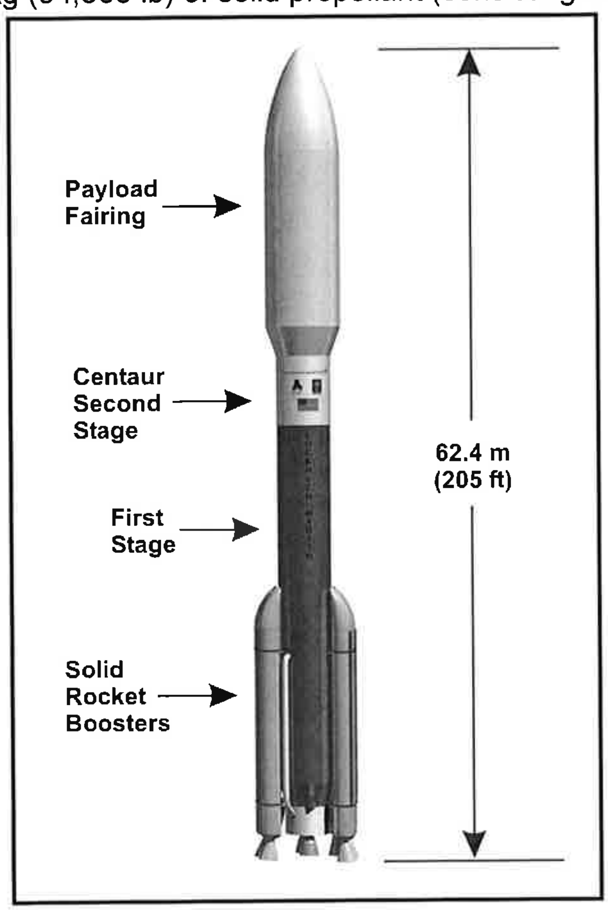
The Atlas V first stage is constructed mostly of aluminum and composite material, and is about 3.8 m (12.5 ft) in diameter and about 32.5 m (107 ft) in length. The first stage is powered by a liquid-fueled engine and contains about 284,089 kg (626,303 lb) of propellant. The fuel is rocket propellant-1 (RP-1), a thermally stable kerosene; and the oxidizer is liquid oxygen (LOx). Each SRB is 1.5 m (5 ft) in diameter, 20 m (66 ft) in length, and is fueled with about 43,000 kg (94,800 lb) of solid propellant (consisting of ammonium perchlorate, aluminum, and hydroxyl-terminated polybutadiene (HTPB) binder) for a total propellant mass of about 172,000 kg (379,000 lb) for the four SRBs, and about 215,000 kg (474,000 lb) for five SRBs (ULA 2010).

2.1.6.1.2. Centaur Second Stage

The Atlas V Centaur second stage is constructed of stainless steel and is about 3.1 m (10 ft) in diameter and about 12.7 m (42 ft) in length. The Centaur is powered by a single, cryogenic engine, and contains about 20,830 kg (45,922 lb) of propellant, consisting of liquid hydrogen (LH_2) as the fuel and LOx as the oxidizer (ULA 2010). The Centaur uses less than 91 kg (200 lb) of hydrazine for reaction control (USAF 2000).

2.1.6.1.3. Payload Fairing

The PLF for the Atlas V is about 5.4 m (18 ft) in diameter and about 20.7 m (68 ft) in length and is constructed of aluminum, carbon



Source: Adapted from, ULA 2010

Figure 2-10. An Atlas V Launch Vehicle with Solid Rocket Boosters

fiber, and composite materials. The PLF encloses and protects the spacecraft from thermal, acoustic, electromagnetic, and environmental conditions during ground operations and lift-off through atmospheric ascent (ULA 2010). Figure 2-11 depicts the spacecraft within the PLF envelope.

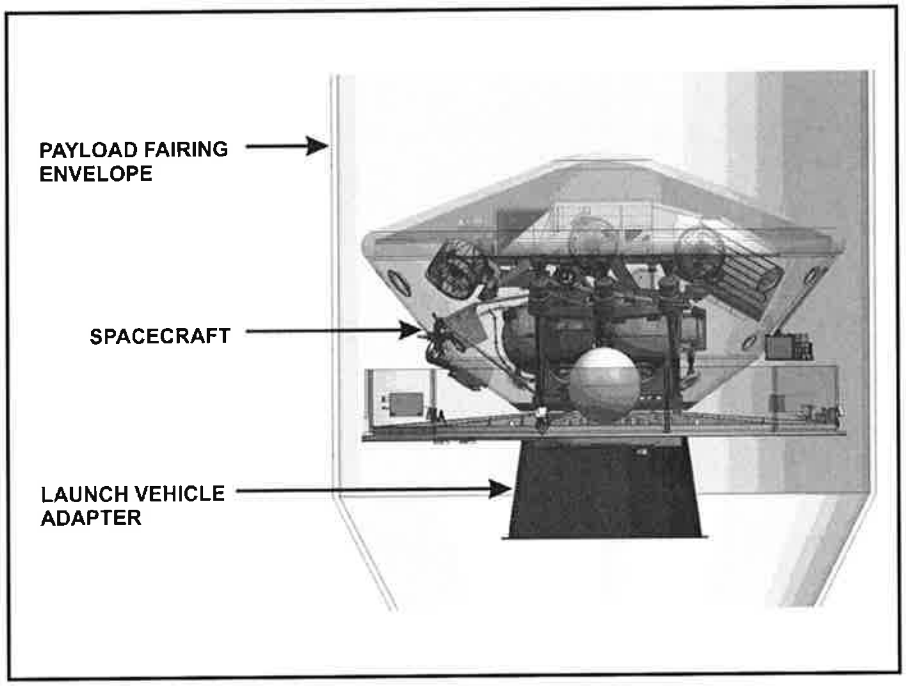


Figure 2-11. The Spacecraft Within the Payload Fairing Envelope

2.1.6.1.4. Atlas V Space Launch Complex-41

Space Launch Complex (SLC)-41 is located in the northernmost section of CCAFS. The launch complex consists of a launch pad, an umbilical mast, propellant and water storage areas, an exhaust flume, catch basins, security services, fences, support buildings, and facilities necessary to prepare, service, and launch Atlas V vehicles (USAF 1998, ULA 2010).

Security at SLC-41 is ensured by a perimeter fence, guards, and restricted access. Since all operations in the launch complex would involve or would be conducted in the vicinity of liquid or solid propellants and explosive devices, the number of personnel permitted in the area, safety clothing to be worn, the type of activity permitted, and equipment allowed would be strictly regulated. The airspace over the launch complex would be restricted at the time of launch.

2.1.6.1.5. Launch Vehicle Processing

Atlas launch vehicle preparation activities and procedures during and after launch have been previously documented (USAF 1998, ULA 2010). All NASA launches follow the current standard operating procedures.

The Atlas V launch vehicle components for the Mars 2020 mission would be received at CCAFS, where they would be inspected, stored, and processed at appropriate facilities.

When needed for launch, the components would be moved to the Vertical Integration Facility (VIF) at SLC-41, where the launch vehicle would be assembled, integrated, and tested. The PLF, containing the Mars 2020 spacecraft, would then be transported from the PHSF at KSC to the VIF and mated to the Centaur second stage. The Atlas V launch vehicle would then be moved via rail on a mobile launch platform (limited to a speed of 3.2 km/h (2 mph)) to the launch pad at SLC-41 for a rehearsal of loading the RP-1, LOx, and LH_2 liquid propellants, and then unloading the LOx and LH_2. The vehicle (with RP-1) would then be moved back to the VIF, where hydrazine would be loaded and final vehicle processing would be performed. The MMRTG would then be installed on the spacecraft. The launch vehicle would then be moved back to the pad for LOx and LH_2 loading, final systems tests, and launch (USAF 1998, USAF 2000, ULA 2010).

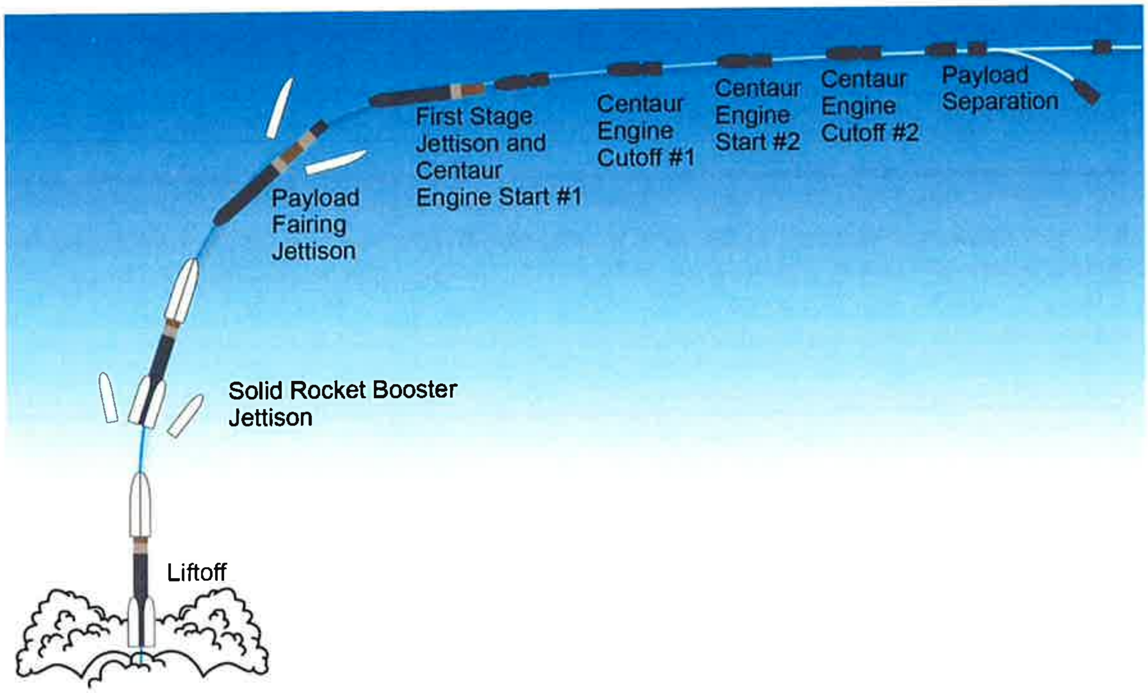
Processing activities for the Mars 2020 Atlas V vehicle would be similar to those routinely practiced for other Atlas launches from CCAFS. Effluents and solid or hazardous wastes that may be generated by these activities are subject to federal and state laws and regulations. NASA or its contractors would dispose of hazardous wastes. CCAFS has the necessary environmental permits and procedures for conducting launch vehicle processing activities (see Section 4.10).

2.1.6.1.6. Launch Profile

Launch of the Atlas V would begin with the ignition of the first stage main engine followed approximately 3 seconds[5] later by ignition of the four SRBs (Figure 2-12). The SRB casings would be jettisoned after propellant burnout. The first stage main engine would continue to thrust and the PLF would be jettisoned. The main engine cutoff sequence would be initiated when low propellant levels are detected by the first stage propellant sensors (ULA 2010). The first stage would then separate from the second stage. The SRB casings, the PLF, and the first stage would fall into the Atlantic Ocean in predetermined drop zones and would not be recovered (USAF 2000).

The Centaur second stage would be ignited shortly after separation from the first stage. Upon achieving Earth parking orbit, the Centaur engine thrust would be cut off via a timed command. After a brief, predetermined coast period in an Earth parking orbit, the Centaur engine would restart and the vehicle would accelerate to Earth escape velocity. After Centaur engine cutoff, the Mars 2020 spacecraft would separate from the Centaur and continue on its trajectory to Mars. The Centaur would continue separately into interplanetary space.

[5] The engine undergoes an automatic "health check" during this period. Should a malfunction be detected, the engine would be shut down and the launch would be aborted.



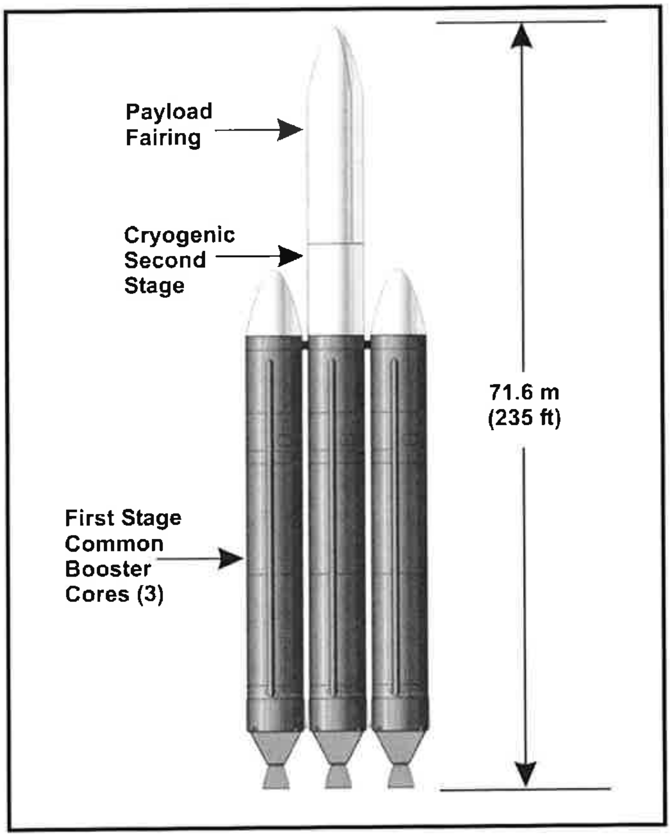
Source: Adapted from ULA 2010

Figure 2-12. Typical Atlas V Ascent Profile

2.1.6.2. Description of the Delta IV Heavy Launch Vehicle

The Delta launch vehicle program was initiated in the late 1950s by NASA with Douglas Aircraft (which then became McDonnell Douglas, which became part of The Boeing Company) and is now provided by ULA, a NLS-approved contractor. The Delta IV launch system, evolved from the Delta II and Delta III launch systems, is the latest generation in this nearly 50-year evolution. The Delta IV is currently available in Medium, Medium+, and Heavy configurations.

The representative Delta IV configuration for the proposed Mars 2020 mission is the Delta IV Heavy, which would consist of a liquid propellant first stage (called the common booster core (CBC)), two strap-on CBCs, a liquid propellant second stage, the Mars 2020 spacecraft, and a 5-m PLF. The additional CBCs are attached to the first stage, and the second stage is mounted atop the first stage. The Mars 2020 spacecraft would be mounted atop the second stage. The PLF encloses and protects the spacecraft. The Delta IV Heavy, depicted in Figure 2-13, is approximately 71.6 m (235 ft) in height (ULA 2013, ULA 2013).



Source: Adapted from ULA 2013

Figure 2-13. A Delta IV Heavy Launch Vehicle

2.1.6.2.1. First Stage

The Delta IV Heavy first stage CBCs are constructed mostly of aluminum and composite material. Each CBC is about 5 m (16.4 ft) in diameter and about 39.6 m (130 ft) in length. The CBCs are each powered by a cryogenic engine and each contains about 202,100 kg (445,600 lb) of propellant consisting of LH_2 as the fuel and LOx as the oxidizer for a total first stage propellant load of 606,300 kg (1,336,650 lb). A cylindrical interstage that encloses the second stage is mounted on the central CBC. Aerodynamic nosecones are mounted on the two strap-on CBCs in place of the interstage (ULA 2013, Freeman 2006).

2.1.6.2.2. Second Stage

The Delta IV second stage, constructed of aluminum and composite material, is about 5 m (16.4 ft) in diameter and about 13 m (42.7 ft) in length. The stage is powered by a single cryogenic engine and contains about 27,200 kg (60,000 lb) of propellant, consisting of LH_2 as the fuel and LOx as the oxidizer. The stage also uses about 154 kg (340 lb) of hydrazine for reaction control (Freeman 2006, ULA 2013).

2.1.6.2.3. Payload Fairing

The PLF for the Delta IV is about 5.1 m (16.8 ft) in diameter and about 19.1 m (62.7 ft) in length and constructed of composite materials. The PLF encloses and protects the spacecraft from thermal, acoustic, electromagnetic, and environmental conditions during ground operations and lift-off through atmospheric ascent (ULA 2013). Figure 2-11 depicts the Mars 2020 spacecraft within the PLF envelope.

2.1.6.2.4. Delta IV Space Launch Complex-37 (SLC-37)

SLC-37 is located in the northeastern section of CCAFS. The launch complex consists of a launch pad, a mobile service tower (MST), a fixed umbilical tower, propellant and water storage areas, an exhaust flume, catch basins, security services, fences, support buildings, and facilities necessary to prepare, service, and launch Delta IV vehicles (USAF 1998, ULA 2013).

Security at SLC-37 is ensured by a perimeter fence, guards, and restricted access. Since all operations in the launch complex would involve or be conducted in the vicinity of liquid or solid propellants and explosive devices, the number of personnel permitted in the area, safety clothing to be worn, the type of activity permitted, and equipment allowed would be strictly regulated. The airspace over the launch complex would be restricted at the time of launch.

2.1.6.2.5. Launch Vehicle Processing

Delta launch vehicle preparation activities and procedures during and after launch have been previously documented (USAF 1998, ULA 2013). All NASA launches follow the current standard operating procedures.

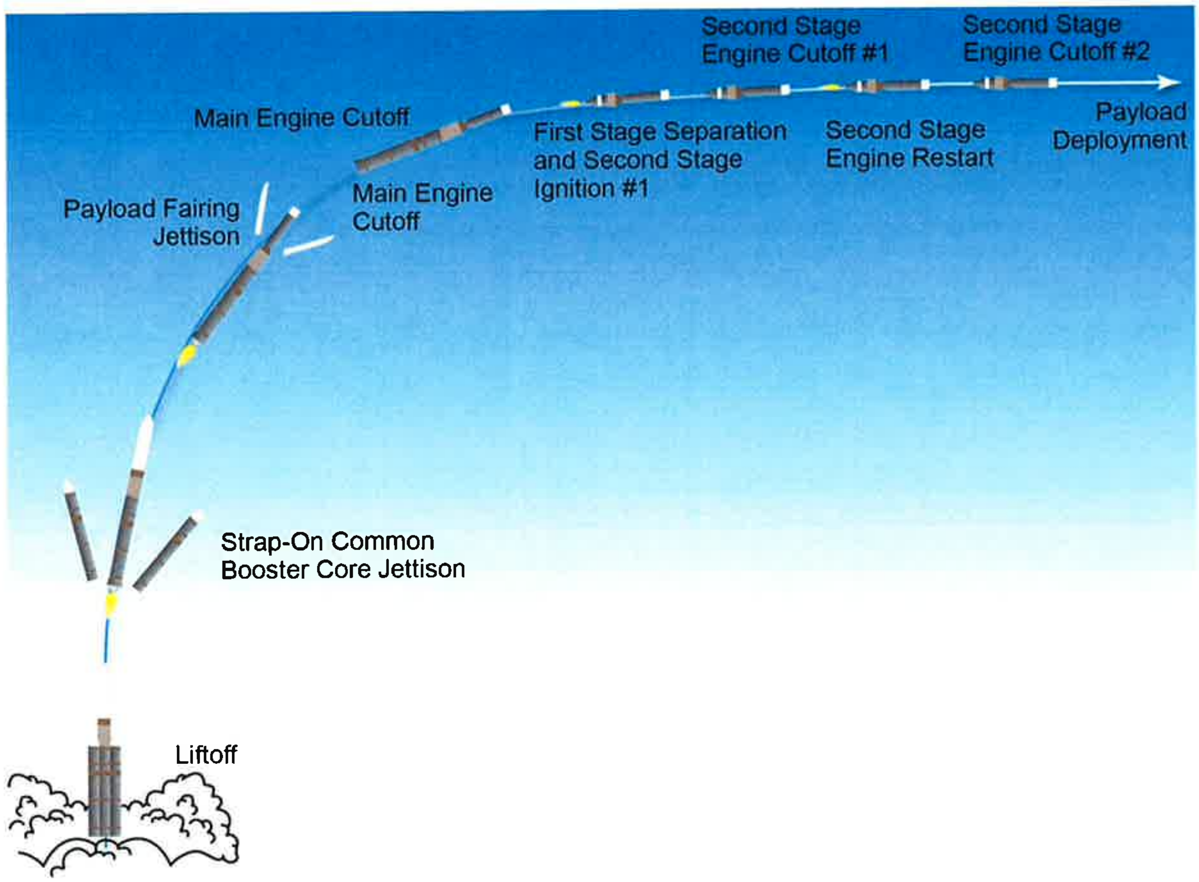
The Delta IV launch vehicle components for the Mars 2020 mission would be received at CCAFS, where they would be inspected, stored, and processed at appropriate facilities. When needed for launch, the components would be moved to the Horizontal Integration Facility at SLC-37, where the launch vehicle would be assembled, integrated, and tested. The Delta IV launch vehicle would then be moved via rail on the MST to the launch pad at SLC-37. The PLF, containing the Mars 2020 spacecraft, would then be transported from the PHSF at KSC directly to the launch pad at SLC-37 and mated to the second stage. The MMRTG would then be installed on the spacecraft. The vehicle would then be loaded with hydrazine and the LOx and LH_2 liquid propellants, and undergo final preparations for launch (ULA 2013).

Processing activities for the Mars 2020 Delta IV vehicle would be similar to those routinely practiced for other Delta launches from CCAFS. Effluents and solid or hazardous wastes that may be generated by these activities are subject to federal and state laws and regulations. NASA or its contractors would dispose of hazardous wastes. CCAFS has the necessary environmental permits and procedures for conducting launch vehicle processing activities (see Section 4.10).

2.1.6.2.6. Launch Profile

Launch of the Delta IV Heavy would begin with simultaneous ignition of the main engines[6] in the three first-stage CBCs (Figure 2-14). The two strap-on CBCs would thrust at a higher level than the central CBC, and their propellant would be depleted sooner. After engine cutoff, the strap-on CBCs would be jettisoned. The central CBC engine would continue to thrust until main engine cutoff, after which the first stage would separate from the second stage. The three depleted CBCs would fall into the Atlantic Ocean in predetermined drop zones and would not be recovered (USAF 2000).

[6] The engines undergo an automatic "health check" 5 seconds before liftoff. Should a malfunction be detected, the engines would be shut down and the launch would be aborted.



Source: Adapted from ULA 2013

Figure 2-14. Typical Delta IV Heavy Ascent Profile

The second stage would be ignited shortly after separation from the first stage. The PLF would then be jettisoned and would also fall into the Atlantic Ocean in predetermined drop zones and would not be recovered. Upon achieving Earth parking orbit, the second stage engine thrust would be cut off via a timed command. After a brief, predetermined coast period in an Earth parking orbit, the second stage engine would restart and the vehicle would accelerate to Earth escape velocity. After second stage engine cutoff, the Mars 2020 spacecraft would separate from the second stage and continue on its trajectory to Mars. The second stage would continue separately into interplanetary space.

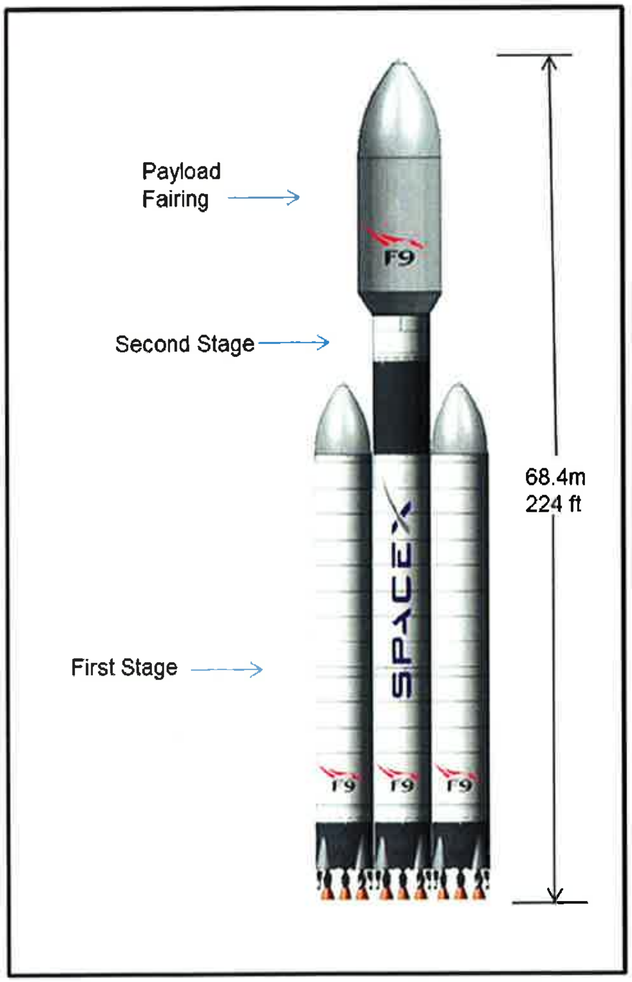
2.1.6.3. Description of the Falcon Heavy Launch Vehicle

The Falcon launch vehicle program was initiated in 2002 when SpaceX was launched as a commercial venture. The current launch vehicle is the Falcon 9. The proposed Falcon Heavy launch vehicle is an evolutionary version of the Falcon 9, with greater payload capability.

The representative Falcon Heavy configuration for the proposed Mars 2020 mission would consist of a liquid propellant first stage (similar to the first stage of the Falcon 9), and two boosters (also similar to the first stage of the Falcon 9), a liquid propellant second stage, the Mars 2020 spacecraft, and a 5-m PLF. The three first stage components are attached to each other, and the second stage is mounted atop the first stage. The Mars 2020 spacecraft would be mounted atop the second stage. The PLF encloses and protects the spacecraft. The Falcon Heavy, depicted in Figure 2-15, is approximately 68.4 m (224 ft) in height and is capable of delivering a 13,200 kg (29,100 lb) payload to Mars. Unlike the Atlas V and Delta IV vehicles, the first stages of the Falcon Heavy are designed to be reusable and could be recovered from the Atlantic Ocean (SpaceX 2013, SpaceX 2013b, USAF 2011).

2.1.6.3.1. First Stage

The Falcon Heavy first stage fuel tanks are constructed mostly of an aluminum and lithium alloy. Each Falcon 9 first stage is about 3.66 m (12 ft) in diameter and about 45.7 m (150 ft) in length. The Falcon 9 first stages are each powered by nine of SpaceX's Merlin engines (a Saturn V heritage engine) in an octagonal arrangement with one center engine. Each contains about 261,000 kg (576,000 lb) of propellant consisting of 81,600kg (180,000 lb) of RP-1 as the fuel, and 180,00 kg (397,000 lb) LOx[7] as the oxidizer for a total first stage propellant load of 245,000 kg (540,000 lb) of RP-1 and 539,000 kg (1,190,000 lb) of LOx. The Falcon Heavy utilizes a propellant cross-feed system; propellant is supplied from the two boosters to the center core so that fuel is preferentially drawn from the boosters first. This allows the central core to continue to maintain a significant portion of its initial fuel load and to operate well



Source: Adapted from FAA 2013

Figure 2-15. A Falcon Heavy Launch Vehicle

[7] Fuel quantities are for a Falcon 9. Falcon Heavy quantities may differ slightly from these amounts.

after the boosters are jettisoned. A cylindrical interstage that encloses the second stage is mounted on the central Falcon 9 first stage. Aerodynamic nosecones are mounted on the two Falcon 9 boosters in place of the interstage (SpaceX 2013; NASA 2011; FAA 2013).

2.1.6.3.2. Second Stage

The Falcon Heavy second stage is constructed of aluminum and composite material and is about 3.66 m (12 ft) in diameter and about 12.5 m (41 ft) in length. The stage is powered by a single Merlin engine, and contains about 49,000 kg (108,000 lb) of propellant, consisting of 15,100 kg (33,300 lb) of RP-1 as the fuel and 33,900 kg (74,700 lb) LOx as the oxidizer (NASA 2011, FAA 2013).

2.1.6.3.3. Payload Faring

The PLF for Falcon Heavy is about 5.2 m (17.1 ft) in diameter and about 15.2 m (50 ft) in length and is constructed of an aluminum core with carbon fiber face sheets. The PLF encloses and protects the spacecraft from thermal, acoustic, electromagnetic, and environmental conditions during ground operations and lift-off through atmospheric ascent (FAA 2013). Figure 2-11 depicts the Mars 2020 spacecraft within the PLF envelope.

2.1.6.3.4. Falcon Heavy Space Launch Complexes 39A and 40

Space X has launch privileges at both LC-39A and SLC-40. As currently configured, neither complex is capable of supporting the launch of the Falcon Heavy, although it is anticipated that LC-39A would be modified to support launch of the Falcon Heavy. Modifications to either launch complex to support this vehicle would be performed as part of the Falcon Heavy launch program and not specifically for the Mars 2020 mission.

LC-39, located on KSC, has been used as part of the Apollo program and for the Space Shuttle program. SpaceX recently won launch privileges from LC-39A, one of two launch pads within the launch complex. The launch complex is composed of, among other facilities, the two launch pads, the Vehicle Assembly Building, the Orbiter Processing Facility buildings, the Launch Control Center (which contains the firing rooms), and various logistical and operational support buildings.

2.1.6.3.5. Launch Vehicle Processing

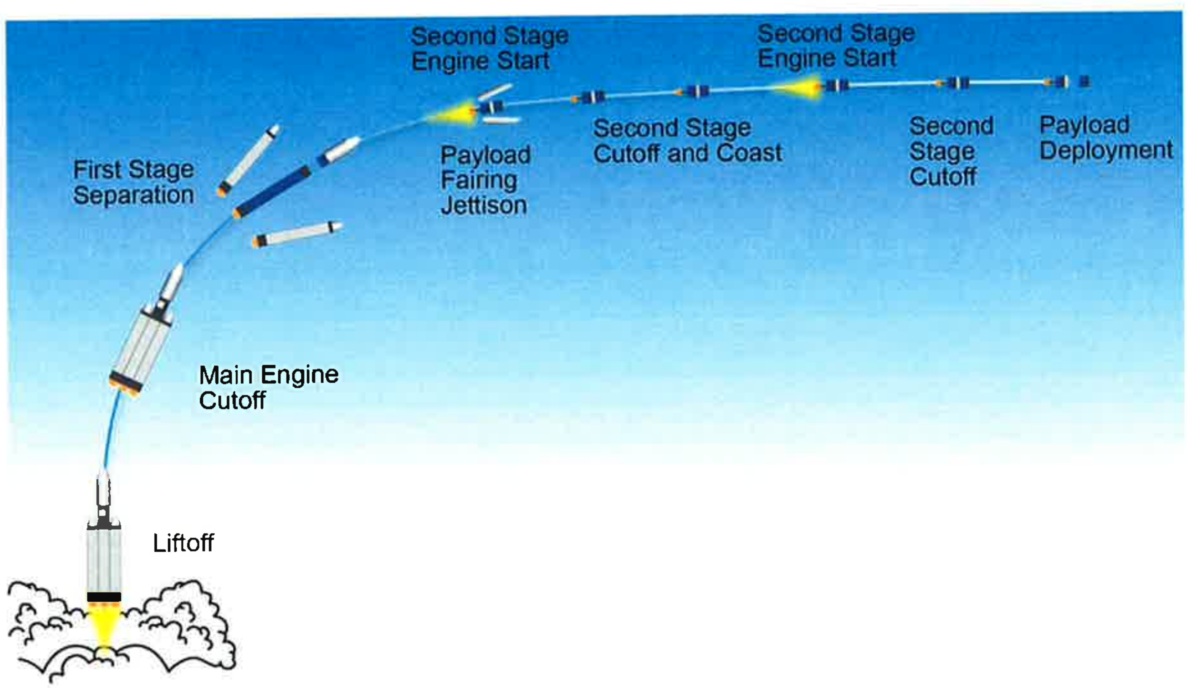
All NASA launches follow the current standard operating procedures.

A Falcon Heavy has not been launched from KSC or CCAFS. The following descriptions are based on the process used for the Falcon 9. The Falcon Heavy launch vehicle components for the Mars 2020 mission would be received at KSC or CCAFS, where they would be inspected, stored, and processed at appropriate facilities. When needed for launch, the components would be moved to the Falcon 9 facility at LC-39A where the launch vehicle would be assembled, integrated, and tested. The PLF, containing the Mars 2020 spacecraft, would then be transported from the PHSF at KSC to the Falcon 9 facility at LC-39A and mated to the second stage. The MMRTG would then be installed on the spacecraft. The Falcon Heavy launch vehicle would then be moved via the vertical transporter-erector to the launch pad at LC-39A. The launch vehicle would be transported in a horizontal position and raised to a vertical position at the launch pad. The vehicle would then be loaded with hydrazine and the LOx and RP-1 liquid propellants and undergo final preparations for launch (Univ 2011).

Processing activities for the Mars 2020 Falcon Heavy vehicle would be similar to those routinely practiced for other Falcon launches from CCAFS. Effluents and solid or hazardous wastes that may be generated by these activities are subject to federal and state laws and regulations. NASA, or its contractors, would dispose of hazardous wastes. CCAFS has the necessary environmental permits and procedures for conducting launch vehicle processing activities (see Section 4.10).

2.1.6.3.6. Launch Profile

Launch of the Falcon Heavy would begin with simultaneous ignition of the main engines in the core first stage and two first-stage boosters (Figure 2-16). The two boosters would be jettisoned when the booster fuel tanks (which have been supplying fuel to the core first stage and both boosters) are nearly depleted. The central core engines would continue to thrust until main engine cutoff, after which the first stage would separate from the second stage. The three depleted first-stage components would fall into the Atlantic Ocean in predetermined drop zones and could be recovered (SpaceX 2013).



Source: Adapted from Spacex 2013

Figure 2-16. Falcon Heavy Ascent Profile

The second stage would be ignited shortly after separation from the first stage. The PLF would then be jettisoned and would also fall into the Atlantic Ocean in predetermined drop zones and would not be recovered. Upon achieving Earth parking orbit, the second stage engine thrust would be cut off via a timed command. After a brief, predetermined coast period in an Earth parking orbit, the second stage engine would restart and the vehicle would accelerate to Earth escape velocity. After second-stage engine cutoff, the Mars 2020 spacecraft would separate from the second stage and continue on its trajectory to Mars. The second stage would continue separately into interplanetary space.

2.1.6.4. Flight Termination System

Range Safety requires launch vehicles to be equipped with safety systems, collectively called the Flight Termination System (FTS), which are capable of causing destruction of the launch vehicle in the event of a major vehicle malfunction. Range Safety further specifies in the *Range Safety User Requirements Manual* (USAF 2004) that for any launch vehicle, the FTS reliability goal shall be a minimum of 0.999 at the 95 percent confidence level. The FTS for the Mars 2020 mission would provide the capability to destroy the launch vehicle either (1) autonomously after detecting an inadvertent breakup of the vehicle or unintentional separation of vehicle stages, or (2) by commands issued via secure radio links. The primary elements of the FTS, common for any of the candidate launch vehicles, would consist of an Automatic Destruct System (ADS) and a Command Destruct System (CDS). The FTS for the Atlas V would also include a Centaur Automatic Destruct System (CADS).

If inadvertent vehicle breakup or premature stage separation occurs, the ADS would automatically initiate ordnance components that split open all first- and second-stage propellant tanks to disperse the liquid propellants and split any strap-on solid rocket casings to terminate solid motor thrusting. Upon receipt of valid commands from Range Safety, the CDS would shut down the first stage or second stage main engines (depending on the timing of the event), and initiate destruction of the vehicle in the same manner as the ADS.

The FTS for all candidate LVs would be armed shortly before liftoff. Each major component of the FTS would be safed (automatically deactivated) at various times during the vehicle's ascent when the component would no longer be needed and to preclude its inadvertent activation. The ADS would be safed prior to separation of the first and second stages and the CDS would be safed immediately after the second stage with the Mars 2020 spacecraft has achieved Earth parking orbit.

For the Atlas V candidate LVs, an Inadvertent Separation Destruct System (ISDS) would be incorporated on each of the four SRBs. In the event of an inadvertent or premature separation of an SRB, the ISDS would initiate a linear-shaped charge to disable the SRB after a brief time delay to assure clearance from the Atlas V. The ISDS would be deactivated during a normal SRB separation event.

2.1.6.5. Range Safety Considerations

CCAFS has implemented range safety requirements (USAF 2004) that support launches from KSC and CCAFS. For the Mars 2020 mission, predetermined flight safety limits would be established for each day of the launch period. Wind criteria, impacts from fragments that could be produced in a launch accident, dispersion and reaction (e.g., toxic plumes, fire) of liquid and solid propellants, human reaction time, data delay time, and other pertinent data would be considered when determining flight safety limits. The Mission Flight Control Officer would take any necessary actions, including destruction of the vehicle via the CDS, if the vehicle's trajectory indicates flight malfunctions (e.g., exceeding flight safety limits) (USAF 2004).

Range Safety at CCAFS uses models to predict launch hazards to the public and launch site personnel prior to a launch. These models calculate the risk of injury resulting from toxic exhaust gases from normal launches, and from potentially toxic concentrations due to a failed launch. The launch would be postponed if the predicted collective risk of injury from exposure to toxic gases exceeds established limits (USAF 2004). Range Safety monitors launch surveillance areas to ensure that risks to people, aircraft, and surface vessels are within acceptable limits. Controlled surveillance areas and airspace are closed to the public as required (USAF 2004).

2.1.6.6. Electromagnetic Environment

Launch vehicles may be subject to electromagnetic conditions such as lightning, powerful electromagnetic transmissions (e.g., radar, radio transmitters), and charging effects (i.e., electrical charges generated by friction and the resultant electrostatic discharges). NASA and the USAF address such conditions with respect to the design of the launch vehicle, as well as with ordnance (e.g., explosives, explosive detonators, and fuses), fuels, exposed surfaces of the vehicle, and critical electronic systems that must have highly reliable operations. A large body of technical literature exists on these subjects and has been used by NASA and the USAF in designing safeguards (see, for example, USAF 2004). The launch vehicle, the Mars 2020 spacecraft, and the launch support systems would be designed and tested to withstand these environments in accordance with requirements specified in USAF 2004.

2.1.7 Radiological Emergency Response Planning

Prior to launch of the Mars 2020 mission, a comprehensive set of plans would be developed by NASA to ensure that any launch accident could be met with a well-developed and tested response. NASA's plans would be developed in accordance with the National Response Framework (NRF) (DHS 2013) and the NRF Nuclear/ Radiological Incident Annex (DHS 2008) with the combined efforts of the U.S. Department of Homeland Security (DHS), DHS's Federal Emergency Management Agency (FEMA), DOE, the U.S. Department of Defense (DoD), the U.S. Department of State (DOS), the U.S. Environmental Protection Agency (EPA), the state of Florida, Brevard County, and local governmental organizations. These organizations and other federal agencies, as appropriate, could be involved in response to a radiological emergency. The radiological contingency planning and implementation for a Mars 2020 mission would be expected to be similar to the process used for the 2011 MSL mission launch (Scott 2012).

The radiological emergency response plan would be exercised prior to launch to verify that the response interfaces, command channels, and field response organizations would be prepared to respond in the unlikely event of a launch accident. Thus, in the event of a declaration of an *Incident of National Significance* (e.g., launch accident) whose impact is within United States jurisdiction, NASA, as the coordinating agency, would work with the DHS to coordinate the entire federal response. Should a release of radioactive material occur in the launch area, NASA would provide information on the estimated release and its recommendations to the state of Florida, Brevard County, and local governments who, in turn, would determine an appropriate course of action (such as sheltering in place, evacuation, exclusion of people from contaminated land areas, or no action required), and with full access to the coordinated federal response. For accidents outside United States' jurisdiction and defined as *Incidents of National Significance*, NASA and DHS would assist the DOS in coordinating the United States' response via diplomatic channels and deploy federal resources as requested.

To manage the radiological contingency response, NASA would establish a radiological emergency response capability that would include a radiological assessment and command center as well as field monitoring assets that would be deployed prior to launch both onsite and offsite. The assessment and command center would be the focal point for NASA and DHS coordination efforts. This center would also be used to coordinate the initial federal response to a radiological contingency until the Mars 2020 spacecraft has left Earth orbit. Pre-deployed assets to support a response to a potential launch accident would include representation from NASA, DHS, DOE, DoD, DOS, EPA, USAF, the National Oceanic and Atmospheric Administration (NOAA), the state of Florida, and Brevard County. If measureable amounts of plutonium are detected after a launch vehicle accident, the center would issue appropriate direction to KSC/CCAFS personnel as well as the public to ensure minimal or no potential exposures.

If impact occurs in the ocean following an accident, NASA would coordinate with the DHS, the U.S. Coast Guard, the U.S. Navy, and DOE to initiate security measures and assess the feasibility of search and retrieval operations. Efforts to recover the MMRTG or its components would be based on technical feasibility and in consideration of any potential health hazards presented to recovery personnel and potential environmental impacts.

2.2 DESCRIPTION OF ALTERNATIVE 2

The mission and spacecraft for Alternative 2 would be designed and developed, to the extent practicable, to meet the operational capabilities summarized in Table 2-1. In Alternative 2, the MMRTG power source would be replaced by a solar power array. The rover used in this alternative would rely on the power generated by solar arrays to generate electricity to operate the rover's scientific instrumentation and communication equipment and provide motive power. Power from the solar arrays would also power electric heaters to maintain the thermal environment required to ensure the survival of the rover's engineering subsystems and science payload. The descriptions presented in this section for Alternative 2 are based on the information available at the time this DEIS was prepared, as presented in the *Mars 2020 Solar Feasibility Study* (JPL 2014). Should NASA make changes in Alternative 2 that are relevant to environmental concerns, NASA would evaluate the need for additional environmental analysis documentation.

2.2.1 Mission and Spacecraft Description

Many of the technical aspects of the mission and spacecraft designs for Alternative 2 would be similar to those described in Section 2.1 for the Proposed Action (Alternative 1). These would include the following major features.

- The Mars 2020 spacecraft would be launched from KSC or CCAFS onboard an expendable launch vehicle from the Atlas V, Delta IV, or Falcon Heavy class of vehicles (see Section 2.1.5 for representative descriptions of these vehicles).
- The mission design would be as described in Section 2.1.1, including a launch opportunity in July to August of 2020, with a backup opportunity in August to September 2022, and an Earth-Mars trajectory leading to direct entry of the spacecraft into the Martian atmosphere.
- The Mars 2020 flight system would consist of a high-heritage MSL cruise stage, entry vehicle, and descent stage (as described in Section 2.1.2), and a science rover.
- The rover's science instrument payload would be as described in Table 2-2. Planning for the rover science mission would be based upon an operational timeline similar to that described in Section 2.1.1.

2.2.1.1. Solar Power Supply System

The Mars 2020 rover for Alternative 2 would use a solar array as the source of electrical power for its engineering subsystems and science payload (JPL 2014). The size of the array would be limited by the volume constraints of the rover in its stowed configuration within the descent stage inside the entry vehicle, which in turn is limited in size by the diameter of the launch vehicle payload fairing (see Figure 2-11). Use of a solar array would be expected to increase the mass of power supply systems for the rover by less than 10 kg (22 lb) compared to the use of the MMRTG (JPL 2014). The solar array would attach to the back section of the rover and would be folded for stowage inside the entry vehicle. The array would be deployed after the rover has landed on the surface of Mars. A representative deployed array configuration is illustrated in Figure 2-17.

Source: Adapted from JPL 2014

Figure 2-17. A Representative Solar-Powered Alternative 2 Mars 2020 Rover

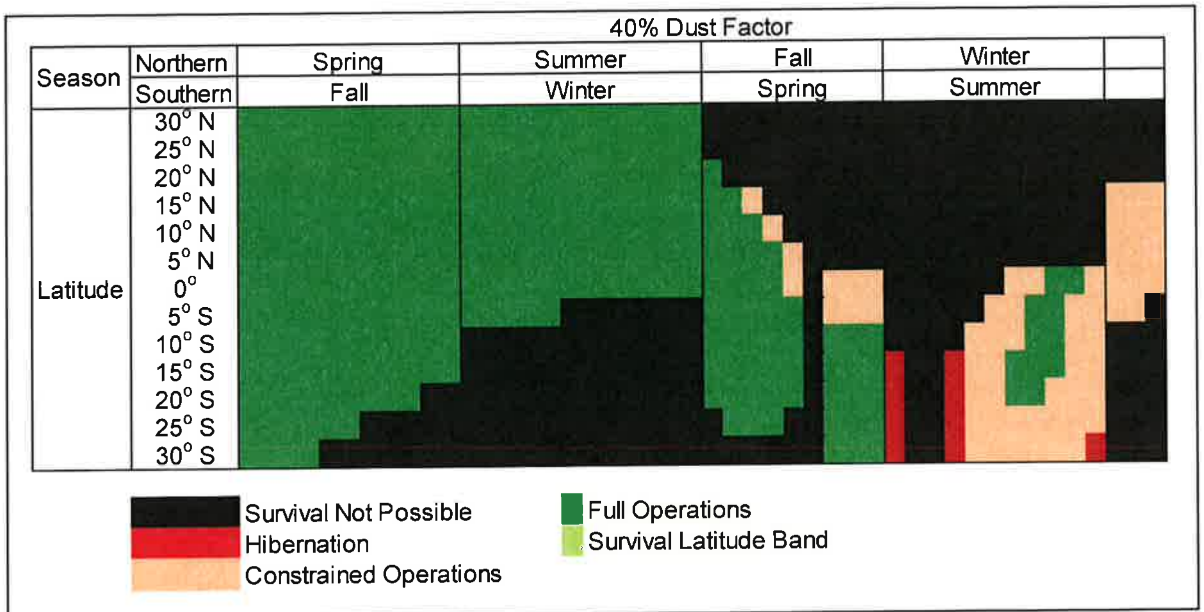
After landing, the solar array would be deployed into two separate panels and would be in a fixed position parallel with the upper surface of the rover chassis. The deployed array of two panels would have an active cell surface area of approximately 7.4 square meters (80 square feet). The array would consist of the same type of multi-junction solar cells as were used on the Mars Exploration Rovers (MERs), which landed on Mars in January 2004. At the atmospheric temperatures of the MER landing sites near the equator of Mars, this array would have a conversion efficiency of about 26 percent.

2.2.2 Solar Power Availability

The available electrical power produced by the solar array described in Section 2.2.1.1 would be a function of several factors (JPL 2014). The most important of these are the landing site latitude and time of year on Mars, which affect the incidence angle of the sunlight shining on the array and the amount of time sunlight is available per sol. Low incidence angles at high latitudes, reduced solar intensity near Mars aphelion, and short periods of daylight during a Martian Winter would reduce the available amount of electrical power produced by the solar array. Other factors affecting array output would include shadowing of the array from the masts and antennas, the amount of dust in the Martian atmosphere, and dust deposition and accumulation on the array.

All of the energy that this solar array would generate per sol could not be used exclusively to perform science operations. The rover would need to maintain its thermal health and mechanical functionality so that it could communicate with Earth and drive to specified science locations. The solar energy required to maintain the rover's thermal health would vary with latitude (i.e., landing site) and time of year. During the Martian Winter there would be a higher demand for heat to maintain the rover's components within acceptable thermal limits, but there would be less total energy available from the solar array for the reasons discussed above.

Of the available energy per sol, approximately 100 to 600 watt-hours would be needed to perform science operations, which would include driving to science locations, site reconnaissance, and acquiring and analyzing samples and other scientific data (JPL 2013). The remainder of the available energy would be needed for the rover's engineering functions, including communications and thermal control. Figure 2-18 illustrates locations on the surface of Mars where the baseline solar array configuration would provide sufficient power for the rover to perform science operations and maintain its health and functionality as a function of latitude over the course of one Mars year.



Note: The expected arrival date for the Mars 2020 mission would coincide with the transition from winter to spring in the northern hemisphere of Mars. This date appears on the far left of this figure.

Source: Adapted from JPL 2014

Figure 2-18. Mars 2020 Solar-Powered Rover Operability (40% Dust Factor)

As shown in the figure, one of the factors that affects the feasibility of using solar power is the dust factor. This factor is a measure of the remaining electric power output from the solar arrays when some energy is blocked due to the accumulation of dust on the surface of the arrays panels. The dust factor is the percentage of the effective array surface that remains clean, and is roughly equivalent to the total power still available given the accumulation of dust—the lower the dust factor, the lower the amount of electrical energy produced. A 40% dust factor[8] means that 40% of the array surface area is clean and the electrical output of the solar arrays is reduced proportionally. Higher dust factors, may be achievable only with the use of active dust mitigation technology or with the assumption of more frequent environmental cleaning events.

For Alternative 2, sufficient solar power for one Mars year is not available at any latitude assuming a dust factor of 40% (the solar array remains at least 40% dust free). With more frequent dust cleaning or mitigation resulting in a dust factor of at least 70% (the solar array remaining at least 70% dust free), the rover could operate for one Mars year only at approximately 5° south latitude.

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[8] The 40% dust factor is based upon the dust accumulation rates on the MER solar arrays. The accumulation of dust is limited by naturally occurring cleaning events. The 70% dust factor relies upon more frequent cleaning events or active dust mitigation technology. Dust mitigation technology may improve the dust factor to beyond 70%, but these technologies have not been demonstrated to function in Martian environmental conditions.

The solar feasibility assessment (JPL 2014), which developed these estimates of rover operability, was performed with sufficient detail to develop estimates for a representative solar-powered rover configuration. Should NASA select Alternative 2, the solar-powered rover design would be finalized, but any changes would likely not change the fundamental results presented in the solar feasibility assessment.

2.2.3 Operational Considerations

As shown in Figure 2-18, for all latitudes between 30° south and 30° north, the survival of the rover for a full Martian year would not be expected. There are times when the rover would have to operate at less than full capabilities (Constrained Operations), times when the rover would have to cease scientific operations and operate in a mode where only functions needed for rover survival—primarily maintaining an acceptable thermal environment—are performed (Hibernation), and times the rover would not survive. These periods of reduced science operability impact the amount of science investigation that can be performed at the various landing sites which adversely impacts the ability of the rover to reach all of the baseline goals for the mission.

Table 2-5 shows the estimated operational lifetime of the solar-powered Mars 2020 rover as a function of landing site latitude for the anticipated arrival dates. The MMRTG power option, which is capable of full operations for an entire Mars year, is included for comparison purposes. This table reflects the fact that the solar power alternative with a 40% dust factor is not capable of surviving for a full Mars year, although science operations could be performed for parts of the year. The ability of the rover to survive longer in the northern latitudes is a result of the mission arrival dates coinciding with spring in the northern Martian latitudes while these arrival dates are in the fall in the southern Martian latitudes.

A larger dust factor (70%) would extend the operational lifetime of the rover and would allow for full year operation between 5° south latitude and 0° and would marginally extend the operational lifetime of the rover at some latitudes, thereby increasing the amount of science that could be performed.

The science capabilities associated with partial-year operation are provided in Table 2-6. These capabilities are expressed in terms of the percentage of the samples that could be obtained given a full year of operation with no limitations (constrained operations or hibernation).

For comparison, Alternative 1 provides 100% capability. Given the assumptions for initial checkout and rover movement (driving from site to site) any landing site with an operational lifetime of 40% or less would not provide the opportunity to perform any sampling activities unless the rover movement was curtailed.

Table 2-5. Operational Lifetime for a Solar-Powered Rover

Option		Operational Lifetime[a]	
		Solar	MMRTG
Latitude	30° N	50%	100%
	25° N	50%	100%
	20° N	50%	100%
	15° N	55%	100%
	10° N	60%	100%
	5° N	60%	100%
	0°	60%	100%
	5° S	35%	100%
	10° S	25%	100%
	15° S	25%	100%
	20° S	20%	100%
	25° S	15%	100%
	30° S	10%	100%

Source: Adapted from JPL 2014

(a) Lifetime expressed in terms of a full Martian year. Lifetime assuming a 40% dust factor (solar cells remain 40% clean)

Rover operates for less than a full Martian year

Rover operates for a full Martian year

Table 2-6. Science Capability

Operational Lifetime in Mars Years[(a)]	Percent of Mars Year Assumed for Initial Checkout and Driving[(a)]	Percent of Mars Year Available For Sampling Activities[(a)]	Percent of Sampling Activities Available on an MMRTG Mission[(b)]
50%	40%	10%	17%
60%	40%	20%	33%
70%	40%	30%	50%
100%	40%	60%	100%

(a) These values are in terms of a full Martian year (689 Earth days). For example, 40% of a Martian year is 276 Earth days.

(b) The fourth column represents the expected sampling capability, expressed as a percentage of the capabilities associated with unconstrained operation for a full year.

2.3 DESCRIPTION OF ALTERNATIVE 3

The mission and spacecraft for Alternative 3 would be designed and developed, to the extent practicable, to meet the operational capabilities summarized in Table 2-1. The descriptions presented in this section for Alternative 3 are based on the information available at the time this DEIS was prepared, as presented in the *Mars 2020 Solar Feasibility Study* (JPL 2014). In Alternative 3, the MMRTG would be replaced and the rover would be powered by solar power arrays, similar to that proposed in Alternative 2. The rover used in this alternative would rely on the power generated by solar arrays to generate electricity to operate the rover's scientific instrumentation, communication equipment, and to provide motive power. In addition to the solar arrays, the rover in this alternative would incorporate up to 71 LWRHUs as a heat source. Power from the solar arrays would also power electric heaters to augment the LWRHUs to help maintain the thermal environment required to ensure the survival of the rover's engineering subsystems and science payload. As described in the following sections, the additional thermal power from the LWRHUs extends the operational capabilities of the rover to include an expanded selection of landing sites and an increased science return capability. Should NASA make changes in Alternative 3 that are relevant to environmental concerns, NASA would evaluate the need for additional environmental analysis and documentation.

2.3.1 Mission and Spacecraft Description

Many of the technical aspects of the mission and spacecraft designs for Alternative 3 would be similar to those described in Section 2.1 for Proposed Action (Alternative 1). These would include the following major features.

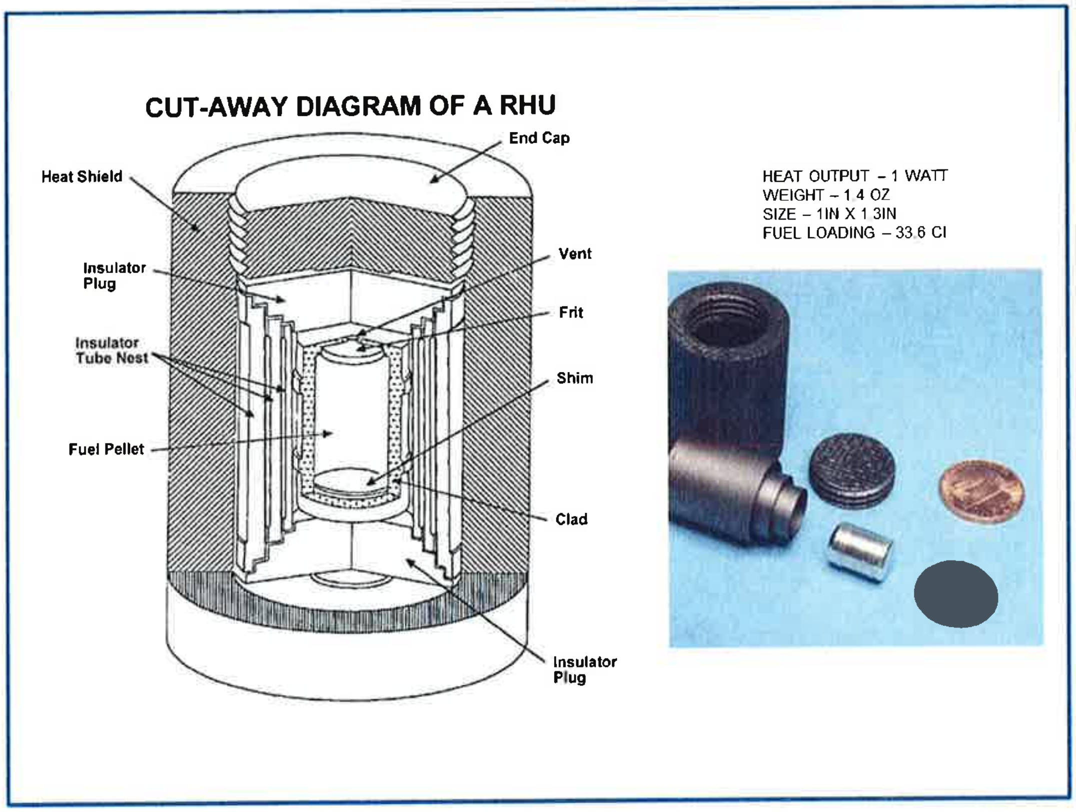
- The Mars 2020 spacecraft would be launched from KSC or CCAFS onboard an expendable launch vehicle from the Atlas V, Delta IV, or Falcon Heavy class of vehicles (see Section 2.1.5 for representative descriptions of these vehicles).
- The mission design would be as described in Section 2.1.1, including a launch opportunity in July to August of 2020, with a backup opportunity in August to September 2022, and an Earth-Mars trajectory leading to direct entry of the spacecraft into the Martian atmosphere.
- The Mars 2020 flight system would consist of a high-heritage MSL cruise stage, entry vehicle, and descent stage as described in Section 2.1.2, and a science rover.
- The rover's science instrument payload would be as described in Table 2-2. Planning for the rover science mission would be based upon an operational timeline similar to that described in Section 2.1.1.

2.3.1.1. Solar Power Supply System

The solar power system that would be used for Alternative 3 is the same system described in Section 2.2.1.1 above for Alternative 2.

2.3.1.2. Radioisotope Heater Units

The Mars 2020 rover could use a combination of LWRHUs and electric heaters to maintain internal temperature during periods of extreme cold. Alternative 3 considers the use of up to 71 such LWRHUs (JPL 2014). Each LWRHU (see Figure 2-19) would produce about 1 thermal watt of heat derived from the radioactive decay of 2.7 grams (g) (0.095 ounce (oz)) of plutonium (mostly plutonium-238) in the form of a ceramic of plutonium dioxide. Each LWRHU would contribute approximately 33.2 Ci for a total plutonium inventory of up to 2,360 Ci. Table 2-7 provides the typical radionuclide composition of a LWRHU's fuel. The exterior dimensions of a LWRHU are 2.6 cm (1.03 in) in diameter by 3.2cm (1.26 in) in length. Each LWRHU has a mass of about 40g (1.4 oz).



Source: Adapted from SNL 2014

Figure 2-19. Principal Features of a Light-Weight Radioisotope Heater Unit (LWRHU)

LWRHUs are designed to contain the plutonium dioxide during normal operations and under a wide range of accident environments. The integrity and durability of LWRHUs have been well documented by the U.S. Department of Energy (SNL 2014). The plutonium dioxide ceramic is encapsulated in a 70% platinum and 30% rhodium alloy clad. A fine weave pierced fabric of carbon graphite used as a heat shield provides protection against high-temperature accident environments, and a series of concentric pyrolitic graphite[9] sleeves and end plugs thermally insulate the encapsulated radioactive material. The LWRHU's plutonium dioxide is principally protected from ground or debris impact by the alloy clad. The heat shield and inner pyrolitic graphite insulators provide additional protection.

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[9] Pyrolitic graphite is a man-made form of graphite, created by heating graphite and allowing it to cool into a crystalline form. This type of graphite has enhanced thermal conduction properties compared to ordinary graphite.

Table 2-7. Typical Radionuclide Composition of a LWRHU Fuel Pellet

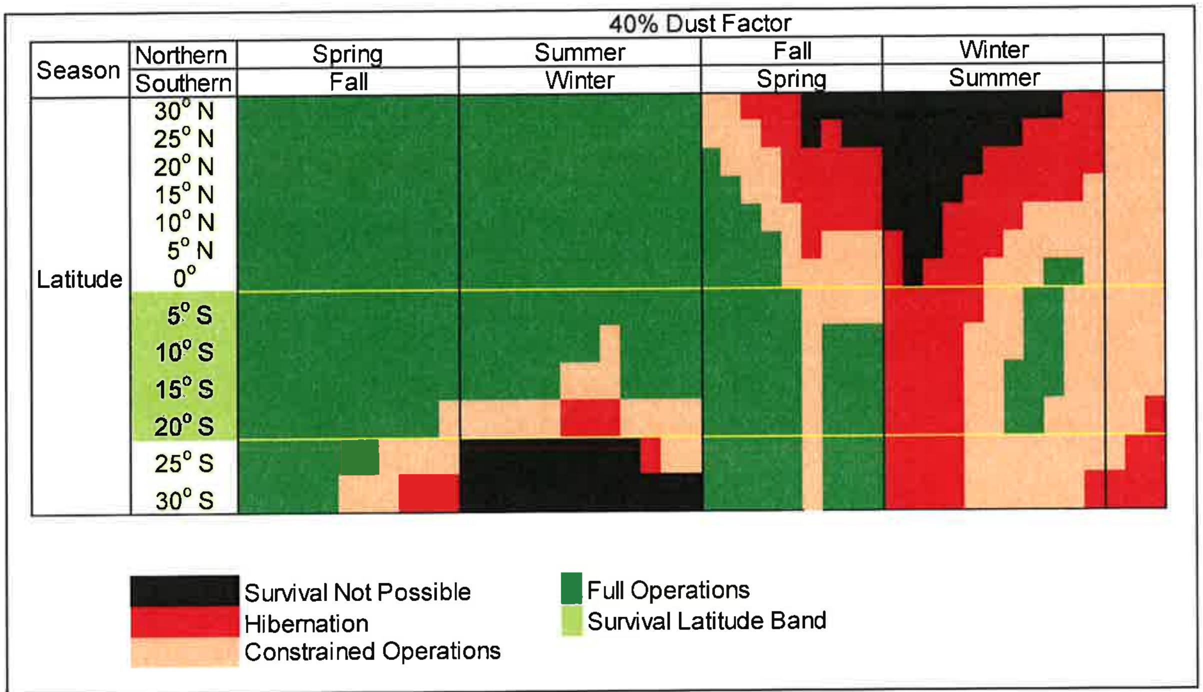
Fuel Component	Weight Percent	Half-Life (yrs)	Specific Activity (Ci/g of Fuel Component)a	Total Activity (Ci)
Plutonium (Pu)	85.735			
Pu-236	0.0000010	2.851	531.3	0.00001
Pu-238	70.810	87.7	17.12	32.7312
Pu-239	12.859	24.131	0.0620	0.02153
Pu-240	1.787	6.569	0.2267	0.01094
Pu-241	0.168	14.4	103.0	0.4672
Pu-242	0.111	375,800	0.00393	0.00001
Actinide impurities	2.413	NA	NA	NA
Oxygen	11.852	NA	NA	NA
Total	**100**	**NA**	**NA**	**33.2312**

2.3.2 Solar Power Availability

The factors affecting the ability of a solar-powered rover to operate on the surface of Mars were discussed in Section 2.2.2 and 2.2.3 for Alternative 2, and are applicable to this alternative as well. Figure 2-20 illustrates the locations on the surface of Mars where there would be sufficient solar power (augmented by the thermal output of the LWRHUs) for the rover to perform science operations and maintain its health and functionality as a function of latitude over the course of one Mars year. The analysis of this alternative assumes the same dust factors as assumed in the analysis of Alternative 2.

For Alternative 3, sufficient solar power for one Mars year of operation (although the rover would be required to hibernate for at least part of the winter) is available between 20° south and 5° south latitudes assuming a dust factor of 40%. With improved dust cleaning or mitigation resulting in a dust factor of 70%, the rover could operate for one Mars year between 20° south and 15° north latitudes.

The solar feasibility assessment (JPL 2014), which developed these estimates of rover operability, was performed with sufficient detail to develop estimates for a representative solar-powered rover configuration. Should NASA select Alternative 3, the solar-powered rover design would be finalized, but any changes would likely not change the fundamental results presented in the solar feasibility assessment.



Note: The expected arrival date for the Mars 2020 mission would coincide with the transition from winter to spring in the northern hemisphere of Mars. This date appears on the far left of this figure.

Source: Adapted from JPL 2014

Figure 2-20. Mars 2020 Solar-Powered (with LWRHUs) Rover Operability with 40% Dust Factor

2.3.3 Operational Considerations

As shown in Figure 2-20, for all latitudes between 30° south and 30° north, the ability of the rover to fully perform for a full year is restricted. There are times when the rover would have to operate at less than full capability (Constrained Operations), times when the rover would have to cease scientific operations and operate in a mode where only functions needed for rover survival—primarily maintaining an acceptable thermal environment—are performed (Hibernation), and times the rover would not survive. Full year survival is only possible between 20° and 5° south latitudes. The periods of reduced science operability impact the amount of science investigation that can be performed at the various landing sites which adversely impacts the ability of the rover to reach all of the baseline goals for the mission.

Table 2-8 shows the estimated operational lifetime of the solar-powered Mars 2020 rover as a function of landing site latitude for the anticipated arrival dates. The MMRTG power option, which is capable of full operations for an entire Mars year, is included for comparison purposes. The numbers shown for a partial-year operation are indicative of how long the rover would be expected to survive before failing due to cold weather. The ability to survive longer in the northern latitudes is a result of the mission arrival dates coinciding with spring in the northern Martian latitudes, while these arrival dates are in the fall in the southern Martian latitudes.

Table 2-8. Operational Lifetime for a Solar-Powered Rover with LWRHUs

Operational Lifetime[a]			
	Option	Solar plus LWRHUs[b]	MMRTG
Latitude	30° N	50%	100%
	25° N	55%	100%
	20° N	60%	100%
	15° N	60%	100%
	10° N	60%	100%
	5° N	65%	100%
	0°	70%	100%
	5° S	100%	100%
	10° S	100%	100%
	15° S	100%	100%
	20° S	100%	100%
	25° S	25%	100%
	30° S	15%	100%

Source: Adapted from JPL 2014

(a) Lifetime expressed in terms of a full Martian year

(b) Lifetime assuming a 40% dust factor (solar cells remain 40% clean)

Rover operates for less than a full Martian year

Rover operates for a full Martian year

Larger dust factors would improve the operational capabilities of the rover; however, even with a dust factor of 70%, a full year of rover operation is possible only between 20° south and 15° north latitudes. The improvement in survivability would result in an increase in the amount of science that could be performed and an increase in the range of locations and, therefore, the number of potential landing sites.

The science capabilities associated with a partial-year operation are provided in Table 2-6. These capabilities are expressed in terms of the percentage of the samples that could be obtained given a full year of operation with no limitations (constrained operations or hibernation). For comparison, Alternative 1 provides 100% capability. Although the rover would be expected to survive for an entire year at latitudes between 20° and 5° south, it would not be able to operate at full capacity for the entire year (Figure 2-20). The limited operational capability during the winter (constrained operation and hibernation) limit the amount of science that can be performed to 60 to 70% of that possible during a full year of unrestricted operations. Given the assumptions for initial checkout and rover movement (driving from site to site) any landing site with an operational lifetime of 40% or less would not provide the opportunity to perform any sampling activities unless rover movement was curtailed.

2.4 DESCRIPTION OF THE NO ACTION ALTERNATIVE

Under the No Action Alternative, NASA would discontinue preparations for the Mars 2020 mission. The next step in NASA's Mars Exploration Program following the Mars Atmosphere and Volatile EvolutioN (MAVEN) mission in 2014 would not be conducted as currently envisioned (excluding the joint NASA – European Space Agency ExoMars missions), and NASA would need to reevaluate its programmatic options for the 2020 launch opportunity to Mars and beyond.

Without development and implementation of a large mobile science platform, such as the rover planned for the Mars 2020 mission, NASA's ability to meet the highest recommendation of the National Research Council's Planetary Science Decadal Survey—to acquire detailed scientific information on the habitability and biosignature potential of Mars—would be severely limited, and the advancements in technological and operational capabilities necessary for the future exploration of Mars may not be achieved.

2.5 ALTERNATIVES CONSIDERED BUT NOT EVALUATED FURTHER

There were no alternatives considered but not evaluated further. Alternative radioisotope power sources to the MMRTG were considered in previous environmental impact statements (NASA 2005b, NASA 2006). These alternatives were not considered here since no new information has been developed that would indicate that these power sources would present a viable alternative to the MMRTG.

2.6 COMPARISON OF ALTERNATIVES INCLUDING THE PROPOSED ACTION

For the purpose of the evaluations presented in this DEIS, the primary difference between the baseline Mars 2020 mission described in the Proposed Action (Alternative 1) and the Mars 2020 mission described in Alternatives 2 and 3 is the source of electrical power that would be used for the Mars 2020 rover. For the Proposed Action, the rover power source would be an MMRTG, described in Section 2.1.3; whereas, for Alternative 2, the rover power source would be a solar array, described in Section 2.2.1; and for Alternative 3, the power source would be a solar array augmented by up to 71 LWRHUs, described in Section 2.3.1.

2.6.1 Comparison of Mission Science Capabilities

Since the Mars 2020 rover designs in the Proposed Action (Alternative 1), Alternative 2, and Alternative 3 would carry the same science instruments, any of these three alternatives could conduct the same set of experiments. The estimated science capability for these alternatives, expressed in terms of the percentage of the full science return that could be attained at a given latitude on Mars, is summarized in Table 2-9.

Alternative 1. The MMRTG-powered rover would be capable of achieving all of the target operational capabilities (100% science return) as summarized in Table 2-1, including landing at a scientifically interesting location between 30° south and 30° north latitude, and operating and conducting science for at least one Mars year.

Table 2-9. Estimated Science Capability Comparison of the Mars 2020 Mission Alternatives

Rover Power Alternative	Landing Site Latitude Range	Operational Capability	Percentage of Science Achieved at Landing Site Latitude
MMRTG (Alternative 1)	30°S to 30°N	100%	100%
Solar Array (Alternative 2) (40% dust factor[a])	0° to 30°N	Unable to Operate for Full Year Maximum Operational Lifetime[b] 60%	20-30%
	30°S to 0°	Unable to Operate for Full Year Maximum Operational Lifetime 35%	a few percent
Solar Array with LWRHUs (Alternative 3) (40% dust factor)	30°S to 20°S	Unable to Operate for Full Year Maximum Operational Lifetime 25%	a few percent
	20°S to 5°S	Constrained Operations (up to 28%) Hibernation (up to 9%)	60-70%
	5°S to 30°N	Unable to Operate for Full Year Maximum Operational Lifetime 70%	20-40%

Notes:

a)The MER Opportunity dust factor has always stayed above 40%, but the MER Spirit dust factor fell below 25% (more than 2 Mars years into the mission). The factors controlling dust accumulation are not well known, so there is a risk that a solar-powered mission without dust mitigation technology assuming a minimum dust factor of 40% may fail if the actual dust accumulation exceeds that seen on Opportunity and is closer to that seen on Spirit late in its mission. Meeting a 70% dust factor (i.e., the loss of power from the solar arrays due to accumulated dust is limited to 30%) while promising greater science return would require development of dust removal technology.

b) For each latittude range, the Maximum Operational Lifetime represents the longest time the rover would be expected to survive before failing due to environmental conditions. It is expressed in terms of a full Martian year.

All values are approximate. N = North Latitude; S = South Latitude.

Source JPL 2014

Alternative 2. At most latitudes on Mars, the amount of time that a solar-powered rover could perform science operations would be limited by the ability of the solar array to generate sufficient power for the rover to survive the extreme thermal environment. A solar-powered rover with arrays stowable in the available volume would not be able to survive for a full Martian year at any latitude assuming the solar arrays remain at least 40% dust free. Partial-year operation with reduced science capability is possible over a range of latitudes from 0° to 30°north. More favorable dust factors would result in an increase in the operational range of the rover, expanding the latitudes at which a partial year operation would be possible, with a full year of operation possible only at latitudes ranging from 0° to 5° north. Operations would be limited (constrained operations or hibernation) for parts of the year.

Alternative 3. At most latitudes on Mars, the amount of time that a solar-powered rover, with additional thermal power from LWRHUs, could perform science operations would be limited by the ability of the solar array and LWRHUs to generate sufficient power for the rover to survive the extreme thermal environment. A solar-powered rover with LWRHUs (solar arrays 40% dust free) would have sufficient power to operate for a full Martian year at latitudes on Mars between 20° south and 5° south. Partial-year operation with further reduced science capability is possible over a wider range of latitudes. This solar/RHU-powered rover could operate for at least one Mars year at latitudes ranging from 20° south to 15° north, if a more favorable solar array dust factor of 70% is assumed.

Alternative 1, 2, and 3: 2022 Launch Opportunity. Should the mission be delayed, the proposed Mars 2020 mission would be launched during the next available launch opportunity in August through September 2022. The science potential associated with Alternatives 1, 2, and 3 with a 2022 launch would be similar to those projected for each alternative with a 2020 launch. Under all circumstances, an MMRTG-powered rover would provide more power for science activities.

No Action Alternative. The No Action Alternate would not accomplish any science on the surface of Mars; this does not fulfill the purpose and need for the Mars 2020 mission as discussed in Chapter 1 of this DEIS.

2.6.2 Comparison of Potential Environmental Impacts

This section summarizes and compares the potential environmental impacts of the Proposed Action (Alternative 1), Alternative 2, Alternative 3, and the No Action Alternative. The anticipated impacts associated with nominal or normal implementation of Alternatives 1, 2, and 3 are considered first (Section 2.6.2.1). This is followed by a summary of the non-radiological impacts that could occur due to a potential launch accident with Alternatives1, 2, and 3 (Section 2.6.2.2); and finally a summary of potential radiological consequences and risks from a launch accident associated with each of the Alternatives (Section 2.6.2.3). Details of these results are addressed in Chapter 4.

As noted in Section 2.1.5, the evaluations presented in this DEIS, based on representative configurations of the possible launch vehicles, were completed prior to NASA's selection of the mission launch vehicle. NASA considers these evaluations to adequately bound the potential environmental consequences of the alternatives described in this DEIS. Should NASA's continuing evaluations produce results that differ substantially from the information presented in this DEIS, NASA would consider the new information, and determine the need, if any, for additional environmental analysis and documentation

2.6.2.1. Environmental Impacts of a Normal Launch

Table 2-10 provides a summary comparison of the anticipated environmental impacts associated with normal implementation of Alternatives 1, 2, and 3, and the No Action Alternative.

Alternatives 1, 2, and 3. The impacts associated with a successful launch were addressed in the *Final Environmental Assessment for Launch of NASA Routine Payloads on Expendable Launch Vehicles* (Routine Payload EA) (NASA 2011) for all candidate launch vehicles. These impacts were determined to have no significant impacts, as detailed in the Finding of No Significant Impact (FONSI) for the Routine Payloads EA.

Table 2-10. Summary of Anticipated Environmental Impacts of the Mars 2020 Mission Alternatives

Impact Category	Mars 2020 Mission Alternatives	
	Normal Implementation of the Proposed Action and Alternatives 2 and 3	No Action Alternative
Land Use	Consistent with designated land uses at KSC and CCAFS; no adverse impacts on non-launch-related land uses at KSC and CCAFS would be expected.	No change in baseline condition.
Air Quality	High levels of solid propellant combustion products occur within the exhaust cloud for a launch vehicle using solid rockets boosters (e.g., the Atlas V). The exhaust cloud would rise and begin to disperse near the launch complex. Some short-term local ozone impacts. No long-term adverse air quality impacts would be expected in the region.	No change in baseline condition.
Noise and Sonic Boom	Sound exposure levels during launch are estimated to be within OSHA and EPA regulations/guidelines for affected workers and the public.	No change in baseline condition.
Geology and Soils	Some deposition of Al_2O_3 particulates and HCl near the launch complex for a launch vehicle using solid rockets boosters. No long-term adverse impacts would be expected.	No change in baseline condition.
Water Quality	Water used for pre-launch fire protection, heat suppression, acoustic damping, and post-launch wash down is recovered and treated if necessary. No long-term adverse impacts to groundwater or surface water would be expected; short-term increase in the acidity of nearby surface waters would be expected.	No change in baseline condition.
Offshore Environment	The offshore environments at KSC or CCAFS would be impacted by the jettisoned launch vehicle sections in pre-approved drop zones. Small amounts of residual propellants would be released to the surrounding water. Toxic concentrations would be unlikely because of the slow rate of the corrosion process and the large volume of ocean water available for dilution.	No change in baseline condition.
Biological Resources	Biota near the launch complex could be damaged or killed during launch, although no animal mortality has been observed that could be attributed to previous Delta and Atlas launches. Possible acidification of nearby surface waters from solid propellant exhaust products is not expected to cause any mortality of aquatic biota. No long-term adverse effects would be expected. No short-term or long-term impacts would be expected to threatened or endangered species. No long-term impacts would be expected to critical habitat.	No change in baseline condition.
Socioeconomics	No adverse impacts to socioeconomic factors such as demography, employment, transportation, and public or emergency services.	No change in baseline condition.
Environmental Justice	No disproportionately high and adverse impacts would be expected.	No change in baseline condition.
Cultural/Historical/ Archaeological Resources	No impacts would be expected.	No change in baseline condition.
Global Environment	Not anticipated to adversely affect global climate change. Temporary localized decrease in stratospheric ozone with rapid recovery would be anticipated along the launch vehicle's flight path.	No change in baseline condition.

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The environmental impacts associated with implementing the Proposed Action (Alternative 1), Alternative 2, or Alternative 3 would center largely on the exhaust products emitted from the launch vehicle's strap-on solid rockets and the short-term impacts of those emissions, should a vehicle that uses solid rockets (i.e., one of the Atlas V configurations) be selected. High concentrations of solid rocket motor exhaust products, principally aluminum oxide (Al_2O_3) particulates, carbon monoxide (CO), hydrogen chloride (HCl), nitrogen (N_2), and water (H_2O), would occur in the exhaust cloud that would form at the launch complex. CO would be quickly oxidized to carbon dioxide (CO_2); and at the high exhaust plume temperatures, N_2 may react with oxygen to form nitrogen oxides (NO_X). Due to the relatively high gas temperatures, this exhaust cloud would be buoyant and would rise quickly and begin to disperse near the launch pad. High concentrations of HCl would not be expected, so prolonged acidification of nearby water bodies and long-term or cumulative damage to vegetation should not occur. First-stage liquid propellant engines that use RP-1 and LOx, such as the Atlas V and Falcon Heavy, would primarily produce CO, CO_2, and water vapor as combustion products. First-stage liquid propellant engines that use LH_2 and LOx, such as the Delta IV, would produce water vapor. For either launch vehicle, no adverse impacts to local air quality would be expected.

If rain were to occur shortly after launch, some short-term acidification of nearby water bodies could occur with the accompanying potential for some mortality of aquatic biota. Biota that happened to be in the path of the exhaust could be damaged or killed. Threatened or endangered species would not be jeopardized nor would critical habitats be affected at KSC or CCAFS. As the launch vehicle gains altitude, a portion of the solid rocket motor exhaust (specifically, HCl, Al_2O_3, and NO_X) would be deposited in the stratosphere, resulting in a short-term reduction in ozone along the launch vehicle's flight path. Recovery, however, would be rapid and cumulative impacts would not be expected.

Noise and sonic booms would be associated with the launch. However, neither launch site workers nor the public would be adversely affected. Increased noise levels, anticipated to be below Occupational Safety and Health Administration (OSHA) regulations for unprotected workers, would occur for only a short period during the launch vehicle's early ascent, and would diminish rapidly as the vehicle gains altitude and moves downrange. No impacts to cultural, historical or archaeological resources would be expected from a normal launch. The Mars 2020 mission launch would not be expected to disproportionately impact either minority or low-income populations.

No Action Alternative. Under the No Action Alternative, NASA would discontinue preparations for the Mars 2020 mission, and the spacecraft would not be developed and launched. Thus, none of the anticipated impacts associated with a normal launch would occur.

2.6.2.2. Potential Non-radiological Environmental Impacts of Launch Accidents

Alternatives 1, 2, and 3. As with the impacts associated with a successful launch, these impacts were addressed in the *Final Environmental Assessment for Launch of NASA Routine Payloads on Expendable Launch Vehicles* (Routine Payload EA) (NASA

2011) for all candidate launch vehicles. These impacts were determined to have no significant impact and described in the FONSI for the Routine Payload EA.

Non-radiological accidents could occur during preparation for and launch of the Mars 2020 spacecraft at KSC or CCAFS. The two most significant non-radiological accidents would be a liquid propellant spill associate with fuel loading operations and a launch vehicle accident.

The potential for environmental consequences would be limited primarily to liquid propellant spills of RP-1, LH_2, LOx, and hydrazine (depending on the propellants used in the selected launch vehicle); during fueling operations; and a launch accident at or near the launch pad. USAF safety requirements (USAF 2004) specify detailed policies and procedures to be followed to ensure worker and public safety during liquid propellant fueling operations. Propellant spills or releases of RP-1, LH_2, and LOx would be minimized through remotely operated actions that close applicable valves and safe the propellant loading system. Workers performing propellant loading (e.g., RP-1 and hydrazine) would be equipped with protective clothing and breathing apparatus, and uninvolved workers would be excluded from the area during propellant loading. Propellant loading would occur only shortly before launch, further minimizing the potential for accidents.

A launch vehicle accident on or near the launch area during the first few seconds of flight could result in the release of the propellants (solid and liquid) onboard the launch vehicle and the spacecraft. A launch vehicle accident would result in the prompt combustion of a portion of the liquid propellants, depending on the degree of mixing and ignition sources associated with the accident, and somewhat slower burning of the solid propellant fragments, should a vehicle that uses solid rockets be selected. The resulting emissions would resemble those from a normal launch, consisting principally of CO, CO_2, HCl, NO_X, and Al_2O_3 from the combusted propellants, and depending on the propellants used in the selected launch vehicle. Falling debris would be expected to land on or near the launch pad resulting in potential secondary ground-level explosions and localized fires. After the launch vehicle clears land, debris from an accident would be expected to fall over the Atlantic Ocean. Modeling of accident consequences with meteorological parameters that would result in the greatest concentrations of emissions over land areas indicates that the emissions would not reach levels threatening public health. Some burning solid and liquid propellants could enter surface water bodies and the ocean resulting in short-term, localized degradation of water quality and conditions toxic to aquatic life. Such chemicals entering the ocean would be dispersed and buffered, resulting in little long-term impact on water quality and resident biota.

For suborbital, orbital, and reentry debris, standard safety review processes require that NASA missions comply with the re-entry requirements of NASA Standard 8719.14, *Process for Limiting Orbital Debris*. This NASA Standard (i.e., Requirement 4.7.1) limits the risk of human casualty from reentry debris to 1 in 10,000 and requires that missions be designed to assure that in both controlled and uncontrolled entries, domestic and foreign landmasses are avoided.

The environmental impact of objects falling into the ocean would depend on the physical properties of the materials (e.g., size, composition, quantity, and solubility) and the marine environment of the impact region. Based on past analyses of other space components, it is expected that the environmental impact of reentering orbital debris would be negligible (NASA 2005b; USAF 1998). NASA has studied the potential risks associated with reentry and Earth impact of spacecraft propellant tanks, including those used on prior science missions to the surface of Mars. Specifically, for the MSL spacecraft, an analysis showed that under certain launch accident conditions, there was a small probability the spacecraft with a full propellant load (475 kg) could reenter prior to achieving orbit and impact land in southern Africa or Madagascar. The probability of such an accident occurring and leading to a land impact was determined to be on the order of 1 in 20,000. The overall risk of an individual injury resulting from the land impact of a spacecraft and exposure to hydrazine was determined to be less than 1 in 100,000 (NASA 2010b).

In accident scenarios occurring after achievement of the park orbit, analysis for the MSL spacecraft determined it would be extremely unlikely that there would be any residual hydrazine remaining inside the propellant tanks at the point of ground impact (NASA 2010b).

No Action Alternative. Under the No Action Alternative, a launch would not occur, therefore there would be no potential for either type of accident to occur.

2.6.2.3. Potential Radiological Environmental Impacts of Launch Accidents

This section presents a summary of DOE's *Nuclear Risk Assessment for the Mars 2020 Mission Environmental Impact Statement* (SNL 2014) for the Proposed Action (Alternative 1), Alternative 2, and Alternative 3 as described in this DEIS. More detailed presentations can be found in Sections 4.1.4, 4.1.5, and 4.3.4.

Alternative 1: Figure 2-21 presents summaries of launch-related probabilities for Alternative 1 for the proposed Mars 2020 mission. These probability summaries were derived by combining the estimated failure probabilities from Mars 2020 Representative Data Book (NASA 2013), and DOE's estimated release probabilities (SNL 2014). As such, the estimated probabilities summarized in Figure 2-21 do not reflect the reliability of any single launch vehicle.

The most likely outcome of implementing the proposed Mars 2020 mission, with over a 97% probability, is a successful launch to Mars. The unsuccessful launches (about a 2.5 % probability) would result from either a malfunction or a launch accident. Most malfunctions would involve trajectory control malfunctions, which would occur late in the ascent profile. This type of malfunction would place the spacecraft on an incorrect trajectory escaping from Earth but leading to failure of the spacecraft to reach Mars. Most launch accidents result in destruction of the launch vehicle but would not result in damage to the MMRTG sufficient to cause a release of some plutonium dioxide. The analysis estimates that for less than 0.04% of the time (a probability of 1 in 2,600), a launch could result in an accident with the release of plutonium dioxide, but typically not in a quantity large enough to result in discernible radiological consequences (see Section 2.6.2.3.2).

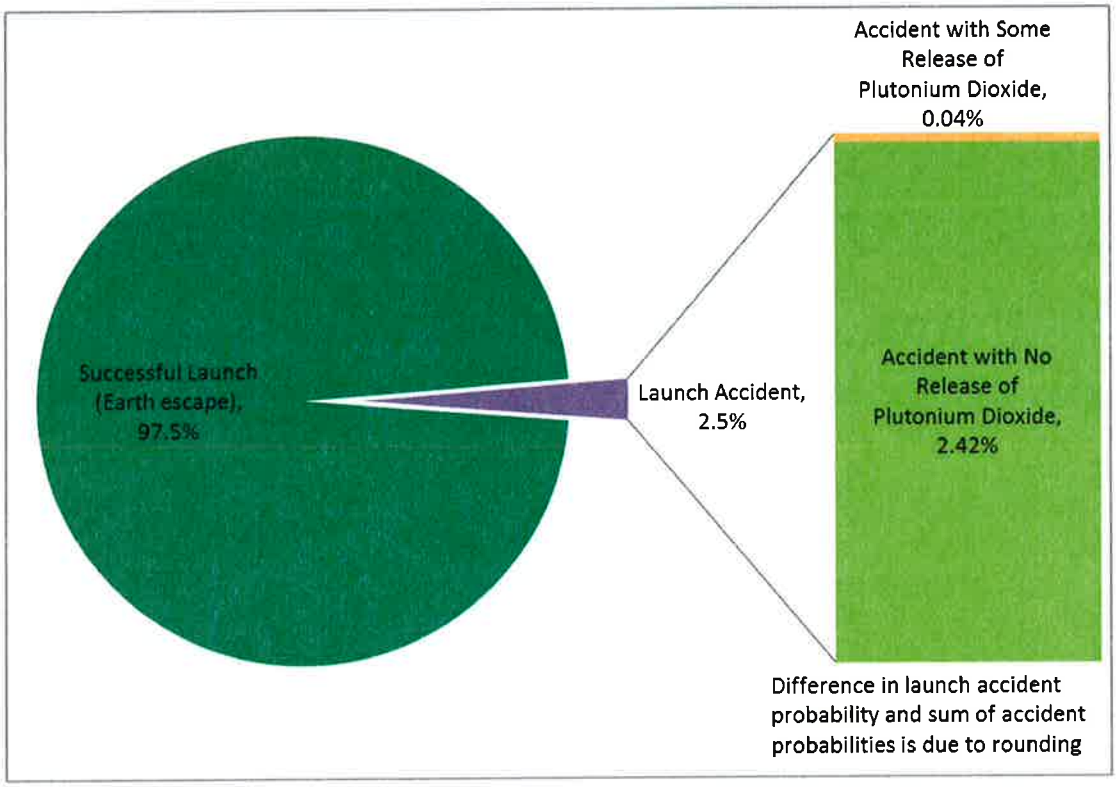

Figure 2-21: Alternative 1 - MMRTG Accident Probabilities

The rover may incorporate science instruments with a small quantity of radioactive sources. NASA has not yet identified the specific instruments that would be used on the Mars 2020 mission. However, DOE has performed a risk assessment using a representative instrument radioisotope source. The results of the instrument source analysis are provided in the following sections and provide a perspective on their relative risks compared to that from the MMRTG or LWRHUs. One significant difference between the small quantity radioactive sources and the plutonium dioxide in the MMRTG is the likelihood of a release following a launch accident. Considering all launch accidents, there is a slightly less than 50% chance that the accident would result in the release of radioactive material from the small quantity of certain radioactive sources. The risks associated with these source terms would be applicable to all three rover configurations (Alternatives 1, 2, and 3).

Alternative 2: For Alternative 2, the rover would rely solely on the power from the solar arrays to provide electric power for rover operations and heat to maintain an acceptable thermal environment for rover equipment and instrumentation. There would be no radioactive material other than the small quantity radioactive sources that may be contained in science instruments that are incorporated into the rover.

Alternative 3: For Alternative 3 the rover would rely upon power from the solar arrays to provide electric power for rover operations and heat to maintain an acceptable thermal environment for rover equipment and instrumentation and incorporate up to 71

LWRHUs[10] as an additional heat source. As with alternatives 1 and 2, Alternative 3 may incorporate science instruments with small quantity radioactive sources.

Figure 2-22 presents summaries of launch-related probabilities for Alternative 3 of the proposed Mars 2020 mission. These probability summaries were derived by combining the estimated failure probabilities from *Mars 2020 Representative Data Book* (NASA 2013), and DOE's estimated release probabilities (SNL 2014). As such, the estimated probabilities summarized in Figure 2-22 do not reflect the reliability of any single launch vehicle.

The differences between the three rover configurations (MMRTG powered, solar-powered with no LWRHUs, and solar-powered with LWRHUs) do not significantly impact the accident probability for the mission. However, the probability of an accident with a release of plutonium dioxide is smaller, 0.006% (1 in 15,000), for the solar-powered rover with LWRHUs configuration, than for the MMRTG powered rover. The amount of material released is typically not large enough to result in discernible radiological consequences. (See Section 2.6.2.3.2)

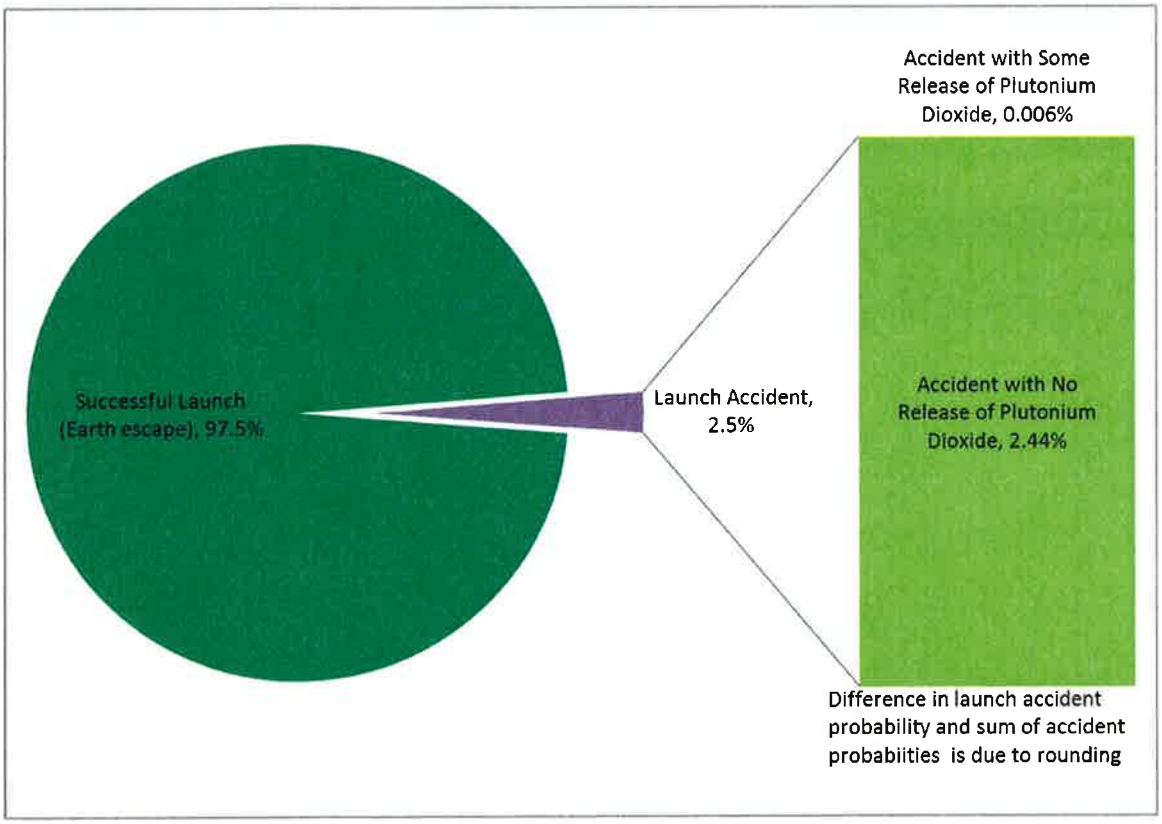

Figure 2-22: Alternative 3 - LWRHU Accident Probabilities

[10] For the purposes of the risk analysis, DOE assumed the rover could include up to 80 LWRHUs

2.6.2.3.1. The DEIS Nuclear Risk Assessment

The nuclear risk assessment for the proposed Mars 2020 mission considers (1) potential accidents associated with the launch and their probabilities and accident environments; (2) the response of the MMRTG and LWRHUs to such accidents in terms of the amount of radioactive materials released and their probabilities; and (3) the radiological consequences and mission risks associated with such releases. The risk assessment was based on a typical MMRTG radioactive material inventory of about 60,000 Ci of primarily plutonium-238 (an alpha-emitter with an 87.7 year half-life).

DOE's risk assessment was developed when the candidate launch vehicles being considered by NASA for the Mars 2020 mission were the Atlas V 541 and 551, the Delta IV Heavy, and the Falcon Heavy. A composite approach was taken in DOE's nuclear risk assessment (SNL 2014) for accident probabilities, potential releases of plutonium dioxide in case of an accident (called source terms), radiological consequences, and mission risks. The composite approach taken in the risk assessment and reported in this DEIS reflects the state of knowledge at this early stage in the mission with respect to the candidate launch vehicles.

The risk assessment for the Mars 2020 mission began with the identification of the initial launch vehicle system malfunctions or failures and the subsequent chain of accident events that could ultimately lead to the accident environments (e.g., explosive overpressures, fragments, fire) that could threaten the MMRTG or LWRHUs. These launch vehicle system failures were based on launch vehicle system reliabilities and estimated failure probabilities (NASA 2013).

Failure of the launch vehicle has the potential to create accident environments that could damage the MMRTG or LWRHUs and result in the release of plutonium dioxide. Based on analyses performed for earlier missions that carried radioisotope devices (RTGs and LWRHUs), DOE identified the specific accident environments that could potentially threaten these devices. DOE then determined the response of the MMRTG, MMRTG components, and LWRHUs to these accident environments and estimated the amount of radioactive material that could be released.

For this risk assessment, the Mars 2020 mission was divided into mission phases, which reflect principal launch events.

- **Phase 0 (Pre-Launch) and Phase 1 (Early Launch):** A launch-related accident during these periods could result in ground impact in the launch area.
- **Phase 2 (Late Launch):** A launch accident during this period would lead to impact of debris in the Atlantic Ocean.
- **Phase 3 (Sub Orbital):** A launch accident during this period prior to reaching Earth parking orbit could lead to prompt sub-orbital reentry within minutes.
- **Phase 4 (Orbital) and Phase 5 (Long-Term Reentry):** A launch accident that occurs after attaining parking orbit could result in orbital decay reentries from minutes to years after the accident.

2.6.2.3.2. Accident Probabilities and Consequences

Section 4.1.4 provides a detailed quantitative discussion of the accident probabilities and associated potential consequences for the proposed Mars 2020 mission.

The radiological consequences of a given accident that results in a release of radioactive material have been calculated in terms of radiation doses, potential health effects, and land area contaminated at or above specified levels. The radiological consequences have been determined from atmospheric transport and dispersion simulations incorporating both worldwide and launch-site specific meteorological and population data.

Sections 4.1.4 and 4.1.5 (Alternative 1), and 4.3.4 (Alternative 3) describe the risk assessment in greater detail, with the results presented for both mean and 99th percentile values. For the purposes of this summary, the accident consequences and associated risks are presented only in terms of the mean.

Consequences of Radiological Release on Human Health

Human health consequences are expressed in terms of maximum individual dose, collective dose to the potentially exposed population, and the associated health effects. The maximum individual dose is the maximum dose, typically expressed in units of rem (Roentgen equivalent in man), delivered to a single individual assumed to be outside during the time of radiological exposure for each accident. Collective dose (also called a population dose) is the sum of the radiation dose received by all individuals exposed to radiation from a given release. Health effects represent statistically estimated additional latent cancer fatalities resulting from an exposure over a 50-year period to a release of radioactive material, and are determined based on Interagency Steering Committee on Radiation Standards (ISCORS) health effects estimators (DOE 2002). The estimated radiological consequences by mission phase and for the overall mission are summarized below.

Alternative 1: For alternative 1, an accident resulting in the release of plutonium dioxide from the MMRTG occurs with a probability of 1 in 2,600. The mean mission human health consequences are:

- maximum dose received by an individual would have a mean of 0.016 rem which is equivalent to about 5% of the natural annual background dose received by each member of the population of the United States during a year[11]
- a mean collective dose resulting in about 0.076 additional latent cancer fatalities within the entire group of potentially exposed individuals.

For individual phases of the mission, the maximum dose received by an individual ranges from 0.000016 to 0.060 rem, and the additional latent cancer fatalities range

[11] An average of about 0.3 rem per year is received by an individual in the United States from natural sources. The dose from man-made sources, such as medical diagnosis and therapy, could be as high as an additional 0.3 rem. See Section 3.2.6 for further information.

from 0.000078 to 0.29. The largest values are both associated with accidents with releases that occur during the Early Launch Phase (Phase 1).

Alternative 3. For alternative 3, an accident resulting in the release of plutonium dioxide from the LWRHUs occurs with a probability of 1 in 15,000. The mean mission human health consequences are:

- maximum dose received by an individual would have a mean of 0.0041 rem which is equivalent to about 1% of the natural annual background dose received by each member of the population of the United States during a year
- a mean collective dose resulting in about 0.020 additional latent cancer fatalities within the entire group of potentially exposed individuals.

For individual phases of the mission in which accidents can result in a plutonium dioxide release, the maximum dose received by an individual ranges from 0.0013 to 0.0042 rem. and the additional latent cancer fatalities range from 0.006 to 0.020. Accidents occurring during phases 2, 4, and 5 are not expected to release any plutonium dioxide. The largest values are both associated with accidents with releases that occur during the Early Launch Phase (Phase 1).

Alternatives 1, 2, and 3. For alternatives 1, 2, and 3, the probability of an accident resulting in the release of some of the instrumentation radioisotope is about 1 in 87. The mean mission human health consequences are:

- maximum dose received by an individual would have a mean of 0.00003 rem which is equivalent to about one-hundredth of 1% of the natural annual background dose received by each member of the population of the United States during a year,
- a mean collective dose resulting in about 0.00014 additional latent cancer fatalities within the entire group of potentially exposed individuals.

For individual phases of the mission, the maximum dose received by an individual ranges from of 0.000011 to of 0.000061 rem and the additional latent cancer fatalities range from of 0.000053 to 0.00029. The largest values are both associated with accidents with releases that occur during the long-term reentry phase (Phase 5).

In summary, for accidents in and near the launch area (Phases 0 and 1), as well as Phase 3 and Phase 4 accidents, the mean health effects are estimated to be small within the potentially exposed population. This estimate assumes no intervention (mitigation), such as sheltering and exclusion of people from contaminated land areas.

Also, the predicted mean maximum radiological dose to an individual within the exposed population (i.e., the maximally exposed individual) ranges from very small to less than a rem for all accidents with a release. None of these potential exposures would lead to short-term radiological effects, only to a statistical increase in the likelihood of cancer.

Table 2-11 provides a summary of the human health consequences for all mission phases for each alternative.

Should the mission be delayed, the proposed Mars 2020 mission would be launched during the next available launch opportunity in August through September 2022. Since this launch period is in a similar season as the 2020 launch period, the projected radiological impacts would be similar, with only a small increase in population impacts due to population growth. Thus, within the overall uncertainties, the radiological impacts associated with a 2022 launch would be the same as those for the proposed 2020 launch.

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Table 2-11. Summary of Estimated Mean Radiological Health Consequences

Consequence Contributing Source		Launch Area Accidents		Accidents Beyond The Launch Area				Overall Mission Accidents
		Pre Launch	Early Launch	Late Launch	Sub-Orbital	Orbital	Long-term reentry	
Alternative 1 - MMRTG	Probability of an Accident with a Release	1 in 93,000	1 in 11,000	1 in 130,000	1 in 67,000	1 in 3,800	1 in 11,000,000	1 in 2,600
	Maximum Individual Dose, rem	0.00029	0.06	0.000016	0.043	0.0005	0.0008	0.016
	Latent Cancer Fatalities[(a)]	0.0014	0.29	0.000078	0.20	0.0026	0.0038	0.076
Alternative 3 - LWRHUs	Probability of an Accident with a Release	1 in 3,200,000	1 in 16,000	0[(b)]	1 in 430,000	0[(b)]	0[(b)]	1 in 15,000
	Maximum Individual Dose, rem	0.0030	0.0042	-	0.0013	-	-	0.0041
	Latent Cancer Fatalities	0.015	0.020	-	0.0060	-	-	0.020
Alternatives 1, 2 and 3 – Instrumentation Source	Probability of an Accident with a Release	1 in 550,000	1 in 1,700	1 in 8,500	1 in 150	1 in 240	1 in 1,000,000	1 in 87
	Maximum Individual Dose, rem	0.000034	0.000011	0.000031	0.000031	0.000031	0.000061	0.000030
	Latent Cancer Fatalities	0.00016	0.000053	0.00015	0.00015	0.00015	0.00029	0.00014

(a) A latent cancer fatality of less than 1.0 can be interpreted as the probability of the occurrence of one or more latent cancer fatalities. For example, a value of 0.25 would be a one in four chance that the accident would result in one or more latent cancer fatalities.

(b) The multiple protective layers of the LWRHUs would be sufficient to prevent the release of fuel under all circumstances during these types of launch accidents. Therefore, the release probability is 0.0 and there are no associated radiological consequences as indicated by the "-."

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Impacts of Radiological Releases on the Environment

In addition to the potential human health consequences of launch accidents that could result in a release of plutonium dioxide, environmental impacts could also include contamination of natural vegetation, wetlands, agricultural land, cultural, archaeological and historic sites, urban areas, inland water, and the ocean, as well as impacts on wildlife.

Potential environmental contamination was evaluated in terms of areas exceeding various screening levels and dose-rate-related criteria considered in evaluating the need for land cleanup following radioactive contamination. In the risk assessment for this DEIS, land areas which could be contaminated at or above a level of 0.2 microcuries per square meter ($\mu Ci/m^2$) have been identified. This is a screening level used in prior NASA environmental documentation (e.g., NASA 1989, NASA 1997, NASA 2005b, NASA 2006 (MSL EIS)) to identify areas potentially needing further action, such as monitoring or cleanup. The results for the mean land area contaminated at or above a level of 0.2 $\mu Ci/m^2$ are summarized in Table 2-12.

Table 2-12. Mars 2020 Mission Alternatives: Land Contamination

Phase	MMRTG (Alternative 1)		LWRHU (Alternative 3)		Science Sources (Alternatives 1, 2, & 3)	
	Release Probability[a]	Land Contamination	Release Probability [a]	Land Contamination	Release Probability[a]	Land Contamination
Pre-launch	1 in 93,000	0.035 km^2 (0.014 mi^2)	1in 3.3 million	0.37 km^2 (0.14 mi^2)	1in 550,000	0.0041 km^2 (0.0016 mi^2)
Early launch	1 in 11,000	7.4 km^2 (2.9 mi^2)	1 in 16,000	0.51 km^2 (0.20 mi^2)	1 in 1,700	0.0014 km^2 (0.0005 mi^2)
Late launch	1 in 130,000	0.0020 km^2 (0.00077mi^2)	0	----	1 in 8,500	0.0038 km^2 (0.0015 mi^2)
Sub-Orbital	1 in 68,000	5.2 km^2 (2.0 mi^2)	1 in 430,000	0.15 km^2 (0.058 mi^2)	1 in 152	0.0038 km^2 (0.0015 mi^2)
Orbital	1 in 3,800	0.066 km^2 (0.025 mi^2)	0	----	1 in 240	0.0038 km^2 (0.0015 mi^2)
Long-term Reentry	1 in 11 million	0.097 km^2 (0.037 mi^2)	0	----	1 in 1 million	0.0075 km^2 (0.0029 mi^2)
Overall Mission	1 in 2,600	1.94 km^2 (0.75 mi^2)	1 in 15,000	0.50 km^2 (0.19 mi^2)	1 in 87	0.0036 km^2 (0.0014 mi^2)

a) Probability of an accident with a radionuclide release. A value of '0' indicates that there are no accidents that result in a release and therefore no corresponding land contamination (---); the multiple protective layers of the LWRHUs would be sufficient to prevent the release of fuel under all circumstances during these types of launch accidents.

For alternatives 1, 2, and 3, costs associated with potential characterization and cleanup, should decontamination be required, could vary widely ($110 million to $600 million per km^2 or about $285 million to $1.6 billion per mi^2) depending upon the characteristics and size of the contaminated area. The Price-Anderson Act of 1957, as amended (42 U.S.C. 2210), governs liability and compensation in the event of a nuclear incident arising out of the activities of the DOE. In the case of the Mars 2020 mission, DOE retains responsibility for the MMRTG or LWRHUs. The MMRTG or LWRHUs would, therefore, be subject to Price-Anderson Act provisions. In the unlikely event that an accident were to occur resulting in release of plutonium dioxide, affected property owners within or outside the United States would be eligible for reimbursement for loss of property due to contamination.

In addition to the potential direct costs of radiological surveys, monitoring, and potential cleanup following an accident, there are potential secondary societal costs associated with the decontamination and mitigation activities due to launch area accidents. Those costs may include: temporary or longer term relocation of residents; temporary or longer term loss of employment; destruction or quarantine of agricultural products, including citrus crops; land use restrictions; restriction or bans on commercial fishing; and public health effects and medical care.

The areas that could be contaminated to the extent that these secondary costs would be incurred are not necessarily the same as the area contaminated above 0.2 $\mu Ci/m^2$. For example, the Food and Drug Administration has provided guidelines for crop contamination intended to ensure contaminated foodstuffs would not endanger the health and safety of the public. These guidelines, in the form of Derived Intervention Levels (DILs) identify the level of contamination above which some action (decontamination, destruction, quarantine, etc.) is required. For potential launch area accidents, DOE has estimated that the crop area contaminated above the DIL would be over 50 times smaller than the area contaminated above 0.2 $\mu Ci/m^2$.

2.6.2.3.3. Mission Risks

To place the estimates of potential health effects due to launch accidents for the proposed Mars 2020 mission into a perspective that can be compared with other human undertakings and events, it is useful to use the concept of risk. Risk is commonly viewed as the possibility of harm or damage. For the Mars 2020 mission, public risk is characterized in terms of the expectation of health effects in a statistical sense. The risk for each mission phase and for the overall mission is estimated by multiplying the total probability of a release by the health effects resulting from that release. Risk calculated in this manner can also be interpreted as the probability of one or more health effects occurring in the exposed population.

Population Risks

For Alternative 1 of the Mars 2020 mission, overall population health effects risk from the release of plutonium dioxide is estimated to be about 1 in 34,000—that is, one chance in 34,000 of an additional health effect. For accidents that may occur in the launch area, not everyone within 100 km (62 mi) of the launch site would be potentially exposed. Who would be potentially exposed is dependent upon several factors, including the weather conditions at the time of the accident. The total probability of a health effect within the regional population is about 1 in 61,000, or about 57% of the total risk for the overall mission. For the global population (excluding those exposed in the launch area region) the risk would be due to the potential for accidental release occurring from pre-Launch through Mars trajectory insertion and was estimated to be about 1 in 79,000, or about 43% of the total risk for the mission.

For Alternative 3 of the Mars 2020 mission, overall population health effects risk from the release of plutonium dioxide is estimated to be about 1 in 790,000. For accidents that may occur in the launch area, only a portion of the total population within 100 km (62 mi) of the launch site would be potentially exposed. The total probability of a health effect within the regional population is about 1 in 1,200,000, or about 64% of the total risk for the overall mission. For the global population (excluding those exposed in the launch area region) the risk would be due to the potential for accidental release occurring from pre-Launch through Mars trajectory insertion and was estimated to be about 1 in 2,200,000, or about 36% of the total risk for the mission.

Alternatives 1, 2, and 3 may include science instrumentation that could include small radioactive sources. The results discussed above include only the risks due to the plutonium dioxide in either the MMRTG or in LWRHUs and do not include the risks associated with these small sources. As stated previously, the science instruments have not been selected for a Mars 2020 mission; however, DOE performed a risk assessment for a representative small radioactive source.

The overall radiological risk from the instrument small source radioisotopes for the Mars 2020 mission is estimated to be about 1 in 610,000. The global risks due to accidents in all mission phases would be over 98% of the total risk. The contribution to risk within 100 km (62 mi) of the launch site would be about 1.4% of the total risk for the mission.

For Alternative 1, this would increase the overall mission risk to 1 in 33,000, and for Alternative 3 to 1 in 340,000.

Individual Risks (Maximum Individual Risks)

Those individuals within the population that might receive the highest radiation exposures, such as those very close to the launch area, would face very small risks. The risk to the maximally exposed individual within the regional population is estimated to be less than 1 in several million for all alternatives considered for the Mars 2020 mission. Most people in the potentially exposed population would have much lower risks.

These risk estimates are small compared to other risks. Annual fatality statistics indicate that in the year 2010 the average individual risk of accidental death in the United States was about 1 in 2,600 per year, while the average individual risk of death due to any disease, including cancer, was about 1 in 130 (see Section 4.1.4.7 of this DEIS for additional details).

2.6.3 Summary Comparison of the Alternatives

Table 2-13 presents a summary comparison of the Proposed Action (Alternative 1), Alternative 2, Alternative 3, and the No Action Alternative in terms of each alternative's capabilities for operating and conducting science on the surface of Mars, the anticipated environmental impacts of normal implementation (i.e., a successful launch to Mars) of each alternative, and the potential environmental impacts in the event of an unlikely launch accident for each alternative.

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Table 2-13. Summary Comparison of the Mars 2020 Mission Alternatives

	Proposed Action (Alternative 1)	Alternative 2	Alternative 3	No Action Alternative
Rover Power Alternative	MMRTG	Solar Array no LWRHUs	Solar Array with LWRHUs	Not applicable
Functional Capability	Capable of operating for at least one Mars year at landing sites between 30° north and 30° south latitudes on Mars	Unable to operate for a full year at any latitude[(a)]	Limited lifetime capability for operating at landing sites between 20° south and 5° south latitudes on Mars[(a)]	Not applicable
Science Capability (detailed comparison in Table 2-9)	Capable of accomplishing all science objectives at any scientifically desirable landing site between 30° north and 30° south latitudes	Capable of accomplishing up to 33% of science objectives during partial year operation[(b)]	Capable of accomplishing up to 70% of science objectives due to constrained operations during winter.[(b)]	No science achieved
Anticipated Environmental Impacts	Short-term impacts associated with exhaust emissions from the launch vehicle during a normal launch	Short-term impacts associated with exhaust emissions from the launch vehicle during a normal launch	Short-term impacts associated with exhaust emissions from the launch vehicle during a normal launch	No impacts
Potential Environmental Impacts in the Event of a Launch Accident (detail comparison in Tables 2-11 and 2-12)	Potential impacts associated with combustion of released propellants and falling debris Potential radiological impacts associated with release of small quantity radioisotopes from science instruments and release of some of the PuO_2 from the MMRTG	Potential impacts associated with combustion of released propellants and falling debris Potential radiological impacts associated with release of small quantity radioisotopes from science instruments	Potential impacts associated with combustion of released propellants and falling debris Potential radiological impacts associated with release of small quantity radioisotopes from science instruments and release of some of the PuO_2 from the LWRHUs	No potential impacts

(a) These numbers assume a dust factor of 40%. Assuming dust mitigation technology improvements on the MER solar array performance, the rover (without LWRHUs) is predicted to survive for a full year at latitudes between 0° and 5° south and, with LWRHUs, is predicted to survive for a full year at latitudes between 20° south and 15° north.

(b) Improved solar array performance from dust mitigation technology would result in a corresponding increase in science capability, expanding the range of latitudes the rover could operate for a full year.

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In terms of operational capabilities, the major difference between the Proposed Action (Alternative 1), Alternative 2, and Alternative 3 is the length of time the rover would be expected to survive and successfully operate and conduct science experiments at a selected landing site. The capability to operate the rover within a broad range of latitudes is important because doing so maintains NASA's flexibility to select the most scientifically interesting location on the surface and fulfill the purpose and need for the Mars 2020 mission as discussed in Chapter 1 of this DEIS. The No Action Alternative would not fulfill the purpose and need for the Mars 2020 mission.

In terms of environmental impacts, normal implementation of either the Proposed Action (Alternative 1) or Alternative 2 or 3 would primarily yield short-term impacts to air quality from the launch vehicle's exhaust (see Section 2.6.2.1). Should an unlikely launch accident occur for either of these alternatives, potential environmental impacts would be primarily associated with combustion products from released propellants and from falling debris (see Section 2.6.2.2). For the Proposed Action (Alternative 1), an unlikely launch accident could result in a release of some of the plutonium dioxide from the MMRTG, which could potentially result in consequences to human health and the environment (see Section 2.5.2.3). Similarly in Alternative 3, plutonium dioxide could be released from LWRHUs (see Section 2.6.2.3). For Alternative 1, 2, and 3, during these accidents, releases of the small quantity source terms could also result in consequences to human health and the environment. With the No Action Alternative, no environmental impacts would occur since there would be no launch, but none of the planned science would be achieved.

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