2. ALTERNATIVES, INCLUDING THE PROPOSED ACTION

2.1 ALTERNATIVES EVALUATED

This Final Environmental Impact Statement (FEIS) for the Cassini mission examines mission alternatives available for accomplishing the mission objectives within a reasonable timeframe, as well as the No-Action alternative. In the course of developing the mission alternatives, three major mission components-launch vehicles, mission trajectories to Saturn, and spacecraft electrical power sources-were examined in detail (JPL 1993a, JPL 1993f, JPL 1994a). These three mission components are the principal factors influencing the development of feasible mission designs (mission alternatives) that would allow the Cassini spacecraft to obtain at least the minimum acceptable science return and the overall mission objectives. These components are also the factors determining the potential environmental impacts associated with each mission alternative under normal (incident-free) and accident conditions. Section 2.6 summarizes the evaluations of these three major mission components and their availability in determining the mission alternatives.

The Cassini mission would continue the evolution of the National Aeronautics and Space Administration's (NASA) program for solar system exploration from reconnaissance-level or flyby missions of the outer planets to more detailed exploration missions. These exploration missions involve a wider range and a greater amount of scientific data that are much more difficult to gather than was the case for the earlier flyby missions. The range and amount of data collected by an exploration mission requires a power source to operate reliably over a long period of time. Gathering data for an outer space mission in the vicinity of Saturn where the Sun's intensity is only 1 percent of that experienced in Earth orbit requires a power source that can operate at vast distances from the Sun. These electrical power requirements must be fulfilled using a spacecraft design within the limited lift capacities of available launch vehicles.

Only a few combinations of launch vehicles, mission trajectories, and spacecraft electrical power sources can meet the requirements for the Cassini mission. The feasible launch vehicles evaluated include the most powerful U.S. launch vehicle, the Titan IV (solid rocket motor upgrade [SRMU])/Centaur and the Space Shuttle (i.e., Space Transportation System [STS]). If the new SRMU-equipped Titan IV could not be used, a Titan IV (SRM)/Centaur could be used. Mission trajectories include Earth- and non-Earth-Gravity-Assist trajectories to Saturn. Power system performance criteria require the use of the most reliable electrical power system capable of providing the large amounts of power needed over an extended period of time-the radioisotope thermoelectric generator (RTG). This EIS examines in detail the feasible components that combine to form the following mission alternatives:

- Proposed Action-NASA proposes to prepare for and implement the Cassini mission to collect scientific data from Saturn, its atmosphere, moons, rings, and magnetosphere. The spacecraft would be launched and inserted into a Venus-Venus-Earth-Jupiter-Gravity-Assist (VVEJGA) trajectory to Saturn during the primary opportunity in October 1997 onboard a Titan IV (SRMU or SRM)/Centaur. A secondary launch opportunity exists in December 1997, with a backup opportunity in March 1999, both using a Venus-Earth-Earth-Gravity-

Assist (VEEGA) trajectory. The Proposed Action would enable the Cassini spacecraft to gather the full science return (i.e., data) desired to accomplish each of the mission science objectives. Achievement of the science objectives for the contingency launches would be essentially the same as for the primary launch opportunity but with some reduction in science return. In the event that a Titan IV (SRMU)/Centaur were unavailable, a Titan IV (SRM)/Centaur would be used with the same primary, secondary, and backup launch opportunities. The science objectives would be achieved but with reduced science return. The launch site for both the primary and contingency launch opportunities would be either Launch Complex 40 or 41 located at Cape Canaveral Air Station (CCAS) in Florida.

- 1999 Mission Alternative - For this mission alternative, preparations for and implementation of the Cassini mission to Saturn would involve dual manned Shuttle launches in 1999: one launch to predeploy an upper stage into Earth orbit and a second launch, separated by 21 to 51 days, to deliver the remaining upper stage(s) and the Cassini spacecraft into low Earth orbit. An on-orbit assembly of the upper stages with the spacecraft would occur, followed by upper stage ignition and insertion of the Cassini spacecraft in March 1999 into its 9.8-year VEEGA interplanetary trajectory. A backup launch opportunity, also a VEEGA, occurs in August 2000. This mission alternative, including both the primary and backup launch opportunities, would obtain less science return than the Titan IV (SRMU)/Centaur 1997 primary launch opportunity. The launch site for the primary and backup launch opportunities would be either Launch Pad 39A or 39B located at Kennedy Space Center (KSC) in Florida.

- 2001 Mission Alternative - This mission alternative consists of preparing for and implementing the Cassini mission to Saturn with a primary launch opportunity in March 2001. Launched from CCAS on a Titan IV (SRMU)/Centaur, the primary launch opportunity would place the spacecraft into a 10.3-year Venus-Venus-Venus-Gravity-Assist (VVVGA) trajectory to Saturn. There is no non-Earth-Gravity-Assist backup opportunity capable of meeting the science objectives that can be performed by a U.S. launch vehicle. However, a launch vehicle configuration has been identified that can perform an Earth-Gravity-Assist (EGA) trajectory. The Titan IV (SRMU)/Centaur could place the Cassini spacecraft into a VEEGA trajectory during a backup launch opportunity in May 2002. To perform the VVVGA trajectory, the Cassini spacecraft would require about 20 percent additional propellant and use a different spacecraft propulsion engine, a rhenium engine, to provide greater efficiency and higher performance. Even with the additional propellant and the high performance rhenium engine, limitations in the available propellant for spacecraft maneuvering at Saturn would restrict the acquisition of the desired amount of science return (JPL 1993i). However, the minimum acceptable level of science objectives for the mission could still be met.

- No-Action Alternative - Under the No-Action alternative, preparations for the launch would cease and the mission would not be implemented.

Figure 2-1 provides an overview of these alternatives. Sections 2.2, 2.3, 2.4, and 2.5 describe the alternatives in greater detail.

Section 2.6 summarizes the results of a complete evaluation of the launch vehicles, mission trajectories, and spacecraft electrical power systems, including components determined to be infeasible (JPL 1993a, JPL 1993f, JPL 1994a). Section 2.7 compares the mission alternatives evaluated, and Section 2.8 provides a brief overall summary.

2.2 DESCRIPTION OF THE PROPOSED ACTION

2.2.1 Mission Design

2.2.1.1 SRMU-Equipped Titan IV Configuration

The primary launch opportunity of the Cassini mission occurs within a 25-day launch period beginning October 6, 1997, and closing October 30, 1997 (JPL 1993a). Using the Titan IV (SRMU)/Centaur, described in Section 2.2.6, the spacecraft would be launched and injected into the 6.7-year VVEJGA interplanetary trajectory to Saturn, as shown in Figure 2-2.

After the spacecraft's launch and injection into interplanetary trajectory in October 1997, it would swing by the planet Venus for the first time in April 1998. Following a maneuver in December 1998, the spacecraft would be placed on a course for a second Venus swingby in June 1999. Because of the Earth's unique orientation relative to Venus during this time period, the spacecraft would fly on to Earth in slightly less than 2 months, where it would obtain its third planetary gravity-assist in August 1999. After flying past the Earth, the spacecraft would pass through the asteroid belt. The spacecraft would obtain a fourth and final gravity-assist at Jupiter in December 2000 before proceeding to Saturn. With these swingbys of Venus, Earth, and Jupiter, the spacecraft would gain sufficient velocity to reach Saturn.

For several months before arriving at Saturn in June 2004, the spacecraft would perform scientific observations of the planet prior to executing the Saturn Orbit Insertion (SOI) maneuver. This Saturn arrival date would provide the unique opportunity to have a distant flyby of Saturn's outer satellite Phoebe, 19 days before SOI. The SOI would place the spacecraft in a large elliptical orbit around Saturn. During the S0I, the spacecraft would be about 1.3 Saturn radii from the planet's center, its closest distance during the entire mission. This presents a unique opportunity to observe the inner regions of Saturn's ring system and magnetosphere; the 1.5-hour orbital insertion burn would be delayed from its optimal point to permit such observations (JPL 1993a).

Approximately three-quarters of the way around the SOI orbit and 3 weeks before Cassini's first flyby of Titan, the spacecraft would release the Huygens Probe on a trajectory for entry into Titan's atmosphere. Two days after release of the Probe, the Orbiter (i.e., the spacecraft without the Probe) would perform a deflection maneuver to be in position to receive scientific information gathered by the Probe during its estimated 2.5 hour parachute descent to Titan's surface. The data transmitted by the Probe would be stored on the Orbiter for later playback to Earth (JPL 1993c).

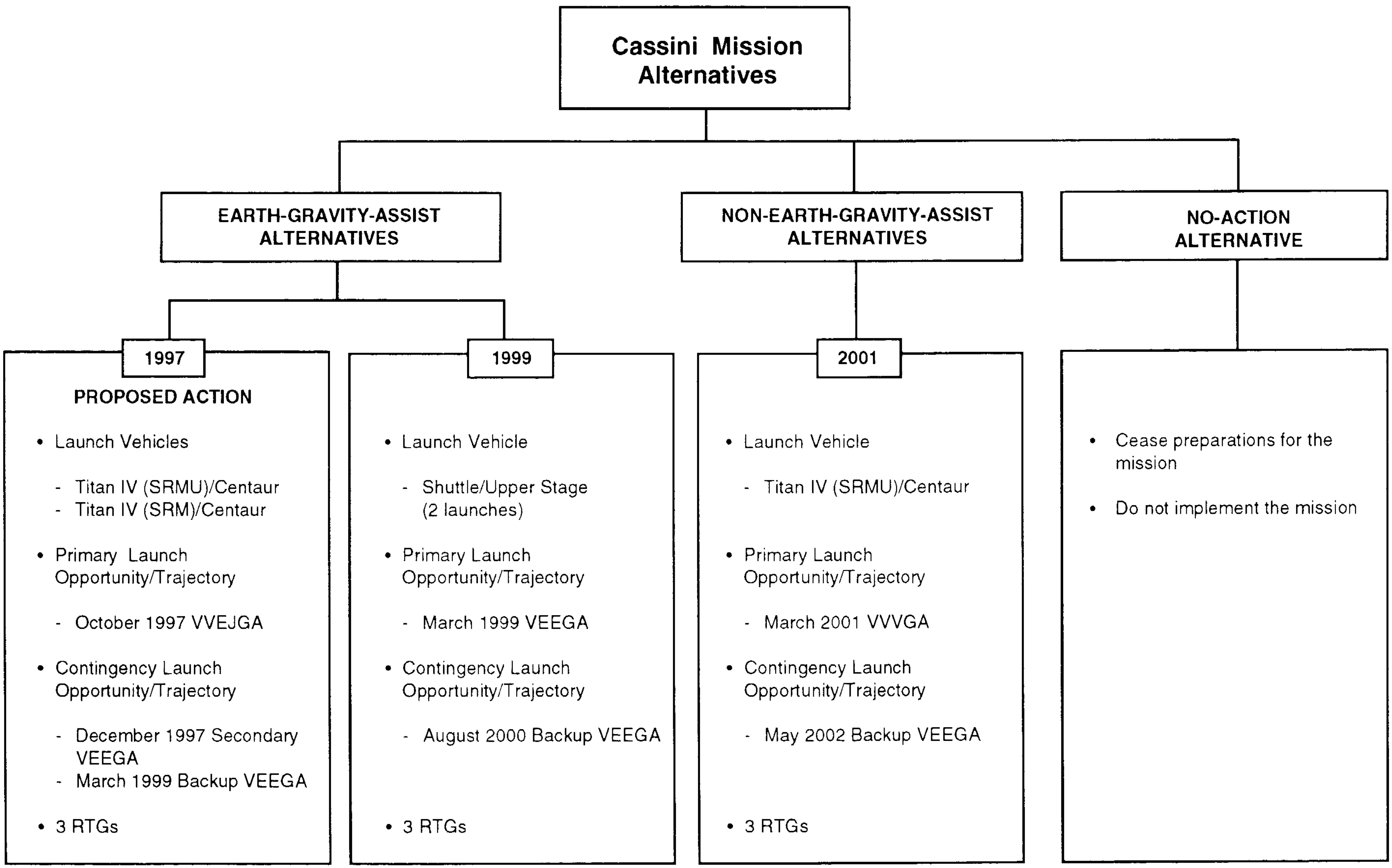
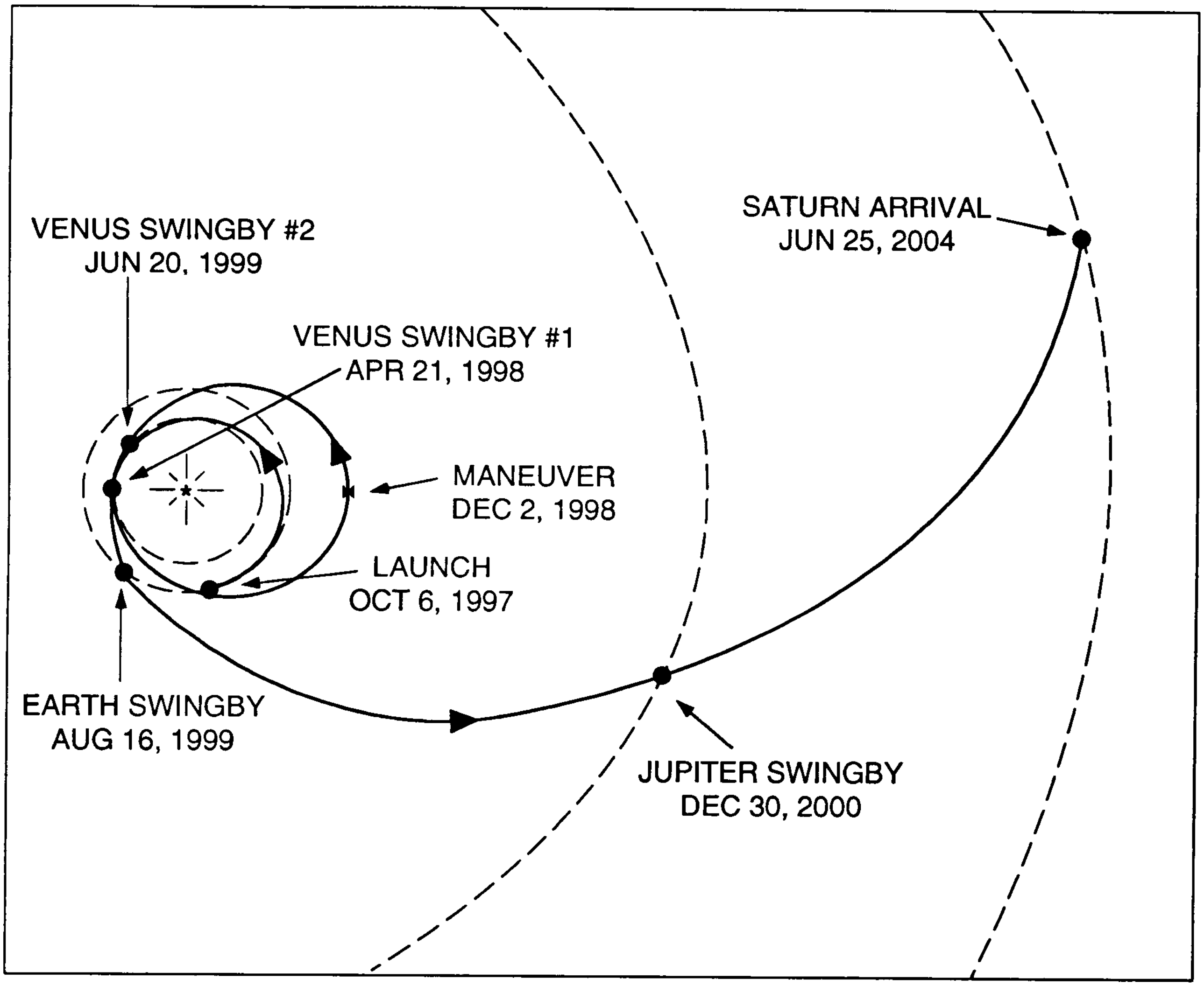


FIGURE 2-1. ALTERNATIVES EVALUATED FOR THE CASSINI MISSION



Source: JPL 1993a

FIGURE 2-2. CASSINI OCTOBER 1997 VVEJGA INTERPLANETARY TRAJECTORY

The spacecraft would then continue on its Saturn orbital tour, providing opportunities for ring imaging; magnetospheric coverage; and radio (Earth), solar, and stellar occultations of Saturn, Titan, and the ring system. A total of 35 close Titan flybys has been planned during the 4-year tour. The spacecraft would use repeated gravity-assist flybys of Titan and associated trajectory correction maneuvers to shape the trajectory. The spacecraft would also be targeted for 4 close flybys of selected icy satellites and would make 29 more distant satellite encounters. By the end of the 4-year tour, the orbital inclination would have been increased to approximately 80 degrees, allowing investigation from higher latitudes. In particular, the spacecraft would investigate the source of the unique Saturn Kilometric Radiation. Although the 10.7-year nominal mission would end in June 2008 (JPL 1993a), science data-gathering activities could continue, limited only by the remaining propellant and mission costs. The spacecraft would continue to orbit Saturn.

2.2.1.2 SRM-Equipped Titan IV Configuration

If the Titan IV SRMU were not available, the mission could be accomplished using the conventional Titan IV SRM configuration. The mission design in this case would be limited by the smaller mass injection capabilities of the SRMs. As currently designed (with the SRMUs), much of the mass that would be injected into the interplanetary trajectory would come from the propellant required for maneuvers during the mission. Reducing the propellant mass onboard the spacecraft would be required to allow a viable mission with this less capable launch configuration. The reduction in spacecraft propellant mass would require a reduction in maneuver activity by the spacecraft upon arrival at Saturn (JPL 1993c).

Specifically, the maneuver activity would be reduced in four major areas. First, the Saturn arrival date would be delayed by 5 months, from June (for the SRMU launch) to November 2004. The delay would decrease the propellant required by the SOI burn because the spacecraft's relative velocity at Saturn would be lower than that for the SRMU launch. Second, the initial orbit period would be increased, resulting in a further reduction of propellant required for SOI. Third, fewer Titan flybys would be planned to reduce the amount of fuel required to correct navigational errors in the trajectory. Fourth, the SOI burn would be centered optimally about the closest approach to Saturn, further reducing the SOI science return JPL 1993c). The adjustments that would be made to compensate for the reduced injection capability and lower spacecraft bipropellant load necessitated by use of the SRMs are shown in Table 2-1.

TABLE 2-1. CASSINI MISSION CHARACTERISTICS FOR THE PROPOSED ACTION'S PRIMARY LAUNCH OPPORTUNITY USING EITHER A TITAN IV (SRMU)/CENTAUR OR TITAN IV (SRM)/CENTAUR

Mission Characteristics	SRMU	SRM
Launch Period	10/06/97 to 10/30/97	10/13/97 to 10/30/97
Bipropellant (kg [lb])	3,000 (6,614)	2,260 (4,982)
Saturn Arrival Date	06/25/2004	11/15/2004
SOI Burn Delay	Yes	No
Initial Orbit Period (days)	152	200
Titan Flybys	35	21

2.2.2 Launch Opportunities

2.2.2.1 SRMU-Equipped Titan IV Configuration

Interplanetary missions can be launched only during specific opportunities (launch periods), depending on the relative positions of Earth and the target planet(s) and on the capabilities of the available launch vehicles. For the Proposed Action, the primary launch opportunity occurs during the 25-day period between October 6, 1997, and October 30, 1997. Problems with the launch vehicle or spacecraft or adverse weather conditions during this period could cause the loss of this primary launch opportunity. To recover from such unplanned events, NASA requires identification of contingency launch opportunities that would allow attainment of the same mission objectives (i.e., 4-year science tour and Probe delivery) as the primary launch opportunity. The mission planners have identified secondary and backup launch opportunities in December 1997 and March 1999, respectively, in case such conditions arise. Secondary launch opportunities, by definition, can occur less than 6 months after the primary launch opportunity; backup opportunities, however, are required to occur at least 6 months after the primary launch opportunity (JPL 1993c).

If a launch opportunity were missed, the spacecraft trajectories and mission operations would probably be altered and mission budgets augmented. Such a change would likely require modification of support facilities for communications, spacecraft tracking, and general operations. Revised launch plans would affect not only a delayed mission but also other missions that depend on the resources of these facilities. Because of the specialized nature of space exploration missions, such as Cassini, trained personnel and supporting facilities would generally be retained between launch opportunities, resulting in additional costs.

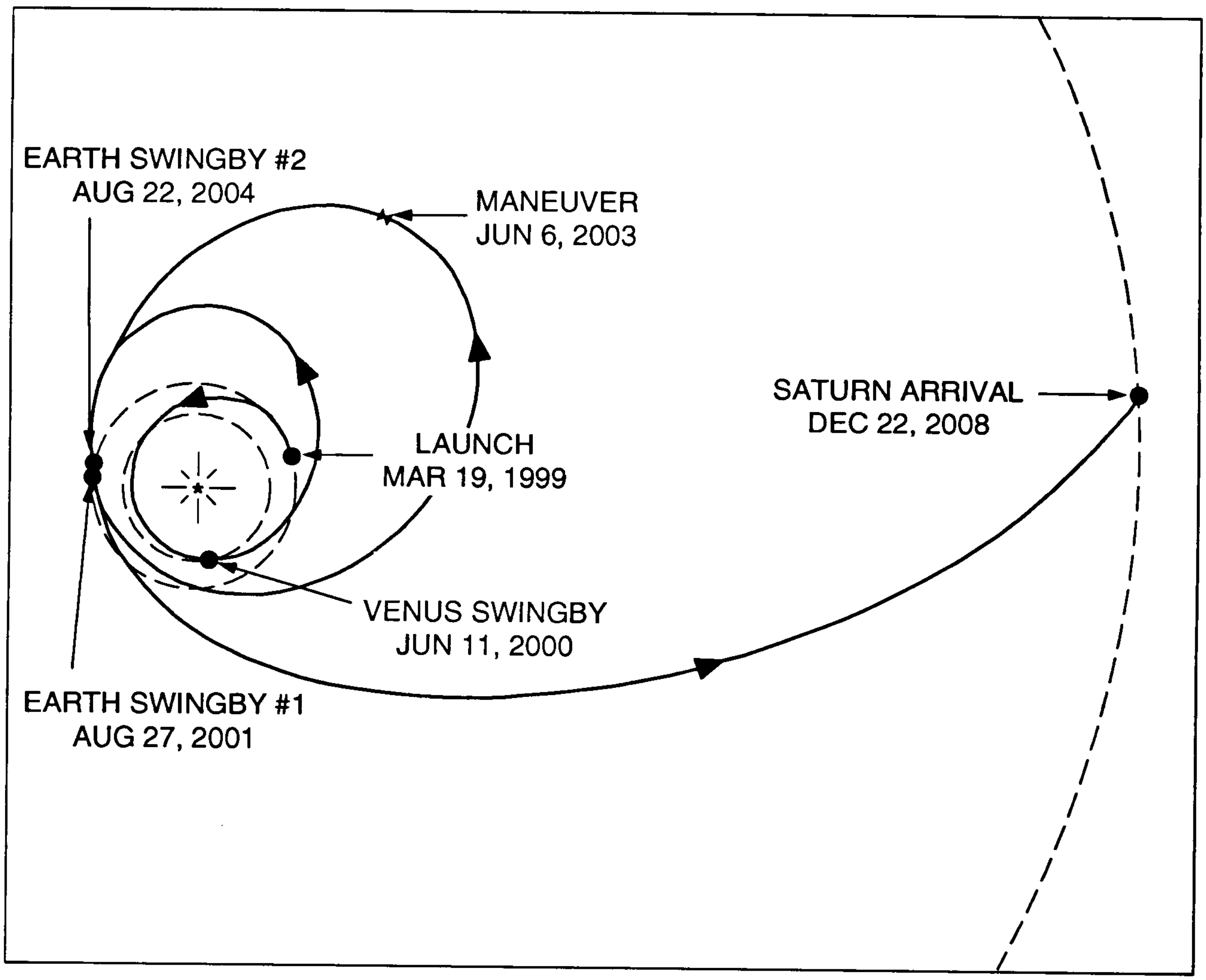
The secondary launch opportunity in December 1997, if utilized, would use an 8.8-year VEEGA trajectory. The interplanetary trajectory for the secondary launch opportunity would include a Venus swingby in June 1998, after which the spacecraft would proceed to Earth for 2 gravity-assists, during November 1999 and July 2002, respectively. The spacecraft would arrive at Saturn in October 2006. The launch opportunity would include a full 4-year tour of the Saturnian system and delivery of the Huygens Probe. The secondary launch opportunity would have a longer interplanetary cruise time and would return less ring science compared with the primary launch opportunity.

The March 1999 Titan IV (SRMU)/Centaur backup launch opportunity, illustrated in Figure 2-3, would use a 9.8-year VEEGA trajectory (JPL 1993a). This backup launch opportunity would have the same science objectives as the primary opportunity in October 1997, including the 4-year tour of Saturn's environment (JPL 1994a). The Huygens Probe experiments would be identical to those in the October 1997 launch. The interplanetary trajectory for the backup launch opportunity would include a Venus swingby in June 2000, after which the spacecraft would proceed to Earth for an additional gravity-assist in August 2001. This second assist would send the spacecraft on a broad sweeping arc through the asteroid belt. In August 2004, the spacecraft would arrive back at the Earth for a final gravity-assist and would arrive at Saturn in December 2008. The later arrival date would be less desirable than the arrival date for the October 1997 launch because there would be a 4-year delay in science return. In addition, the geometry of Saturn's rings would present less than optimum opportunities for both radio and optical science experiments. The 13.8-year nominal mission would end in December of 2012, about 4.5 years later than the primary launch opportunity.

Both the secondary and backup launch opportunities would have adequate allocations of propellant to meet the minimal science objectives. However, the longer flight times would result in lower electrical power output available from the RTGs during the science portion of the mission due to the natural decay of the radioisotopes. It would be during the last 4 years or the science portion of the mission that the electrical needs are greatest to power the science equipment and to perform data gathering activities. Therefore, fewer instruments could be operated at a given time or less engineering support given to some instruments (JPL 1993c). These mission constraints would reduce the science return from levels anticipated for the primary launch opportunity.

2.2.2.2 SRM-Equipped Titan IV Configuration

The launch opportunities for the Titan IV (SRM)/Centaur would be the same as those for the SRMU configuration. Because the SRM is a smaller booster with a lower lift capability, the amount of science return using the SRM-equipped Titan IV (for the same launch opportunities) would be less than that obtained with use of an SRMU-equipped Titan IV (JPL 1994a).



Source: JPL 1993f

FIGURE 2-3. CASSINI MARCH 1999 VEEGA INTERPLANETARY TRAJECTORY

2.2.3 Spacecraft Description

2.2.3.1 SRMU-Equipped Titan IV Configuration

The Cassini spacecraft, illustrated in Figure 2-4, is designed to be a three-axis stabilized probe-carrying orbiter for exploration of Saturn and its atmosphere, moons, rings, and magnetosphere. The Orbiter (i.e., the spacecraft without propellants or the Huygens Probe and its supporting equipment) would have a dry mass of 2,150 kg (4,740 lb), of which 335 kg (739 lb) are scientific instrumentation and 168 kg (370 lb) are the RTGs. The Huygens Probe and its supporting equipment would account for an additional mass of 352 kg (776 lb). The spacecraft launch vehicle adapter would add an additional 190 kg (419 lb). The spacecraft's most visible features would be the cylindrical shell structure. The main electronics and antennas would be mounted onto this structure. The primary bipropellant (hypergol fuels) tanks would be stacked within the shell. In addition, the main engines would be suspended from this structure. The RTGs would be supported by struts that extend from the base of the cylinder. The scientific instruments would be supported by a boom and two pallets or, in some cases, would be attached directly to the main structure (JPL 1993a). Less than 1 millicurie of minor radioactive sources (i.e., americium-241, barium-133, gadolidium-148, strontium-90, and rubidium-87) would be on the spacecraft, the Centaur, and the Probe, principally for instrument calibration.

The spacecraft would also contain communication, pyrotechnics, command and data, attitude and articulation control, propulsion, temperature control, and solid-state recorders subsystems.

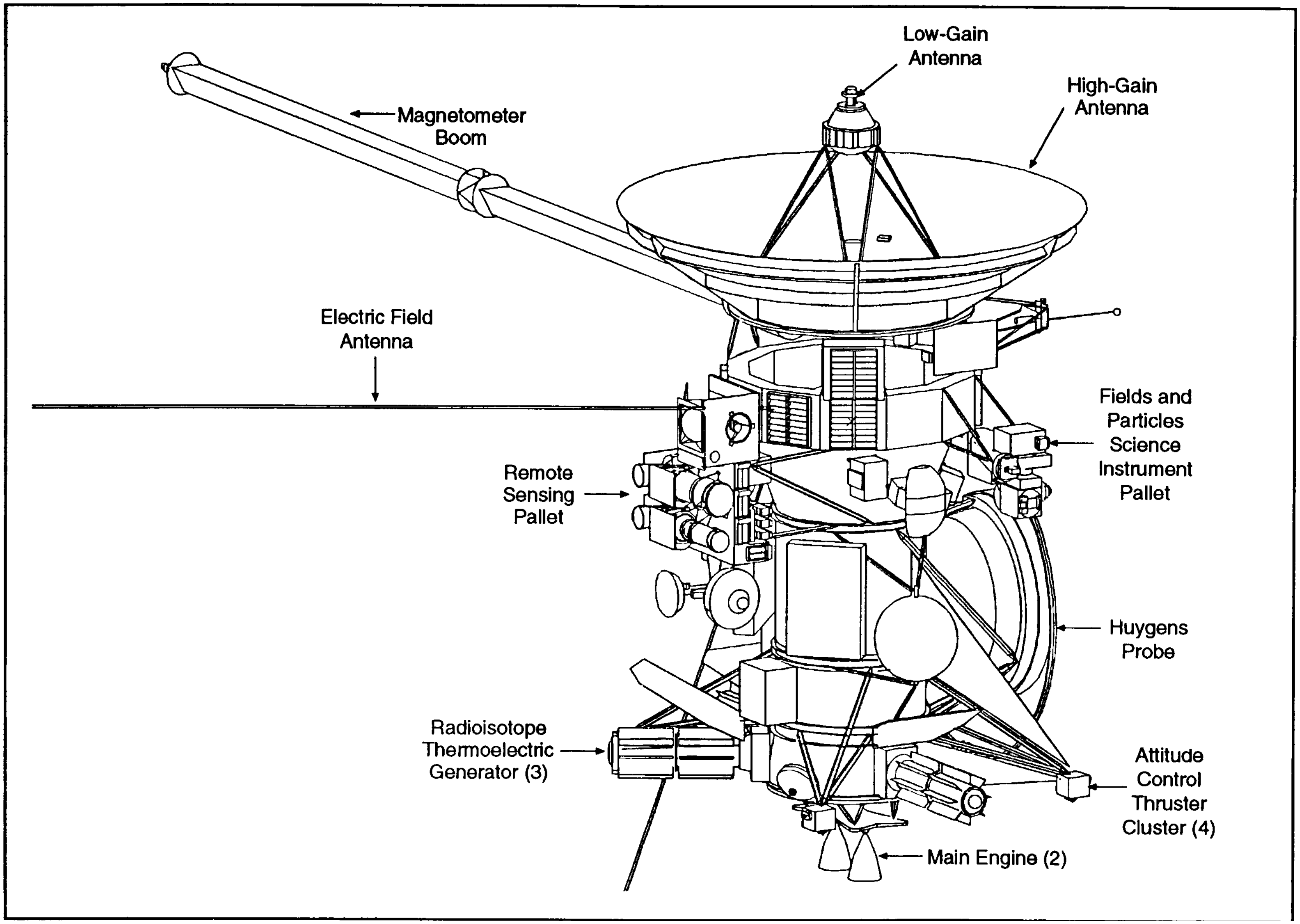
The components of the spacecraft relevant to any assessment of the potential for environmental impacts from the mission are the RTGs, the radioisotope heater units (RHUs), and the propellants. (RTGs and RHUs are discussed in Section 2.2.4.) For propellants, Cassini would carry up to 132 kg (291 lb) of hydrazine for small maneuvers and attitude and articulation control and about 3,000 kg (6,614 lb) of bipropellant (one tank each of monomethylhydrazine [MMH] and nitrogen tetroxide [NTO]) for larger maneuvers. The spacecraft (i.e., the Orbiter, the Probe and its supporting equipment, and the launch vehicle adapter), with propellants, would weigh 5,824 kg (12,840 lb) at launch (JPL 1993a).

2.2.3.2 SRM-Equipped Titan IV Configuration

If the Cassini spacecraft were to be launched using the Titan IV (SRM)/Centaur configuration, it would necessitate a reduction in the spacecraft bipropellant mass from 3,000 kg (6,614 lb) to 2,260 kg (4,982 lb) to compensate for the SRM's lower mass injection capabilities.

2.2.4 Spacecraft Electrical Power and Heating Sources

The Cassini spacecraft would use 3 RTGs as the source of electrical power for its engineering subsystems and science payload and a maximum of 157 RHUs to regulate temperatures of various subsystems on the spacecraft and the Probe (JPL 1994a). The



Source: JPL 1993a

FIGURE 2-4. DIAGRAM OF THE CASSINI SPACECRAFT

U.S. Department of Energy (DOE) would provide the RTGs and RHUs and would retain title to them at all times (DOE and NASA 1991).

2.2.4.1 Electrical Power System Performance Criteria

The Cassini spacecraft's 10.7-year mission (6.7-year VVEJGA trajectory plus 4-year Saturn tour) would impose stringent performance criteria for its systems and components. The electrical power requirement for the Cassini mission would be about 675 watts at end-of-mission JPL 1994a). The mission would involve spacecraft-to-Sun distances of between 0.63 astronomical units (AU) and 9.3 AU (JPL 1994a) (an AU is the distance from the Earth to the Sun equal to 149,599,000 km [92,956,500 mil). Therefore, the electrical power system must satisfy a variety of performance and implementability criteria, including the following:

- Operation during and after passage through intense radiation fields, such as those near the Earth and surrounding Jupiter
- Provision of sufficient power at distances of between 0.63 and 9.3 AU from the Sun
- Operation with a low mass-to-power ratio
- Provision of a long-term (12 years) source of electrical power with high reliability.

To fulfill these requirements, an indepth analysis of the available electrical power systems was performed to identify the most appropriate power source for the Cassini mission (JPL 1994a) (see Section 2.6.3). The use of RTGs was identified as the only feasible power system with the physical and operational characteristics compatible with achieving a high percentage of the science return from the Cassini mission. Previous performance and implementation criteria for other deep space missions have also identified RTGs as the only suitable power system, as was the case recently for both the Galileo and Ulysses missions (NASA 1989b, NASA 1990).

2.2.4.2 Radioisotope Thermoelectric Generators

The Cassini mission proposes to use three RTGs to provide electrical power to operate the spacecraft and its science instruments. An RTG power system uses an energy source and a conversion system. The decay heat from the radioactive energy source (plutonium dioxide) is directly converted into usable electrical energy by a thermoelectric converter.

RTGs were used on 23 previously flown U.S. space missions, including Voyager, Pioneer, Viking, all but the first of the manned Apollo flights, and the recent Galileo and Ulysses missions (Table 2-2). Heat source technology, pursued by DOE, has resulted in several models of an RTG power system, evolving from the Systems for Nuclear Auxiliary

TABLE 2-2 U.S. SPACECRAFT LAUNCHES INVOLVING NUCLEAR POWER SOURCES

Power Source (number of RTGs)	Sapcecraft	Mission Type	Launch Date	Status	Power System Inventory of Pu at Launch (Curies)[a]
SNAP-3B7 (1)	TRANSIT 4A	Navigational	29 Jun 61	Currently in orbit	1,500 - 1,600
SNAP-3B8 (1)	TRANSIT 4B	Navigational	15 Nov 61	Currently in orbit	1,500 - 1,600
SNAP-9A (1)	TRANSIT 5BN-1	Navigational	28 Sep 63	Currently in orbit	17,000
SNAP-9A (1)	TRANSIT 5BN-2	Navigational	5 Dec 63	Currently in orbit	17,000
SNAP-9A (1)	TRANSIT 5BN-3	Navigational	21 Apr 64	Mission aborted; burned up on reentry as designed	17,000
SNAP-10A (reactor)	SNAPSHOT	Experimental	3 Apr 65	Successfully achieved orbit; after 43 days in orbit, was shut down	N/A
SNAP-19B2 (2)	NIMBUS-B-1	Meteorological	18 May 68	Mission aborted; heat source retrieved	34,400
SNAP-19B2 (2)	NIMBUS III	Meteorological	14 Apr 69	Currently in orbit	37,000
SNAP-27 (1)	APOLLO 12	Lunar	14 Nov 69	Station shut down	44,500
SNAP-27 (1)	APOLLO 13	Lunar	11 Apr 70	Mission abort on way to moon; heat source fell in Pacific ocean	44,500
SNAP-27 (1)	APOLLO 14	Lunar	31 Jan 71	Station shut down	44,500
SNAP-27 (1)	APOLLO 15	Lunar	26 Jul 71	Station shut down	44,500
SNAP-19 (4)	PIONEER 10	Planetary	2 Mar 72	Successfully operated to Jupiter and beyond	80,000
SNAP-27 (1)	APOLLO 16	Lunar	16 Apr 72	Station shut down	44,500
TRANSIT-RTG (1)	"TRANSIT" (TRIAD-01-1X)	Navigational	2 Sep 72	Currently in orbit	24,000
SNAP-27 (1)	APOLLO 17	Lunar	7 Dec 72	Station shut down	44,500
SNAP-19 (4)	PIONEER 11	Planetary	5 Apr 73	Successfully operated to Jupiter and Saturn and beyond	80,000
SNAP-19 (2)	VIKING 1	Mars	20 Aug 75	Lander shut down	~40,980
SNAP-19 (2)	VIKING 2	Mars	9 Sep 75	Lander shut down	~40,980
MHW-RTG (4) (2 each spacecraft)	LES 8/9	Communications	14 Mar 76	Currently in orbit	318,800
MHW-RTG (3)	VOYAGER 2	Planetary	20 Aug 77	Successfully operated to Neptune and beyond	240,000
MHW-RTG (3)	VOYAGER 1	Planetary	5 Sep 77	Successfully operated to Saturn and beyond	240,000
GPHS-RTG (2)	GALILEO	Planetary	18 Oct 89	Successfully operating on flight to Jupiter	264,400
GPHS-RTG (2)	ULYSSES	Planetary	6 Oct 90	Successfully operating on flight to the polar regions of the Sun	132,500

Source: updated from NASA 1990

a. 1 Curie is equal to 3.7×10^{10} becquerel (Bq)

Power (SNAP)-RTG, to the Multi-Hundred Watt (MHW)-RTG, and, most recently, to the currently used General Purpose Heat Source (GPHS)-RTG. The GPHS technology is the culmination of almost 25 years of design evolution,

A GPHS-RTG assembly, commonly referred to as an RTG, weighs 56 kg (123.5 lb), is approximately 114 cm (44.9 in.) long and 42 cm (16.5 in.) in diameter, and contains 10.8 kg (23.8 lb) of plutonium dioxide fuel (DOE 1990a). Under space operational conditions, each RTG is designed to provide 285 watts of electrical power from 4,264 watts of heat (rating at launch). An RTG consists of two major functional components: the thermoelectric converter and the GPHS, as shown in Figure 2-5.

The thermoelectric converter consists of the aluminum outer shell assembly, the axial and midspan heat source supports, the thermoelectric elements, the multi-foil insulation packet, and the gas management system. The thermoelectric converter contains 572 silicon germanium (SiGe) thermoelectric couples (unicouples), which convert decay heat from the fuel directly into electricity. The unicouples are surrounded by multifoil insulation to reduce thermal losses. Each unicouple assembly is attached to an aluminum outer case (radiator) by sealing screws inserted through the case wall (DOE 1990a). The converter provides the support structure for the thermoelectrics, as well as for the GPHS modules.

The radioisotope energy source for the RTG is a stacked column of 18 individual GPHS modules, each having the dimensions of approximately 9.32 x 9.72 x 5.31 cm (3.67 x 3.83 x 2.09 in.) and a mass of about 1.45 kg (3.2 lb). The GPHS modules supply the thermal energy to the thermoelectric converter. Each GPHS module, illustrated in Figure 2-6, consists of a graphite aeroshell, two carbon-bonded carbon fiber (CBCF) insulator sleeves, two graphite impact shells (GISs), and four fueled clads. The graphite (carbon-carbon composite) aeroshell has a nominal operating temperature in space of 1,060°C (1,940°F) at the aeroshell surface (DOE 1990b) and serves as the module's primary heat shield to protect the internal components from direct exposure to a reentry's thermal and aerodynamic environment. The two GISs contained in the GPHS module provide the primary resistance to impact or mechanical loads. Each GIS assembly (i.e., the GIS and two fueled clads) is thermally insulated from the aeroshell by a low thermal-conducting CBCF insulator sleeve. Each fueled clad, separated by a graphite floating membrane, consists of one fuel pellet of ceramic (or solid) plutonium (mainly Pu-238)[1] dioxide encased in an iridium shell. The iridium shell protects and immobilizes the fuel. The iridium alloy is compatible (i.e., does not chemically react) with the plutonium dioxide fuel material to temperatures of more than 1,500°C (2,732°F), resists oxidation in air to 1,000°C (1,832°F), and melts at 2,425°C (4,497°F). Each clad also contains a vent designed to release the helium generated by the alpha particle decay of the fuel.

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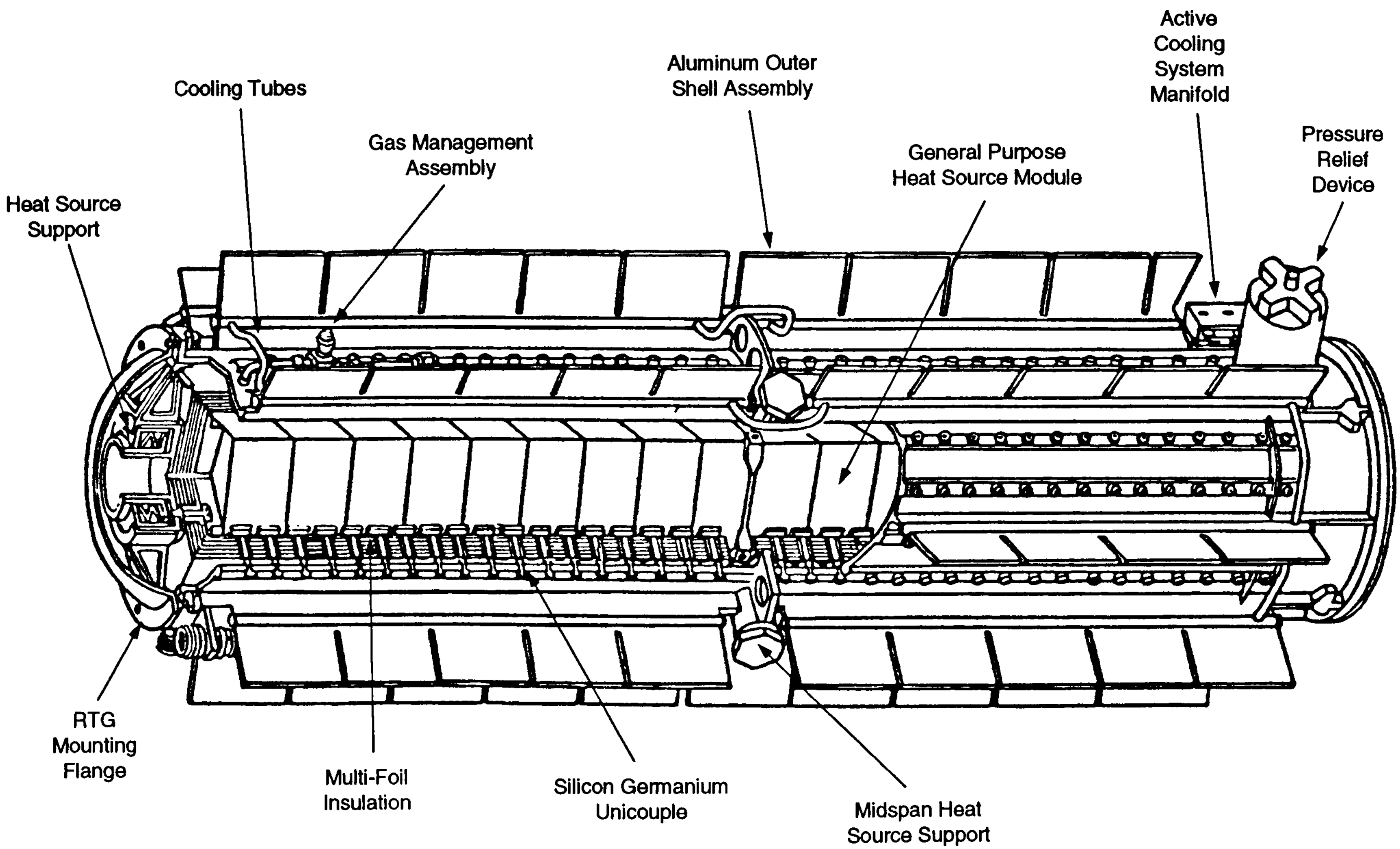
[1] Plutonium, atomic number 94, can exist in a number of radioactive isotopic forms, ranging from Pu-232 to Pu-246. Isotopes of an element have different atomic weights (e.g., 238, 239) but have the same or very similar chemical characteristics. The isotope Pu-238 forms the basis for the fuel in an RTG, whereas Pu-239 is the weapons-grade isotope. Pu-238 comes from the neutron bombardment of neptunium-237 and decays with an 87.75 year half-life to form naturally occurring uranium-234. Pu-239 comes from neutron capture by naturally occurring uranium-238.



Source: DOE 1990a

FIGURE 2-5. DIAGRAM OF GPHS-RTG ASSEMBLY

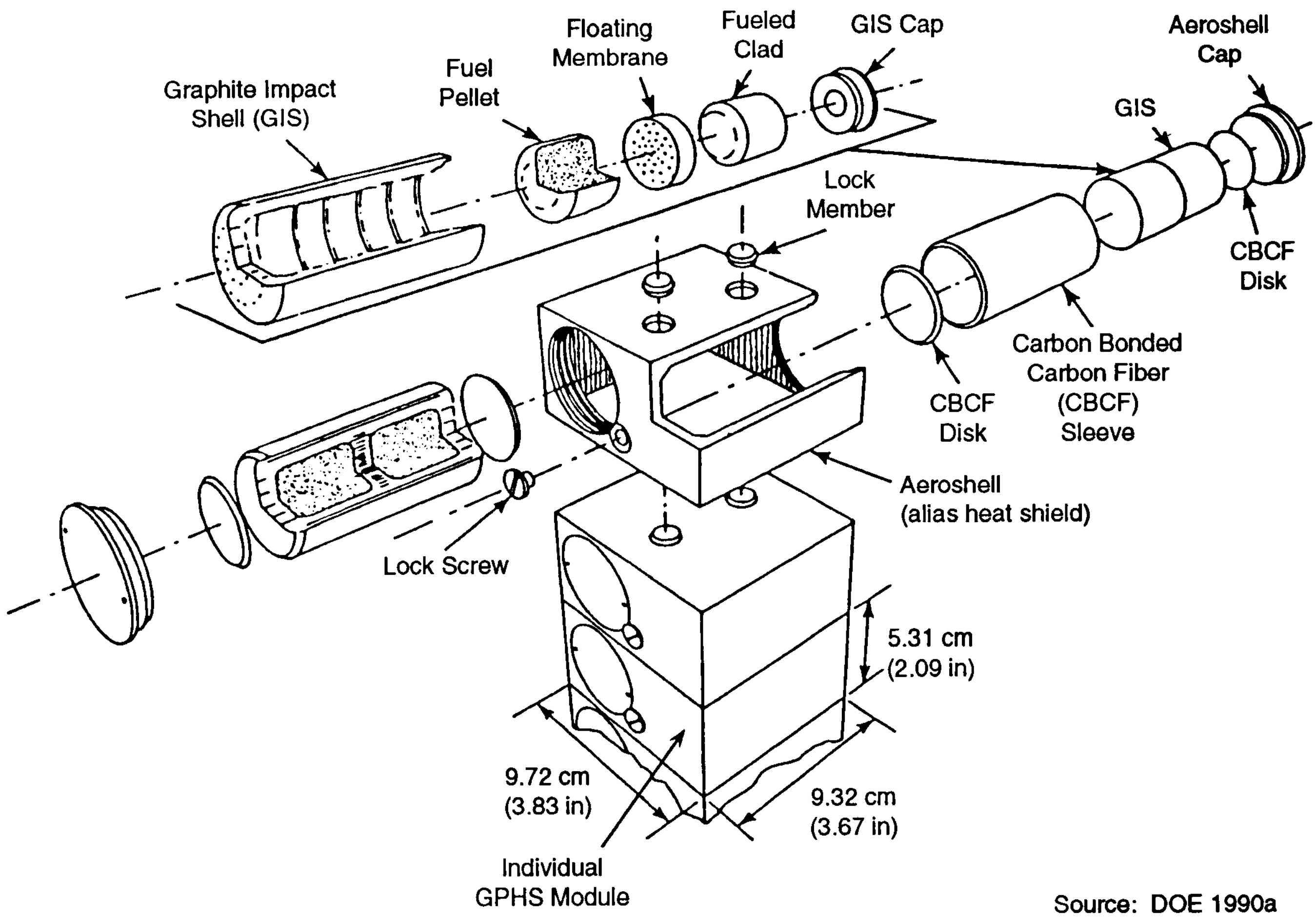


FIGURE 2-6. DIAGRAM OF GENERAL PURPOSE HEAT SOURCE MODULE

Table 2-3 provides representative characteristics and the isotopic composition of the 10.8 kg (23.8 lb) of plutonium dioxide fuel that would be used in one RTG for the Cassini mission. Plutonium dioxide has a density of 9.6 g/cm^3 (600 lb /ft^3), melts at 2,400° C (4,352° F) and boils at 3,870° C (6,998° F) (DOE 1990b).

As noted in Table 2-3, each ceramic pellet of plutonium dioxide fuel at launch (October 1997) for the Cassini mission would contain, by weight, approximately 71 percent Pu-238 and 13 percent Pu-239 (Fairchild Space 1993). Pu-238, with a half-life of 87.75 years, contributes most of the thermal energy to the heat source, with smaller contributions coming from Pu-239 (half-life of 24,131 years). Henceforth, the term plutonium dioxide refers to a mixture of the oxides of several plutonium isotopes, with Pu-238 as the dominant isotope.

Safety Considerations

DOE has designed the GPHS to assure that the plutonium dioxide fuel is contained or immobilized to the maximum extent practical during all mission phases, including ground handling, transportation, launch, and unplanned events, such as atmospheric reentry from Earth orbit, Earth impact, and post-impact situations. The design features for the GPHS-RTG assembly incorporate many safety-related considerations. The graphitic (carbon-carbon composite) materials (i.e., the aeroshell, CBCFs, and GISs) contained in the GPHS modules perform several safety functions. As stated previously, the primary function of the aeroshell is to protect the fueled clads against the hostile environment of atmospheric heating. The GISs protect the fueled clads from ground or debris impact in the event of an accident. Each GIS also serves as a redundant heat shield in the event of a GPHS failure. In addition, the GIS also acts as a redundant reentry aeroshell. The graphitic material used for the aeroshell and impact shell is called fine weave pierced fabric (FWPF)[1]. FWPF is a carbon-carbon composite material woven with high-strength graphite fibers in three perpendicular directions. Upon impregnation and graphitization, this material has outstanding high temperature strength capabilities required to accommodate heat shield mechanical and thermal stresses that occur during reentry. This material, used primarily by the U.S. Air Force (USAF) for missile nose cones (nosetips), is one of the best available for reentry applications (DOE 1988a).

The GPHS modules survive water impact. Given the additional protection of the iridium and the low solubility of the plutonium dioxide in fresh and sea water, the GPHSs will resist significant fuel release for virtually unlimited periods when submerged (DOE 1990a). (See Appendix C for further details.)

Because the GPHS is designed of small modular units, reentry heating and terminal velocity are lower than for previous heat sources, such as those used on the Pioneer and Voyager outer planet missions, due to a lower ballistic coefficient. A modular heat source tends to minimize the amount of fuel that could be released in a given accident.

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[1] FWPF is a trademark of AVCO Corporation.

TABLE 2-3. REPRESENTATIVE CHARACTERISTICS AND ISOTOPIC COMPOSITION OF CASSINI RTG FUEL

Fuel Component	Weight Percent at Launch	Half-Life (years)	Specific Activity (Bequerels/gram [curies/gram] of plutonium}	Total Bequerels (curies)/RTG[a] at Launch
Pu-236	0.0000010	2.851	2.0×10^{13} (531.3)	2.2×10^{9} (0.06)
Pu-238	70.810	87.75	6.3×10^{11} (17.12)	4.84×10^{15} (130,925.20)
Pu-239	12.859	24,131	2.3×10^{9} (0.0620)	3.2×10^{12} (86.11)
Pu-240	1.787	6,569	8.4×10^{9} (0.2267)	1.6×10^{12} (43.75)
Pu-241	0.168	14.4	3.8×10^{12} (103.0)	6.9×10^{13} (1,864.30)
Pu-242	0.111	375,800	1.5×10^{8} (0.00393)	1.8×10^{9} (0.05)
Other[b] Oxygen	2.413 11.852	NA[c]	NA	NA
Total	100.000	NA	NA	4.9×10^{15} (132,920)

Source: Fairchild Space 1993

a. Based on computation of isotopic composition by Fairchild Space for the launch date (October 1997). The radioisotopic fuel for each Cassini RTG is a mixture of plutonium dioxide (PuO_2) containing 70 percent (plus or minus 1 percent) Pu-238 and totaling 10.8 kg (23.8 lb). Three RTGs are planned for Cassini.
b. Small amounts of long-lived actinides and stable impurities.
c. Not applicable.

Considerable testing has been performed to determine the response of the RTG, the GPHS module, and the bare fueled clads to the environments that could result from a potential launch accident. The following list summarizes the relevant safety testing and the RTG's estimated response to the associated accident environments (see additional details available in DOE 1989b, DOE 1990a):

- Explosions-Fueled clads contained in GPHS modules and intact RTG assemblies survive overpressures of 15.25 megaPascals (MPa) (2,210 pounds per square inch [psi]) without any release of fuel. (For an intact RTG, the threshold for removal of the graphite aeroshell has been estimated to occur at overpressures around 3.45 MPa 1500 psi] [DOE 1989b]).

- Fire-Fueled clads contained in the GIS and bare fueled clads (without GIS protection) survive solid propellant fires with temperatures estimated to be about 2,360°C (4,280°F) without fuel release. The major components of the GPHS (graphite, iridium, and the plutonium dioxide) have individual melting points that are greater than the flame temperatures for solid and liquid propellant fires. Although the graphite eutectic temperature is around 2,269°C (4,165°F), solid and liquid propellant fire tests have not indicated eutectic melting of the iridium (DOE 1989b).

- Fragments-Small fragment tests with 18-g (0.64-oz) aluminum bullets at velocities of about 555 m/s (1,820 ft/s) can cause a breach when striking a bare fueled clad; 3.25-g (0.11-oz) titanium bullets at velocities of 423 m/s (1387 ft/s) can cause a bare fueled clad to breach. Tests using 142 cm (56 in.) square steel plates that are 1.27 cm (0.5 in.) thick indicate that an RTG can survive face-on fragment impacts at velocities up to 212 m/s (695 ft/s) with no release of fuel; edge-on fragment impacts on an RTG at 95 m/s (312 ft/s) will rupture only the leading fueled clads of the GPHS module impacted, resulting in a fuel release (DOE 1989b, Martin Marietta Astro Space 1993).

- Reentry From Earth Orbit-GPHS modules survive atmospheric reentry ablation and thermal stress with acceptable design margins up to the Earth escape velocity of 11.1 km/s (36,420 ft/s) (DOE 1989b).

- Earth Impact-A series of tests were performed that simulated the conditions that might be expected during reentry of GPHS modules from Earth orbit. These tests impacted GPHS modules at velocities in excess of their terminal velocity (50.3 m/s [165 ft/s]) onto hard surfaces, including steel, concrete, and granite. Releases from a GPHS module after impacting a hard surface at a velocity 10 percent higher than terminal velocity ranged from 0 g to 0.22 g (0 oz to 0.008 oz) for the tests simulating orbital decay reentry conditions (DOE 1990a). Releases from GPHS modules from Earth orbital decay reentry are not expected for impacts onto water, sand, or normal soils.

  Impact tests were performed with bare fueled clads. The tests concentrated mainly on a velocity range centered on the terminal velocity of the fueled clads, which is about 73.8 m/s (242 ft/s). Bare clads did not fail on soft targets, such as sand, at impact velocities to 250 m/s (820 ft/s), and the failure thresholds were established for impacts on steel and concrete at 53 and 58 m/s (174 ft/s and 190 ft/s), respectively (DOE 1990a).

Overall, DOE has spent more than 12 years in the engineering, fabrication, safety testing, and evaluation of the GPHS, building on the experience gained from previous heat source development programs and an information base that has grown since the 1950s. Test results have demonstrated that the present design exceeds the stringent safety standards achieved by earlier heat source designs. In addition, DOE has considered and continues to consider ways to improve the safety of the current RTGs, including alternative materials and RTG designs in the event there were to be a potential need in future outer space missions. The RTG technology also has a proven record of reliability in space applications and is the only power system available that satisfies all of the performance criteria associated with the Cassini mission. DOE will perform additional safety analyses for the Cassini mission and document the results in a Final Safety Analysis Report (FSAR).

2.2.4.3 Lightweight Radioisotope Heater Units

The Cassini spacecraft and the Huygens Probe would use a maximum of 157 lightweight RHUs to regulate temperatures on the spacecraft and on the Probe (JPL 1993a). Each RHU provides about 1 watt of heat derived from the radioactive decay of 2.7 g (0.006 lb) of mostly plutonium (Pu-238) dioxide, contained in a platinum-30 rhodium (Pt-30Rh) alloy clad. Similar to the GPHS modules, the RHU design provides a high temperature capability by using the FWPF heat shield and a series of concentric pyrolytic graphite (PG) sleeves and end plugs to thermally insulate the fueled clad. The RHUs are protected from ground or debris impact partially by the heat shield, but principally by the Pt-30Rh clad material. Each RHU would contribute approximately 1.3×10^{12} Becquerel (Bq) (36 curies [Ci]) to the total radioactive inventory on Cassini. The exterior dimensions are 2.6 cm (1.03 in.) diameter by 3.2 cm (1.26 in.) long, each with a weight of about 40 g (0.09 lb). Figure 2-7 shows a cutaway view of an RHU.

The RHUs are designed to be lightweight units capable of containing the plutonium dioxide fuel in both normal operations and accidents (DOE 1988a). The integrity and durability of RHUs are well documented (DOE 1988a). DOE will perform additional safety analyses for the Cassini mission and document the results in a Final Safety Analysis Report (FSAR).

2.2.5 Spacecraft Propulsion Module Subsystem

The propulsive power for the Cassini spacecraft would be provided by two redundant bipropellant 490-N (110 lb of thrust) main engines for trajectory and orbit changes and 16 monopropellant thrusters rated at 0.6-N (0.13 lb of thrust) for attitude control and very small orbit changes (JPL 1993c). The bipropellant engines use NTO and MMH. The monopropellant thrusters burn hydrazine. Pressures in both the bipropellant and monopropellant elements are maintained using helium gas (JPL 1993a).

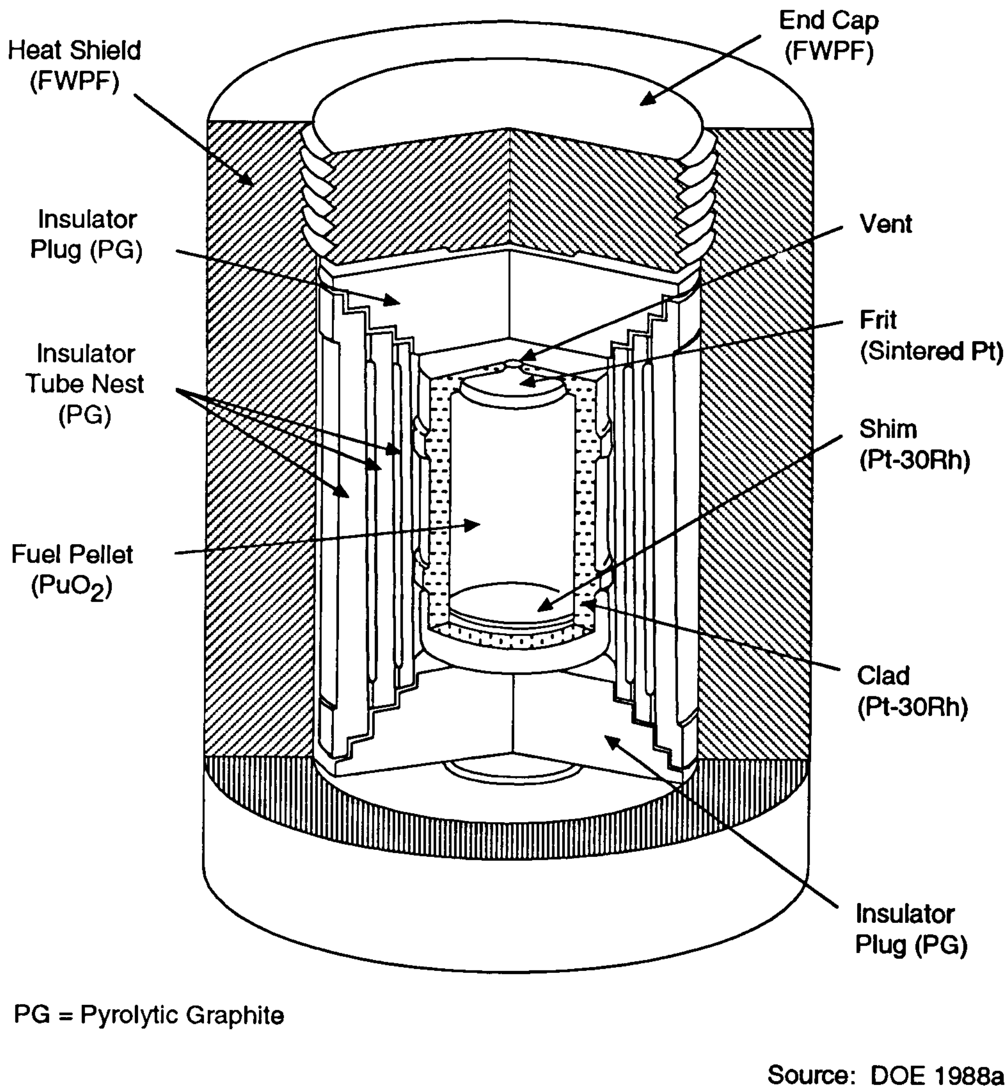


FIGURE 2-7. THE PRINCIPAL FEATURES OF THE RADIOISOTOPE HEATER UNIT

2.2.6 Launch Vehicle (Titan IV [SRMU or SRM]/Centaur) Configuration

The Titan family of unmanned, expendable launch vehicles has a launch history spanning more than 30 years of operations involving more than 320 Titan vehicles of all models. Titans have successfully carried astronauts into space 10 times and have successfully launched RTG-powered spacecraft into space 5 times. The Titan IV/Centaur with the newly developed SRMUs is proposed for this mission to Saturn, but if the SRMUs were not available, the mission would use the conventional SRMs. The SRMUs will be the most capable strap-on U.S. boosters when flight certified.

The Titan IV/Centaur comprises four basic components: core vehicle, the strap-on booster (SRM or SRMU), payload fairing (PLF), and Centaur (upper stage). The Titan IV (SRMU)/Centaur configuration is shown in Figure 2-8.

2.2.6.1 Core Vehicle

The core vehicle, which provides thrust, consists of two stages with their associated airframes, structures, avionics, mechanical systems, and liquid propulsion system. Stage 1 contains two bipropellant liquid rocket engines. The oxidizer is 101,176 kg (223,053 lb) of NTO and the fuel is 53,240 kg (117,373 lb) of Aerozine-50 (i.e., a 50-50 blend of unsymmetrical dimethylhydrazine and hydrazine). Stage 2 contains a single bipropellant engine virtually identical to the two used in Stage 1 . The Stage 2 propellants comprise 22,239 kg (49,028 lb) of NTO and 12,436 kg (27,416 lb) of Aerozine-50 (Martin Marietta 1992).

2.2.6.2 Strap-on Boosters

Two SRMUs (or SRMs), strapped onto the sides of the core vehicle, would provide the initial boost for the launch vehicle at liftoff. The SRMUs are three-segment, graphite-composite-cased motors representing a significant performance gain over the conventional SRM. The SRMU has passed all of its qualification tests and is awaiting final flight certification. Each SRMU weighs 351,220 kg (772,685 lb), of which 313,102 kg (688,824 lb) are propellant. The propellant is a U.S. Department of Defense (DOD) Hazard Class 1.3 solid propellant consisting of 69 percent ammonium perchlorate (oxidizer) and 19 percent nonspherical aluminum (fuel) with 8.84 percent hydroxyl terminated polybutadiene (HTPB) binder. The remaining 3.16 percent includes bonding and curing agents (Martin Marietta 1992).

The conventional SRM booster consists of a steel seven-segment propellant case plus forward and aft closures. Each SRM weighs 302,512 kg (694,470 lb), of which 257,440 kg (591,000 lb) is propellant consisting of 67.8 percent ammonium perchlorate and 16 percent aluminum with 10.2 percent polybutadiene acrylonitrile (PBAN) binder. The remainder consists of catalyst, resins, and stabilizers (Martin Marietta 1989).

2.2.6.3 Payload Fairing

The PLF, mounted on top of the core vehicle, encases the Centaur (upper stage) and spacecraft, thereby providing aerodynamic and thermal protection for these elements

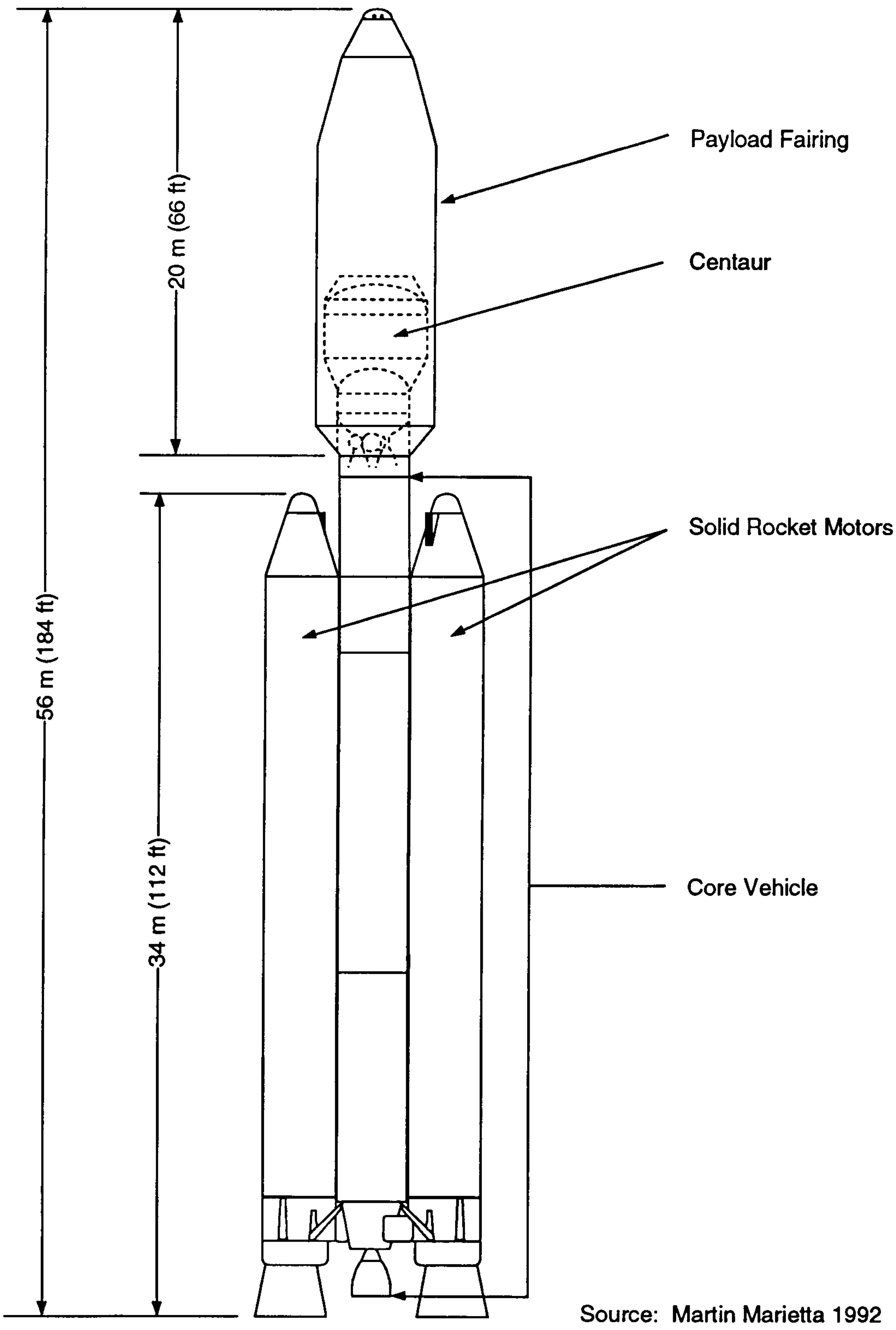


FIGURE 2-8. DIAGRAM OF THE TITAN IV (SRMU OR SRM) / CENTAUR LAUNCH VEHICLE

during ascent (Figure 2-9). The payload fairing is an all metal structure composed primarily of aluminum and pieced together as three segments. Between approximately 240 and 246 seconds after liftoff, the fairing segments would uncouple and would be jettisoned from the rest of the launch vehicle into the ocean (Martin Marietta 1992).

2.2.6.4 Centaur

The Centaur (upper stage) is designed to be compatible with the Titan IV booster. It uses two liquid hydrogen (LH_2)/liquid oxygen (LO_2) combustion engines with multiple restart capability. The Cassini mission would require two burns of the Centaur engine. The first burn would supplement the Titan IV in lifting the spacecraft/Centaur stack into the proper low altitude Earth parking orbit. The second burn would boost the spacecraft to the velocity needed to escape the Earth and would inject it into the proper trajectory toward the first gravity-assist swingby of Venus. The LH_2 and LO_2 are contained in two large tanks that account for the bulk of the Centaur's internal volume. Forward and aft adapters are mounted to these tanks. The forward adapter provides mounting supports for avionics packages and the spacecraft's mechanical and electrical interfaces with the Centaur, and the aft adapter provides the structural interface between the Centaur and the Titan IV (Martin Marietta 1992). Figure 2-10 illustrates the Centaur (upper stage) configuration.

2.2.7 Cassini Mission Timeline

The nominal Cassini mission timeline is subject to slight modifications as the design of the Cassini mission is further refined. As shown below for the Titan IV (SRMU)/Centaur, the mission is divided into phases that primarily serve as the basis for potential launch accident scenario definitions and environmental analyses. For example, Phase 0 starts with fueling of the Titan IV core vehicle 4 days before launch (T - 4 days) and ends with ignition of the SRMUs at T minus zero seconds (T - 0 s). Where necessary, the phases are divided further to separate specific events that show changes in the fundamental characteristics of the accident environments to which the RTG could be subjected (Martin Marietta 1992). These phases are essentially identical for all the launch opportunities associated with the Titan IV (SRMU)/Centaur. The phases and typical timeframes are summarized below (Martin Marietta 1992):

- Phase 0 (T - 4 days to T - 0 s)- Phase 0 identifies the time following the installation of the RTGs when the fueling of the Titan IV core vehicle begins and continues to the instant of SRMU ignition.

- Phase 1 (T - 0 s to T + 11 s)- Phase 1 covers the time period from the instant of SRMU ignition at a mission elapsed time (MET) of zero and continues to the time when the launch vehicle is high enough to provide launch site clearance in the event of an accident.

- Phase 2 (T + 11 s to T + 23 s)- This phase begins at the instant launch site clearance is achieved and continues to the point where the vehicle's instantaneous impact point (IIP- the point of vehicle impact given the termination of thrust, neglecting aerodynamic effects) would clear the Florida

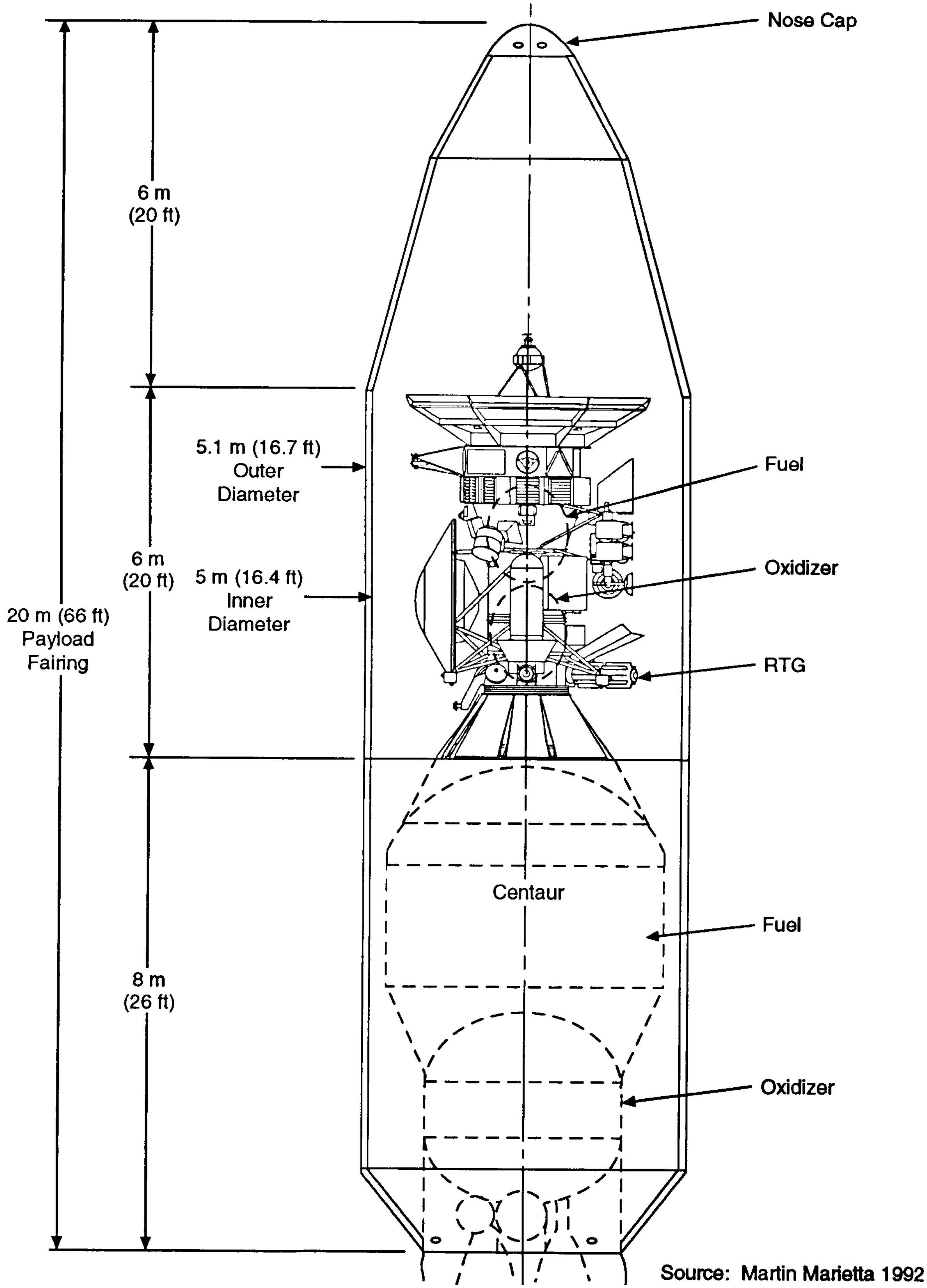
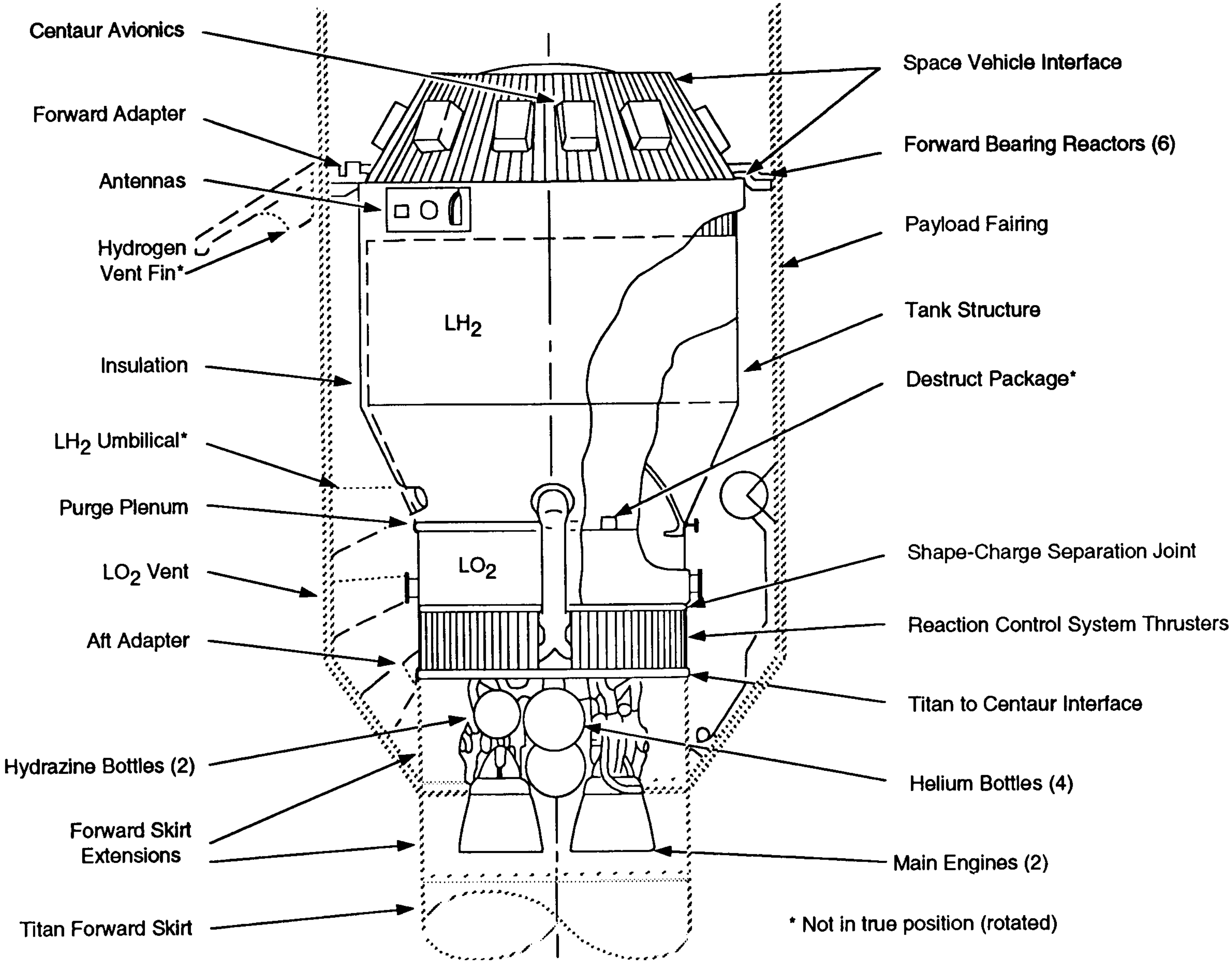


FIGURE 2-9. THE CENTAUR AND SPACECRAFT IN THE PAYLOAD FAIRING



Source: Martin Marietta 199

FIGURE 2-10. DIAGRAM OF CENTAUR CONFIGURATION

coast (i.e., land clear). During this phase, the vehicle would impact land in the event of an accident.

- Phase 3 (T + 23 s to T + 56 s) -Phase 3 begins at the instant of land clear and ends when the vehicle reaches an altitude of 10,000 m (32,808 ft). At this altitude, the potential environmental impacts from an accident resulting in a plutonium dioxide release become global due to high altitude winds.

- Phase 4 (T + 56 s to T + 246 s) -Phase 4 identifies the period from 10,000 m (32,808 ft) altitude and continues to when the jettison of the PLF is completed. Stage I of the liquid-fueled core vehicle main engines are ignited at T + 135 s, and the SRMUs are jettisoned at T + 146 s.

- Phase 5 (T + 246 s to T + 688 s) -Phase 5 covers the period from the completion of PLF jettison to the time when the flight termination system(s) is shut down, becoming inoperable for the remainder of the mission. The FTS shut down occurs 2 seconds after the end of the first Centaur main engine burn. The vehicle's IIP would pass over the African continent between approximately T + 664 s and T + 672 s.

- Phase 6 (T + 688 s to T + 5,576 s) - Phase 6 begins at the time when the flight termination system(s) is shut down and ends when the second Centaur main engine burn is completed.

- Interplanetary Cruise Phase (5,576 s to end of the mission) -This phase involves the time between Earth escape and completion of the mission. Depending on the trajectory (i.e., VVEJGA, VEEGA, or VVVGA), a combination of planetary gravity assists will occur, resulting in SOI, delivery of the Huygens Probe, and ultimately, the data acquisition from the 4-year science tour of Saturn and its environment.

For a launch with the Titan IV (SRM)/Centaur configuration, the mission timeline would typically be as follows (Martin Marietta 1989):

- Phase 0 (T - 4 days to T - 0) -Phase 0 identifies the time period between when the liquid propellants are loaded and the RTGs installed to the instant of SRM ignition.

- Phase 1 (T - 0 to T + 5 s) -This phase covers the time period from the instant of SRM ignition through lift-off and continues to the time when the launch vehicle is high enough to clear the launch tower.

- Phase 2 (T + 5 s to T +-23 s) -This phase covers the time period from launch tower clearance to the time when the IIP would clear the Florida coast (i.e., land clear).

- Phase 3 (T + 23 s to T + 250 s) -Phase 3 identifies the time period from land clear to the time of jettison of the payload fairing.

- Phase 4 (T + 250 s to T + 543 s)-Phase 4 covers the period when the PLF is jettisoned to the separation of the upper stage from the core vehicle.

- Phase 5 (T + 543 s to payload separation)-This phase covers the period between the separation of the upper stage to the injection of the spacecraft.

- Interplanetary Cruise Phase (payload separation to end of mission)-Phase 6 covers the interplanetary trajectory between the Earth and Saturn.

2.2.8 Range Safety Considerations

2.2.8.1 General

Range Safety encompasses all activities from the design concept through test, checkout, assembly, and launch of space vehicles to orbit insertion from any range facility. The goal of the Range Safety program is to protect the general public, foreign and domestic, as well as their property, from harm or damage resulting from the debris or impact of hazardous components during a space flight. The launch and flight of space vehicles should present no greater risk to the general public than that imposed by the overflight of conventional aircraft. Although risk can never be completely eliminated, Range Safety attempts to minimize the risks while not unduly restricting the probability of mission success.

2.2.8.2 Flight Termination System

All space vehicles launched from the Eastern Range, which includes KSC and CCAS, must carry an approved Flight Termination System (FTS) that allows the Flight Control Officer to terminate powered flight if the vehicle violates established flight safety criteria.

The Flight Termination System (FTS), which includes a Titan IV launch vehicle system and a Centaur system, provides ground personnel with the capability to shut down any thrusting liquid stage only (core engines or Centaur) or to shut down any thrusting liquid stage and then destruct the Titan IV SRMUs and all liquid stage tanks. This element of the FTS is called the command shutdown and destruct system (CSDS).

Additionally, the FTS will automatically destruct a stage that separates from the portion of the vehicle carrying the command receivers and antennas. This element, originally called the inadvertent separation and destruct system (ISDS), is currently referred to as the automatic destruct system (ADS). The ADS is activated when a wire, strategically placed to sense a specific critical structural failure, is broken. Upon activation of an automatic destruct, Range Safety can, at their discretion, command destruct the Centaur and the remaining Titan IV elements, which were not destructed automatically or broken up due to collateral damage. (It should be noted that the location of the sensing wires that detect structural failure is subject to review based on Range Safety and nuclear safety issues.)

As of publication of the DEIS, the necessity for and design issues involving a Space Vehicle Destruct System (SVDS) for the Cassini spacecraft were under review.

Subsequent analyses and testing involving the spacecraft's hypergol fuels indicated that the launch vehicle configuration for the Proposed Action would not require an SVDS. Therefore, the Cassini spacecraft would not include an SVDS.

2.2.8.3 Electromagnetic Hazard Conditions

Techniques to respond to concerns for potential electromagnetic hazards have been incorporated into launch vehicle designs and launch procedures. The following potential electromagnetic hazard conditions exist for aerospace launch vehicles and payloads:

- Lightning

- Powerful electromagnetic transmitters (e.g., radars, radio transmitters), also referred to as the electromagnetic environment

- Charging effects (i.e., triboelectric charging effects and resultant electrostatic discharges [ESD]).

NASA and the USAF are concerned with these conditions with respect to the design of the launch vehicle, as well as with ordnance (explosives and explosive detonators/fuses), fuels, exposed skins of the vehicle, and critical electronic systems that must have highly reliable operations. These special concerns are well-known and they include the following:

- Effects of electromagnetic radiation on ordnance and fuels

- Electrostatic discharges

- Electromagnetic interference.

A large body of technical literature exists on these subjects and has been used by NASA and the USAF in designing safeguards. To better understand these hazards, the following paragraphs briefly describe these conditions and hazards.

Lightning

Lightning is the electrical discharge that typically occurs during thunderstorms. Large electrical current, which can approach several hundred thousand amperes, can flow from cloud to cloud or from cloud to ground in a fraction of a second. If a vehicle is in the vicinity of a thunderstorm, there is a chance that all, or some, of the electrical current can flow into or through the vehicle. This possibility is mitigated by avoiding flight through thunderstorms and by using special vehicle designs that prevent the serious effects of lightning strikes.

The conditions in which lightning is likely to occur can be monitored by measuring the local electric fields around the vehicle. Large electric fields indicate the presence of large amounts of electrical charge present in the overhead clouds. Because lightning results from an electrical discharge built up in these clouds, these fields indicate the likelihood of lightning activity in the area. The USAF monitors electrical fields within a 322 km (200 mi) radius of CCAS/KSC during launch times. Operations at the launch pad will neither commence nor continue if an electrical storm is within 8 km (5 mi).

The USAF also employs rigorous design specifications (e.g., Military Standard [MIL-STD]-1818, Electromagnetic Effects Requirements for Systems, dated 8 May 1992) to mitigate the potential effects of lightning strikes and will have strict meteorological criteria for launch of the Titan IV (SRMU or SRM)/Centaur to avoid subjecting the vehicle and its payload to unacceptable environments. In addition to visibility, ambient temperature, and surface wind speed and direction, severe weather restrictions address the maximum weather-induced, ground-level, and flight path electrical fields acceptable for launch (1 kv/m). These restrictions are strictly enforced.

Electromagnetic Environment

The electromagnetic radiation in the environment has, in recent years, become stronger and more prevalent primarily because of the increased number of radar systems and other radio transmitters worldwide.

Controlling the response and interaction of the Titan IV (SRMU or SRM)/Centaur and payload systems to the electromagnetic environment is achieved through two means: control of the radiated power of transmitters in the immediate vicinity of the vehicle and use of proven and effective electrical system design techniques. These techniques include using electromagnetic shields, controlling any naturally occurring electromagnetic leaks in the shields, using proper electrical bonding and grounding, filtering out and/or suppressing undesired effects in the electrical system, and using special signal computer software that recognizes and removes the effects. The techniques are designed to comply with MIL-STD-1818 and in MIL-STD-461, Electromagnetic Emission and Susceptibility Requirements for the Control of Electromagnetic Interference, and MIL-STD-462, Electromagnetic Interference, Measurement of Characteristics.

Electromagnetic compatibility (EMC) is defined as the condition that prevails when pieces of telecommunication (communication/electronic) equipment are performing their individual designed functions in a common electromagnetic environment without causing or suffering unacceptable degradation due to electromagnetic interference to or from other equipment and systems in the same environment.

All payload manufacturers are required to conduct EMC testing of their unintentional radiated and conducted emissions. NASA reviews test data according to strict specifications. When a piece of equipment, system, or subsystem is determined to have an inadequate Electromagnetic Interference Safety Margin, appropriate action is taken (e.g., redesign, substitution, or additional protection). Therefore, the concerns for payload to payload, payload to Titan IV (SRMU or SRM)/Centaur, and Titan IV (SRMU or SRM)/Centaur to payload radiated emissions are thoroughly addressed and resolved before flight readiness is attained.

Many well-known techniques are available to achieve EMC. The procedures used to achieve this operational compatibility are prescribed in Military Specification 6051, Electromagnetic Compatibility Requirements System, updated February 28, 1988, and supporting procedures. All avionics equipment on the Titan IV (SRMU or SRM)/Centaur has been EMC qualified per NASA, USAF, and contractor specifications. All EMC reports are presented and reviewed during a series of payload integration reviews involving all contractor and government parties. Problems are resolved during these reviews. In addition, all payload and the Titan IV (SRMU or SRM)/Centaur EMC requirements are constantly reviewed and updated whenever new information becomes available.

Charging Effects

Electrical charging effects can be associated with picking up an electrical charge that can suddenly discharge when a metallic object is touched. The effect results from rubbing or touching and parting two dissimilar materials together.

Charging can be produced by space vehicles flying through dust and clouds that are composed of water droplets. Such discharges can lead to electrical interference.

Techniques to mitigate the effects of ESD are well-known and generally depend on proper bonding and grounding of the external and internal vehicle assemblies and parts. This prevents large differential charges from building up between surfaces and arc discharging. Most vehicle charge resides on external and payload bay surfaces. The USAF uses Class S bonding, as prescribed by MIL-STD-1818, to prevent the effects of electrostatic discharging.

Ordnance and Fuels

Ordnance and fuels represent special concerns. Electrostatic and electrodynamic energy can potentially trigger fuel ignition of special ordnance (i.e., electroexplosive devices), which can lead to undesired ordnance ignition and possibly equipment separations. Due to the fuel containment design, substantial amounts of energy from the radio frequency environment or electrostatic discharge are needed to trigger the liquid and solid fuels.

Techniques used to protect such ordnance and fuels from lightning, the electromagnetic environment, and discharges are well-known and used in many aircraft and missile systems. These techniques, used by the USAF, are prescribed in MIL-STD-1576, Electroexplosives Subsystem Safety Requirements and Test Methods for Space Systems, which establishes general requirements for the design, development, and testing of electroexplosive subsystems to preclude hazards from ignition and failure to fire. This standard applies to all space vehicle systems (e.g., launch vehicles, upper stages, payloads, and related systems). Special designs of fuel tanks and fuel delivery subsystems are used to prevent ordnance ignition.

On the Cassini spacecraft, for example, there would be two types of such electroexplosive devices: NASA standard initiators (NSIs) and bellows actuators. NSIs would be used to actuate pinpullers, propulsion valves, release nuts, rod cutters, and the Huygens Probe parachute deployment device. An NSI and detonator combination would be used to activate the detonating cord used in the Linear Separation Assembly. The bellows actuators would be used to deploy science instrument covers. Both electroexplosive devices are designed to specifications that require an inability to "fire" when 1 watt or 1 ampere is applied to the device. These devices would be controlled by redundant firing circuits that are inhibited during launch by a series of relays located on the Centaur, the spacecraft, and the Huygens Probe. The inhibit logic and circuitry are designed to comply with the Range Safety Requirements document 45th Space Wing Regulation 127-1 and MIL-STD-1576. The combination of these devices provides a very large margin compared to the energy available from external sources.

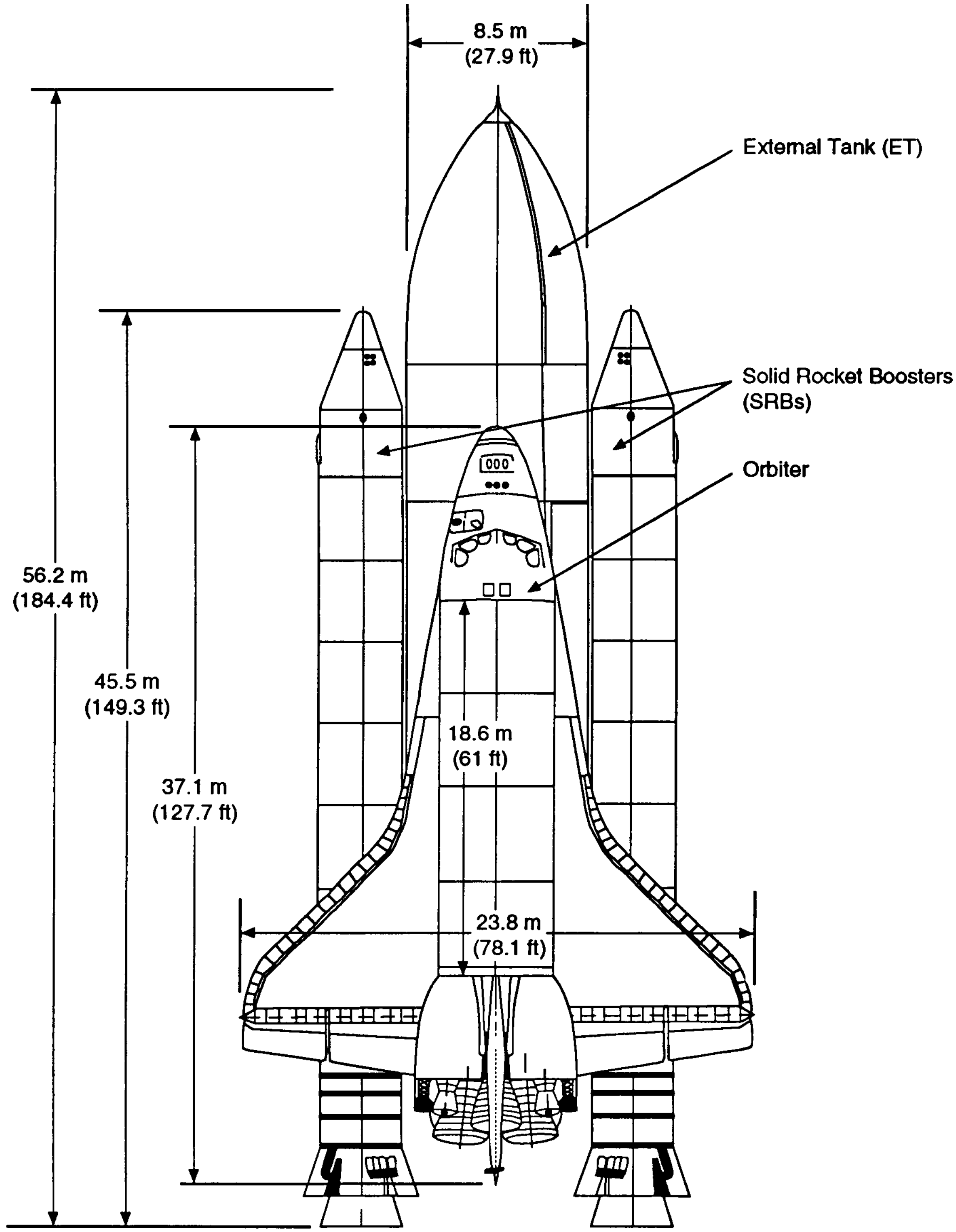
Electromagnetic interference is also a design consideration with respect to the pyrotechnic devices (detonators) on the launch vehicle. These devices and the firing circuits are designed to perform to MIL-STD-1512, Electro-Explosive Systems, Electrically Initiated, Design Requirements, and Test Methods. All spacecraft and upper stages that fly on the launch vehicle undergo an intensive review of their susceptibility to electromagnetic radiation in accordance with strict NASA and USAF specifications. Hazard reports must be prepared and closed out for devices that do not meet the specifications.

The pyrotechnic devices on the Titan IV (SRMU or SRM)/Centaur launch vehicle reflect the design and operational experience gained from the entire U.S. launch vehicle/spacecraft history to date. The launch vehicle design requires that several separate, distinct electrical signals be received in the proper sequence to initiate firing outputs from the pyrotechnic initiator controllers. Circuit designs have been developed to ensure that electrical shorts to either ground or power will not cause any premature firing of these devices. In addition, the explosive materials in these devices have been chosen after extensive material test programs and development testing under flight conditions to ensure that they will not auto-ignite in the flight environment, which includes electromagnetic radiation.

2.3 DESCRIPTION OF THE 1999 MISSION ALTERNATIVE

2.3.1 Mission Design

For the 1999 mission alternative, the Cassini spacecraft with the Huygens Probe and three RTGs would be launched to Saturn by the Space Shuttle from Launch Pad 39A or 39B located at Kennedy Space Center. Figure 2-11 illustrates the Shuttle. This mission alternative would require two Shuttle launches separated by at least 21 days but no more than 51 days. The first Shuttle launch would place an upper stage into low Earth orbit and the second launch would deliver the remaining upper stage(s) and the Cassini spacecraft including the RTGs. The upper stages with the spacecraft would be assembled on-orbit by astronauts. The spacecraft would then be injected into its interplanetary trajectory to Saturn by the upper stages. The spacecraft would travel on a VEEGA trajectory, which would be similar to the trajectory identified for the Titan IV March 1999 backup launch opportunity for the Proposed Action (see Figure 2-3). The backup launch opportunity for this mission alternative would occur in August 2000, using a VEEGA trajectory. Figure 2-1 2 illustrates the dual Shuttle launch and on-orbit assembly of the upper stages with the spacecraft. The dual Shuttle launch would provide full maneuvering capability (similar to the Titan IV [SRMU]/Centaur), but because of requirements of the Huygens



Source: JPL 1994a

FIGURE 2-1 1. DIAGRAM OF THE SHUTTLE (STS) LAUNCH VEHICLE

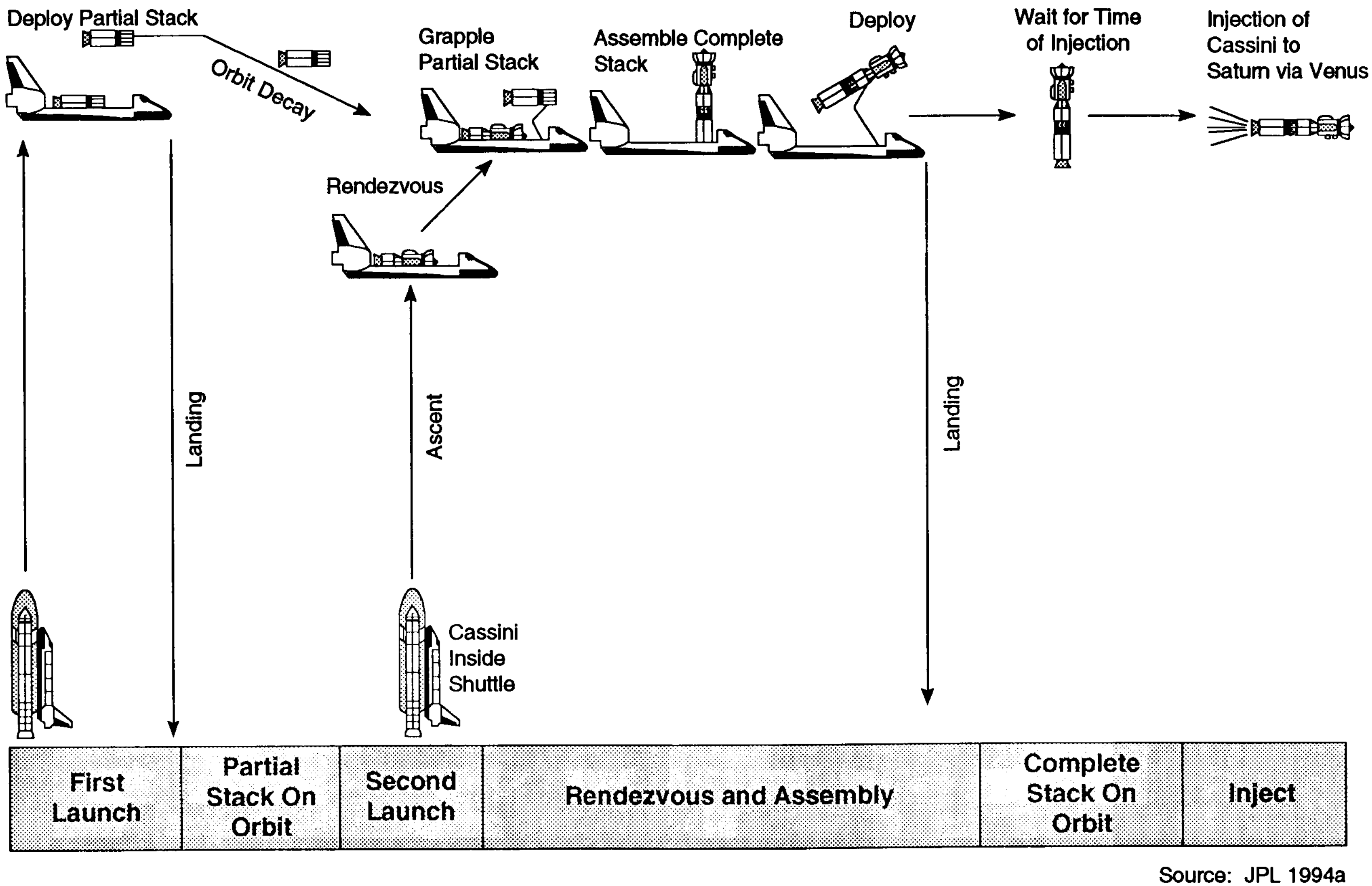


FIGURE 2-12. EXAMPLE OF A DUAL SHUTTLE LAUNCH AND SPACECRAFT/UPPER STAGE ASSEMBLY ON ORBIT

Probe temperature control during the extended Earth orbital phase, the probe coast time would have to be reduced. The reduced coast time would result in a larger required orbiter deflection maneuver, which in turn would result in fewer Titan encounters. Therefore, launching of the spacecraft via the dual Shuttle in March 1999 or August 2000 would provide less science return than that expected from launching the spacecraft with the Titan IV (SRMU)/Centaur on the same launch opportunities (JPL 1994a).

Mission Contingencies

The Shuttle has intact abort capabilities to safely return the Shuttle crew and cargo to a suitable landing site in the event that specific failures (e.g., engine loss, electrical/auxiliary power failure) occur during the early phases of launch. There are three U.S. abort landing sites (i.e., Kennedy Space Center, Edwards Air Force Base, and White Sands Space Harbor). For every Shuttle mission, foreign abort sites are also identified in cooperation with host governments and would have to be identified if this mission were implemented.

2.3.2 Launch Opportunities

As stated previously, interplanetary missions can only be launched during specific opportunities. The primary launch opportunity for the dual Shuttle launches occurs in March 1999. A backup launch opportunity has been identified in August 2000, using a VEEGA trajectory.

As illustrated in Figure 2-12, launches would occur prior to the insertion date of March 1999 or August 2000, depending on the launch opportunity. Two Shuttle launches would be required with on-orbit mating of the upper stages with the spacecraft.

2.3.3 Spacecraft Description

The spacecraft would be essentially identical to the one described in Section 2.2.3.

2.3.4 Spacecraft Electrical Power and Heating Systems

The spacecraft electrical power and heating systems would be identical to those described in Section 2.2.4 for the Proposed Action. There may be a reduction in the number of RHUs for the Huygens Probe.

2.3.5 Spacecraft Propulsion Module Subsystem

The propulsion module subsystem would be as described in Section 2.2.5 for the Proposed Action.

2.3.6 Launch Vehicle (Dual Shuttle) Configuration

The Shuttle (see Figure 2-11) consists of an orbiter, a piloted (manned) reusable vehicle, which is mounted on a non-reusable (expendable) External Tank (ET) and two recoverable and reusable Solid Rocket Boosters (SRBs) (NASA 1988a).

Propulsion for the orbiter is provided by three engine systems: three Space Shuttle Main Engines (SSMEs), two SRBs, and an Orbital Maneuvering System (OMS). The SSMEs and the SRBs together provide thrust for lift-off and for the first 2 minutes or so of ascent. After the SRBs burn out, the SSMEs thrust the orbiter on, almost to orbit. After the SSMEs are shut down, the OMS provides thrust for attaining orbit, maneuvering while in orbit, and decelerating out of orbit (NASA 1988a).

The reusable SSMEs burn for about 8 minutes, and the nozzles are gimbaled for steering. The fuel is liquid hydrogen and the oxidizer is liquid oxygen, both stored in the ET (NASA 1988a).

The SRBs burn in parallel with the SSMEs to provide the initial ascent thrust. Each SRB (steel cased) weighs approximately 586,500 kg (1.293 million lbs), providing 12.76 million N (2.9 million lbs) of thrust at sea level. The nozzles on the SRBs are also gimbaled for steering. The propellant for the SRBs is a composite-type solid propellant formulated of polybutadiene acrylonitrile (PBAN) terpolymer binder, ammonium perchlorate, and aluminum powder (NASA 1988a).

The OMS includes two engines with gimbaled nozzles for steering. The fuel is MMH and the oxidizer NTO.

2.3.7 Mission Timeline

For a launch with the Space Shuttle, a typical mission timeline would be as follows (NASA 1988a):

- Phase 0 Prelaunch/Launch (T - 8 hr to T - 6.6 s)-This phase begins with the initiation of loading the liquid hydrogen (LH_2) and liquid oxygen (LO_2) into the Shuttle's ET at T - 8 hr and ends at SSME ignition at T - 6.6 s.

- Phase 1 Launch and Ascent (T- 6.6 s to T + 128 s)-This phase begins with SSME ignition at T - 6.6 s to SRB ignition at T - 0 and ends with SRB burnout and separation at T + 128 s. At T + 2 s, the vehicle would have sufficient velocity and attitude control to avoid striking the launch tower if the left SRB were to fail and lose thrust. The vehicle would clear the tower at T + 7 s. During this period, the telemetry and visual cues may be insufficient to permit use of the Flight Termination System (FTS). After T + 10 s, however, the FTS would be available. In the event of an accident, the IIP of vehicle debris would pass from land to the ocean at about T + 1 7 s and would be in deep water by T + 30 s, assuming a normal trajectory. After T + 30 s and before SRB burnout and separation at T + 128 s, the Shuttle would pass through the period of maximum dynamic pressure and SSME throttling. At T + 57 s, the Shuttle would reach an altitude where the results of an accidental fuel release would no longer threaten KSC or the local Florida region.

- Phase 2 Second Stage (T + 128 s to T + 532 s)-This phase begins with SRB separation at T + 128 s and ends with MECO at T + 532 s. Normally, the IIP for Africa landfall would occur at about T + 500 s and would reach the Indian Ocean at about T + 505.5 s.

- Phase 3 On Orbit (T + 532 s to T + 24,000 s) - This phase begins at T + 532 s and ends approximately T + 6 hr, just prior to the deployment of the spacecraft and the upper stage from the Shuttle's cargo bay. An OMS burn would be required at the beginning of this phase.

- Phase 4 Payload Deploy (T + 24,000 s to Earth Escape) - This phase begins with the deployment of the spacecraft and upper stage from the cargo bay.

- lnterplanetary Cruise (Earth Escape to end of mission) - This phase covers the interplanetary cruise between Earth and Saturn.

The 1999 mission alternative would have some differences in the Shuttle mission timeline due to the on-orbit mating of the upper stages and the spacecraft by astronaut extra-vehicular activity.

2.3.8 Range Safety Considerations

2.3.8.1 General

Range safety encompasses all activities relevant to launch vehicles at KSC. See Section 2.2.8.1 for details.

2.3.8.2 Flight Termination System

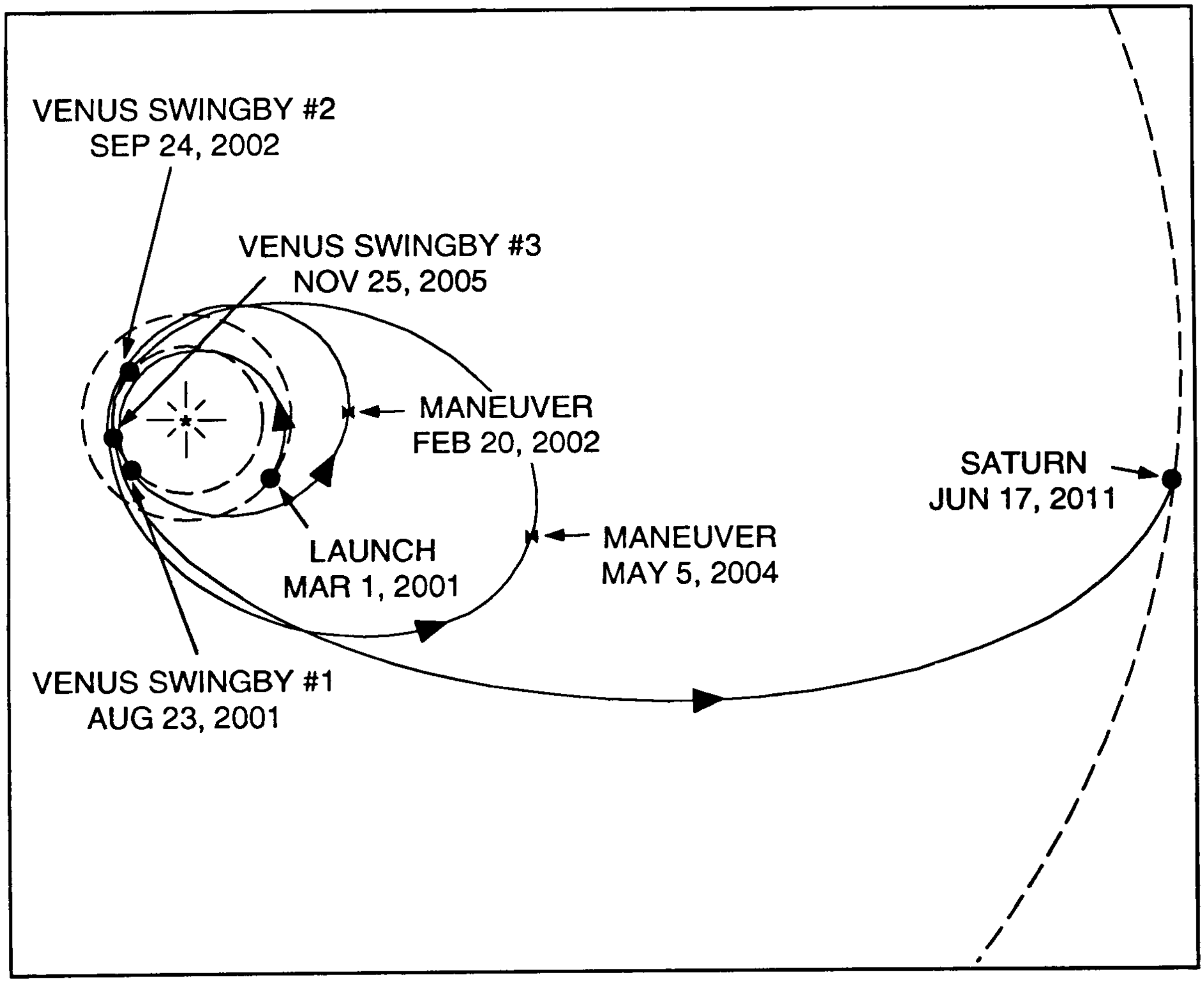
The FTS on the Shuttle, when activated from a ground signal, would destroy the two SRBs and the ET. The onboard systems for the two SRBs and one ET would be connected so that if either SRB received a destruct command all three would receive it. The system for each of these would be redundant to assure reliability (NASA 1988a).

2.3.8.3 Electromagnetic Hazard Conditions

The discussion in Section 2.2.8.3 also applies to the 1999 mission alternative.

2.4 DESCRIPTION OF THE 2001 MISSION ALTERNATIVE

The 2001 mission alternative would be similar to the Proposed Action in that it would include the Cassini spacecraft with the Huygens Probe and the three-RTG electrical power system, as described in Sections 2.2.3 through 2.2.5. This mission alternative, however, would insert the Cassini spacecraft into a non-Earth-Gravity-Assist trajectory. The launch vehicle would be the Titan IV (SRMU)/Centaur and would have a similar mission timeline as described in Section 2.2.7. Neither the Titan IV (SRM)/Centaur nor the dual Shuttle launch would be capable of launching and injecting the Cassini spacecraft into this trajectory. This mission alternative would have a primary launch opportunity during the first 2.5 weeks of March 2001 from CCAS and would use a 10.3-yr Venus-Venus-Venus-Gravity-Assist (VVVGA) trajectory, as depicted in Figure 2-13. The first Venus swingby would occur in August 2001, the second in September 2002, and the third in November 2005, arriving at Saturn in June 2011 for the 4-year tour of the Saturnian system (JPL 1994a).



Source: JPL 1993e

FIGURE 2-13. CASSINI MARCH 2001 VVVGA INTERPLANETARY TRAJECTORY

To accommodate the amount of maneuvering associated with the VVVGA trajectory, the Cassini spacecraft would have to be fitted with larger propellant tanks than used in the Proposed Action, to hold about 20 percent more propellant (increase of about 600 kg [1,323 lb]). In addition, a different spacecraft propulsion engine would have to be used-a rhenium engine. The rhenium engine, currently not space-qualified, is a higher performance engine than the currently designed engine for the spacecraft. A rhenium engine is a spacecraft main engine with a rocket chamber fabricated from rhenium and an internal oxidation-resistant iridium coating. It can perform at 2,200°C (4,000°F) which enables it to run without the need for cooling the rocket chamber. A version of this engine has been in development for NASA missions. Another version of the engine is being developed for commercial spacecraft. To make a rhenium engine available for this mission alternative, NASA would have to invest additional funds to complete engine development and make it flight ready. Only a high performance rhenium engine would have the potential capability to perform all the interplanetary maneuvers necessary to use the VVVGA trajectory and still leave enough propellant for maneuvers in orbit around Saturn. Even with the larger tanks, the amount of propellant available for spacecraft maneuvering upon reaching Saturn would be limited. The number of Titan flybys would have to be reduced from 35 (the Proposed Action) to 21, the SOI burn delay would have to be eliminated, and the initial orbit period would have to be increased significantly. This would reduce the amount of science return obtained from the Titan flybys and from close-in observation of Saturn's rings just prior to orbit insertion (JPL 1994a).

The Titan IV (SRMU)/Centaur would not have any launch mass margin to perform the VVVGA trajectory. Therefore, any increase in spacecraft mass would probably exceed the launch vehicle lift capability. The spacecraft would also be older at the time of arrival at Saturn compared with the Proposed Action's primary launch opportunity. The longer cruise time would also decrease the RTG electrical power output for the science experiments. The longer cruise time increases the probability of spacecraft failure and, therefore, loss of science.

There is no non-Earth-Gravity-Assist backup launch opportunity using the Titan IV (SRMU)/Centaur launch vehicle. A May 2002 VEEGA trajectory exists as a backup to the March 2001 VVVGA. This VEEGA trajectory would have characteristics similar to the December 1997 or March 1999 VEEGA trajectory for the Proposed Action's contingency launch opportunities.

A delay of the Cassini mission until the 2001 mission alternative would disrupt and could possibly strain the international partnerships formed to develop the Cassini Orbiter, Huygens Probe, and other space-related projects.

2.5 DESCRIPTION OF THE NO-ACTION ALTERNATIVE

The No-Action alternative would cancel the Cassini mission to Saturn. No further preparations would be made for the mission and the mission would not be implemented. None of the mission-specific science objectives would be realized. Some tangible benefits discussed in Section 1.4 (e.g., technological advances, such as a solid state data recorder and gyros) from the development of and planning for the Cassini mission have already been realized. Any future benefits directly attributed to the mission would be forfeited. Cancellation of the mission could seriously disrupt and strain the international partnerships NASA has formed to develop space-related projects.

2.6 EVALUATION OF LAUNCH VEHICLES, MISSION TRAJECTORIES, AND SPACECRAFT POWER SYSTEMS

In addition to the basic engineering design of the spacecraft, the other key components associated with the mission are the launch vehicle, the interplanetary trajectory, and the power system for the spacecraft's electrical requirements. These must function together to satisfy the requirements of the mission. Each of the key components were evaluated in developing the Proposed Action and alternatives addressed in this EIS.

The key components (see Figure 2-14) were evaluated in terms of 1) technical feasibility, 2) ability to satisfy the science objectives for the mission, and 3) potential for reducing the possible environmental impacts associated with the mission design for the Proposed Action. A component must provide the performance and operating characteristics required by all other components of the spacecraft and launch vehicle without imposing new requirements (JPL 1994a). The components must, of necessity, be compatible with all the other components for a particular mission. To be considered technically feasible, a component must have been tested for space-flight applications or must be in the development stages on a timetable consistent with the Cassini mission schedule. The requirement for the mission components to satisfy the science objectives is essential because the mission must provide useful information in a timely manner. The mission components were also evaluated with respect to relative environmental impacts.

2.6.1 Launch Vehicles

Performance (lift capabilities) and availability are the overriding considerations in the selection of a launch vehicle for a planetary mission because the launch vehicle must be able to reliably place the spacecraft into the proper trajectory. If the launch vehicle does not have adequate lift capacity (including sufficient margins), then it does not merit further evaluation (JPL 1994a). Performance is derived from an integrated launch vehicle consisting of the booster and an upper stage. The booster operates from the ground to insert the upper stage and payload into a desired parking orbit. The upper stage then injects the payload from the parking orbit into the desired interplanetary trajectory. In certain cases, the booster alone cannot place a fully loaded upper stage and payload (such as with the Cassini spacecraft) into parking orbit. Therefore, a portion of the propellant for the upper stage is used to insert the payload into the parking orbit. For instance, the Titan IV (SRMU)/Centaur for the Proposed Action would require that about 20 percent of the Centaur (upper stage) propellant be used to place the Centaur and the Cassini spacecraft into a low Earth parking orbit.

It is generally not possible to arbitrarily mix and match boosters and upper stages to create a launch vehicle configuration to deliver the payload to the desired trajectory. Upper stages are usually designed for use with certain boosters. Thus, boosters and upper stages must be compatible in both performance and integration. In addition, the size of the PLF (on expendable launch vehicles) or the cargo bay (on the Shuttle) is considered in

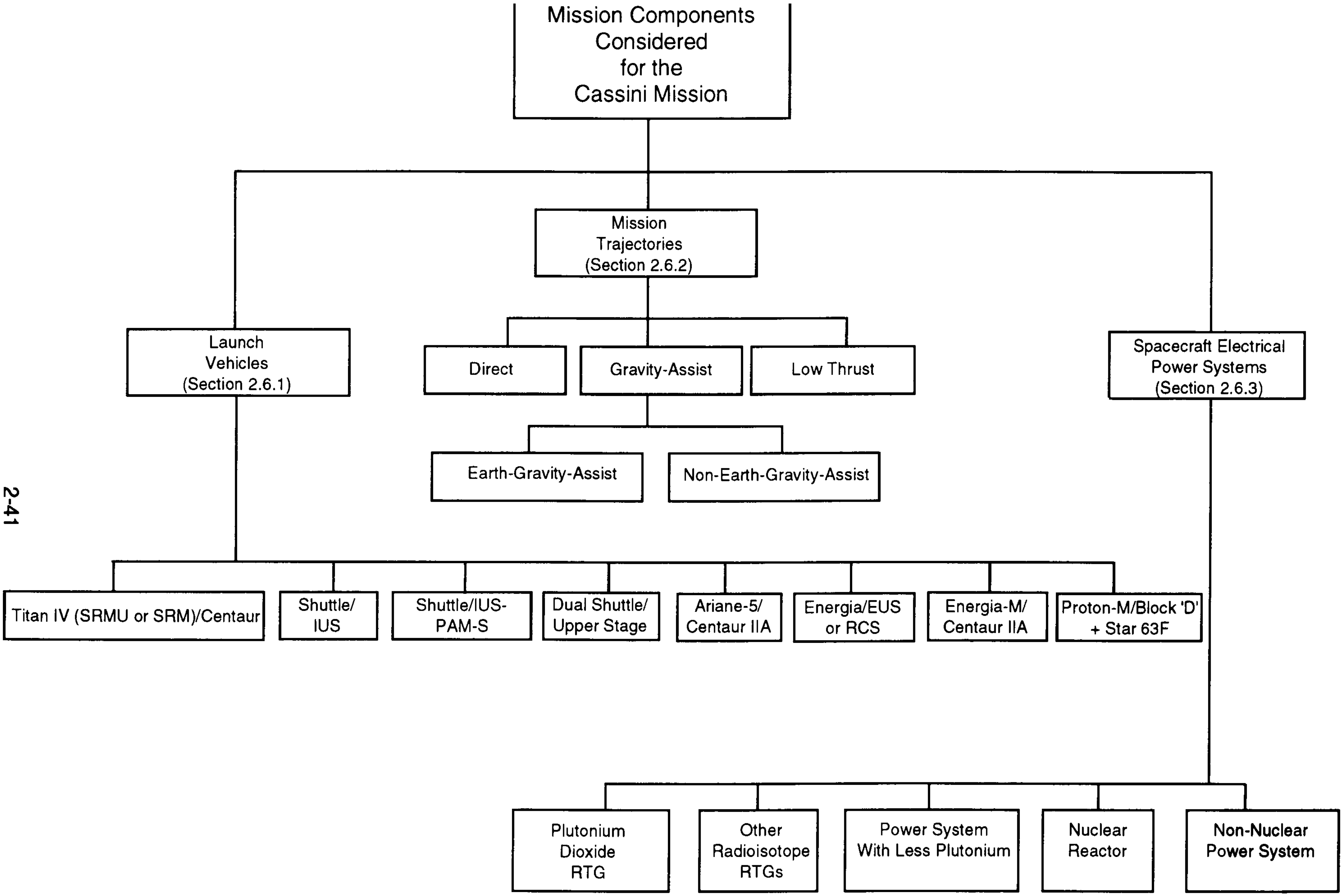


FIGURE 2-14. MISSION COMPONENTS CONSIDERED FOR THE CASSINI MISSION

the configuration of the launch vehicle. For example, the Titan IV (SRMU) can use a variety of PLF sizes, from 17.1 m to 26.2 m (56 ft to 86 ft) long (Martin Marietta 1992); the cargo bay on the Shuttle is 18.3 m (60 ft) long by 4.6 m (15 ft) in diameter (NASA 1988a). The volume available in the PLF or cargo bay for the spacecraft would depend on the size of the upper stage.

In its assessment of launch vehicles, JPL considered as technically feasible those systems that are technically mature and space qualified, and as potentially feasible, those launch vehicles that are under development on a timetable that if met, would be consistent with the schedule for the Cassini mission. Only those systems that would provide the required performance and operating characteristics without imposing any new requirements upon the mission or other mission components were considered in detail by JPL. This narrowed the launch vehicles for the Cassini mission to the U.S. and foreign launch vehicles discussed in Sections 2.6.1.1 and 2.6.1.2, respectively. Table 2-4 provides a summary comparison of the U.S. launch vehicles (JPL 1994a). The mission trajectories are discussed in Section 2.6.2 of this EIS.

2.6.1.1 U.S. Launch Vehicles

Two U.S. launch vehicles (Titan IV and Shuttle) are space-qualified, available and, in certain configurations have the lift capability to place the spacecraft into a low-Earth orbit (LEO) from which it could then be injected into a feasible trajectory to Saturn. The following configurations were evaluated for the Cassini mission:

- Titan IV (SRMU or SRM)/Centaur (for the Proposed Action)
- Shuttle (i.e., Space Transportation System [STS]/Inertial Upper Stage (IUS)
- Shuttle/IUS-Payload Assist Module Special (PAM-S)
- Shuttle/Unspecified Upper Stage(s)

The Titan IV (SRMU or SRM)/Centaur is described in Section 2.2.6 of this EIS. The Titan IV (SRMU) is the most capable U.S. expendable heavy lift vehicle with the capacity of placing 22,680 kg (50,000 lb) into LEO. The existing Titan IV (SRM), successfully flown in ten out of eleven missions (as of March 3, 1995), has an LEO capability of about 18,140 kg (40,000 lb). While either vehicle would be capable of launching the Cassini spacecraft during the launch opportunities of the Proposed Action, the science return for the SRM configuration would be less than that of the SRMU configuration.

The Shuttle, discussed in Section 2.3.6 and in several NASA NEPA documents, including the Galileo and Ulysses EISs (NASA 1989b, NASA 1990), has similar lift capabilities as the Titan IV (SRMU). It has been launched 68 times with 1 failure. The Shuttle, a piloted reusable vehicle, is mounted on a non-reusable ET containing liquid hydrogen and oxygen propellants and two SRBs. The Shuttle has three main rocket engines and a cargo bay 18.3 m (60 ft) long and 4.6 m (15 ft) in diameter (NASA 1978). Crew safety guidelines prohibit the use of the powerful Centaur upper stage in the Shuttle. For interplanetary missions, less energetic solid-propellant fueled upper stages, also

TABLE 2-4. SUMMARY OF POTENTIAL U.S. LAUNCH VEHICLES

Qualifying Launch Opportunities	Titan IV (SRMU) with Centaur	Titan IV (SRM) with Centaur	Single Shuttle with IUS/PAM-S	Dual Shuttle Launch with Upper Stage Assembly On-Orbit
October 1997 VVEJGA	x	x	-	See footnote a
December 1997 VEEGA	x	x	-	See footnote a
March 1999 VEEGA	x	x	-	x
August 2000 VEEGA	x	x	-	x
March 2001 VVVGA	x	-	-	-
March 2002 VVVGA	-	-	-	-
May 2002 VEEGA[b]	x	-	-	x
1997,1998,1999 JGA	-	-	-	-
Science Return for Equivalent Launch Opportunities[c]	Would return best science	Would return less than Titan IV (SRMU)/Centaur	Not applicable	Would return less than the Titan IV (SRMU)/Centaur
Launch Vehicle Considerations	• Spacecraft needs rhenium engine for non-EGA opportunity • No non-EGA backup opportunities for March 2001	• Less performance than Titan IV (SRMU) with Centaur configuration • No non-EGA opportunities	• Not technically feasible	• Requires development of new upper stage • Assembly on-orbit increases technical complexity • No non-EGA opportunities

Source: adapted from JPL 1994a

NOTE: X Launch vehicle has sufficient capability to perform the mission with this trajectory.
- Launch vehicle does not have sufficient capability to perform the mission with this trajectory.

a. There is not enough time to develop and implement the integration design for the spacecraft, launch vehicle, and upper stage without incurring unacceptable development, integration, and schedule risk.
b. If the primary launch opportunity in March 2001 were missed, there would not be enough time to reconfigure the mission for a dual Shuttle backup launch opportunity.
c. Amount of science return expected compared to using a Titan IV (SRMU)/Centaur for the same launch opportunity.

compatible with the Titan IV vehicle, are typically used in the Shuttle. For example, the IUS was used for the Magellan and Galileo missions (in May 1989 and in October 1989, respectively), and the IUS/PAM-S was used for the Ulysses mission in 1990. For the Cassini mission, neither of these upper stages, when coupled with the Titan IV (SRMU or SRM) or the Shuttle, would be capable of placing the spacecraft into a feasible trajectory to Saturn. Therefore, launch vehicle configurations using either of these two upper stages were not considered further. Launch of the Cassini spacecraft using the Shuttle would require two Shuttle launches (separated by at least 21 days, but no more than 51 days) to place the spacecraft and upper stages into LEO, where the final on-orbit mating (of the spacecraft and upper stages) would be performed by astronauts. The upper stages would then inject Cassini into the VEEGA interplanetary trajectory. Though the dual Shuttle launch is technically feasible, there would be insufficient time to develop and implement the integration design for the spacecraft, launch vehicle, and upper stage without incurring unacceptable development, integration, and schedule risk for the 1997 launch opportunity (JPL 1994a). Thus, the dual Shuttle launch can be considered only for later launch opportunities, such as the March 1999 opportunity.

2.6.1.2 Foreign Launch Vehicles

Currently, the United States does not have any programs funded to develop a launch vehicle with a lift capability greater than the Titan IV (SRMU)/Centaur. However, the following foreign launch vehicles could potentially have similar capability to the Titan IV (SRMU)/Centaur (JPL 1994a):

- Ariane-5/Centaur IIA

- Energia/Energia Upper Stage (EUS) and/or the Retro and Correction Stage (RCS) Energia-M/Centaur IIA or Block 'DM' + Star 63F

- Proton-M/Block 'D' + Star 63F.

Though the above foreign vehicles are still in the development stage and/or have uncertain development schedules, their technical capability of launching the Cassini spacecraft can be potentially assessed. It should be noted that the use of a foreign launch vehicle raises special programmatic concerns. The following list summarizes some of these concerns:

- Substantial time would be required to analyze, develop, space-qualify and implement the integration design for the spacecraft, launch vehicle, and upper stage.

- The launch approval process for carrying a radioactive payload into space from foreign soil would require U.S. and foreign government involvement beyond the purview of this EIS.

- Transportation and security requirements specifically for the RTGs and RHUs would require U.S. and foreign government involvement and policy decisions that are beyond the purview of this EIS.

- Foreign relations and domestic/competitiveness policies are beyond the purview of his EIS (White House 1990).

- Difficulty would be associated with spacecraft and launch vehicle integration at unfamiliar overseas launch facilities.

Table 2-5 summarizes the foreign launch vehicles that JPL addressed in detail (JPL 1994a). The table identifies interplanetary trajectories and launch opportunities that can potentially be enabled by the launch vehicles (see Section 2.6.2). The table also identifies the amount of science return expected by using these launch vehicles compared with using a Titan IV (SRMU)/Centaur. Additional limitations associated with the use of these launch vehicles have also been presented in the table.

Ariane-5

The European Ariane-5, an expendable launch vehicle being developed to replace the operational Ariane-4 series by 1999, is proposed to have its first commercial flight in 1996. The Ariane-5 would be launched from Kourou in French Guiana. It would be a two-stage core vehicle with two strap-on boosters. The planned fairing size would be 4.6 m (15.1 ft) diameter and 12 m (39.4 ft) long. The capability of the Ariane-5 would be comparable to the Titan IV (SRMU) at 20,865 kg (46,000 lb) to LEO (JPL 1994a). Currently, there are no plans to develop any upper stages for this vehicle. However, the Ariane-4 H10 cryogenic third stage or Centaur IIA could potentially be modified to inject planetary payloads. (The Centaur IIA is the version of the Centaur cryogenic upper stage which is currently used by the U.S. Atlas II family of launch vehicles). Use of the Ariane-5/Centaur IIA to launch the spacecraft would involve technical complexities with substantial analysis, integration and qualification of the upper stage to the spacecraft. In addition, major launch pad and operational modifications would have to be implemented for the Centaur IIA, the spacecraft, and the necessary interfaces. The payload fairing would also be too small for the currently designed spacecraft. Regardless of the current developmental status and potential of the Ariane-5 with the Centaur IIA, it would not enable any new interplanetary trajectories beyond those identified for the Proposed Action using the Titan IV (SRMU)/Centaur. Therefore, these complexities, the lack of any new trajectories and the programmatic concerns eliminate the Ariane-5 from further consideration for this mission.

Energia

The Russian Energia, the most powerful launch vehicle built in the last decade, has not flown since November 1988. It has had a brief flight history consisting of two test flights, with an upper stage failure during one of these test flights. The Energia was designed to be able to place approximately 93,070 kg (205,000 lb) of payload into a 200 km (125 mi) sub-orbital trajectory (JPL 1994a). Payloads can then be boosted into Earth orbit either by using their own propulsion systems or by an upper stage. The Energia

TABLE 2-5. SUMMARY OF POTENTIAL FOREIGN LAUNCH VEHICLES

Qualifying Launch Opportunities	Ariane-5 with Centaur IIA	Energia with EUS/RCS	Energia-M with Centaur IIA	Proton-M with Block 'D' + Star 63F	Split Mission Proton-M with Block 'D' + Star 63F
October 1997 VVEJGA	See footnote a	See footnote a	See footnote a	See footnote a	See footnote a
December 1997 VEEGA	See footnote a	See footnote a	See footnote a	See footnote a	See footnote a
March 1999 VEEGA	x	x	x	x	x
August 2000 VEEGA	x	x	x	x	x
March 2001 VVVGA	x	x	x	-	-
March 2002 VVVGA	x	x	x	x	x
May 2002 VEEGA	x	x	x	-	-
1997,1998,1999 JGA	-	x	-	-	-
Science Return for Equivalent Launch Opportunities[b]	Would return full sciences; for non-EGAs, meets minimum requirement	Would return full science; for non-EGAs, meets minimum requirement	Would return full science; for non-EGAs, meets minimum requirement	Would return less science	Would return full science
Launch Vehicle Considerations[c]	• Needs development of new upper stage configuration • PLF is too small for the Cassini spacecraft • Spacecraft needs rhenium engine performance for March 2002	• Development of upper stages is uncertain • Lack of information available with respect to the booster and upper stages to assess technical feasibility	• Needs development of new upper stage configuration • 2002 VVVGA opportunity has a cruise duration of 12 years • March 2002 VVVGA requires a substantially larger bipropellant tank	• Needs development of new upper stage configuration • PLF is too small for the Cassini spacecraft • No flight termination system	• Needs development of new upper stage configuration • PLF is too small for the Cassini spacecraft • Twice the number of gravity-assists • Increased mission risk because requires two successful launches and two successful Saturn orbit insertions for full science return

Source: JPL 1994a

NOTE: X Launch vehicle has sufficient capability to perform the mission with this trajectory.
- Launch vehicle does not have sufficient capability to perform the mission with this trajectory.

a. Even it this launch configuration was determined to be available and technical feasible, there would be insufficient time to develop and implement the integration design for the spacecraft, launch vehicle and upper stage without incurring unacceptable development, integration and schedule risk.
b. Amount of science return expected compared to using a Titan IV (SRMU)/Centaur for the same launch opportunity.
c. Transportation and security requirements for the RTGs and RHUs would require both U.S. and foreign government involvement and policy decisions that are beyond the purview of the EIS.

consists of a cryogenic core stage with four to eight strap-on boosters. A 6.7 by 42.1 m (22 by 138 ft) cargo container is side-mounted to carry the upper stage and payload. Depending on the booster and upper stage configuration, the Energia could potentially enable Jupiter-Gravity-Assist (JGA) and direct trajectories for Cassini. Russia is reportedly developing two upper stages for the Energia. These are the Retro and Correction Stage (RCS) (also known as Block to Transfer and Correction [BTK], and the Energia Upper Stage (EUS) (JPL 1994a). Operational and flight schedules for these upper stages are uncertain at this time. The availability of the Energia has not been established and the future of this launch vehicle is uncertain. Due to insufficient information on the status of development and performance of the Energia core vehicle, an accurate evaluation of the Energia's technical performance (e.g., the operation and payload integration details and accident analysis) for the Cassini mission cannot be made. The Energia is eliminated from further consideration for the Cassini mission due to the lack of adequate technical data on its performance and due to the programmatic concerns.

The Energia-M is designed to be a smaller version of the Energia discussed above. It would use two of the Energia's strap-on boosters. Also, the core engine would be a scaled down version of the Energia's. Its payload fairing would be 5.1 m (16.7 ft) in diameter and 21.5 m (70.5 ft) long, in line with the core. The Energia-M would place a 30,845 kg (68,000 lb) payload in near low Earth orbit (sub-orbital) (JPL 1994a). The Centaur IIA (upper stage) could potentially be used with the Energia-M for the Cassini mission. However, major launch pad (i.e., Baikonur Cosmodrome in Kazakhstan) and operational modifications would be necessary to implement and launch such a configuration. The Energia-M has only undergone a full scale model launch pad compatibility evaluation at Baikonur. The Centaur IIA and the Cassini spacecraft would fit the Energia-M's 21.5 m (70.5 ft) long PLF. The less energetic (compared to the Centaur IIA) Block 'DM'+ Star 63F upper stage could also be used. However, the Energia-M configuration would not enable any new interplanetary trajectories different from those identified for the Proposed Action using the Titan IV (SRMU)/Centaur. The actual development of the Energia-M has not begun; therefore, future availability is highly uncertain. Thus, for these reasons and the programmatic concerns, the Energia-M is eliminated from further consideration for the Cassini mission.

Proton

When available, the Russian Proton-M, a modernized version of the Proton D-1-e which has flown over 200 missions, is expected to have the capability of placing about 23,700 kg (52,250 lb) in LEO. Two versions of the Proton upper stage, i.e., the Block 'DM' and the Block 'D' are available. Each is a single engine, liquid-fueled, three-axis stabilized, inertially guided, restartable stage. The 'D' version is lighter and more energetic than the 'DM' version. In order for the Proton-M with the 'D' version upper stage to be potentially feasible for the Cassini mission, an additional upper stage (e.g., a Star 63F) and a larger payload fairing would be required. This configuration (i.e, Proton-M/Block 'D' + Star 63F) could provide a performance comparable to the Titan IV (SRM)/Centaur and it would not be capable of enabling any new interplanetary trajectories different from those identified for the Titan IV (SRMU)/Centaur. Because of these technical issues, its unavailability, and the programmatic concerns associated with foreign launch vehicles, the Proton-M is eliminated from further consideration for the Cassini mission.

The Proton-M with Block 'D' + Star 63F could also enable a split mission. In this configuration, two smaller spacecraft would be launched to Saturn, with only one spacecraft carrying the Huygens Probe to Titan. This approach would require the use of a total of five RTGs. Due to technical complexities and programmatic concerns, the Proton-M with Block 'D' + Star 63F is eliminated from further consideration for the Cassini mission.

2.6.1.3 Summary of Launch Vehicles

Due to the technical complexities, the lack of adequate technical data, and the programmatic concerns associated with the use of foreign launch vehicles, they are eliminated from further consideration for the Cassini mission. For a 1997 launch, the U.S. Titan IV (SRMU)/Centaur is the only feasible launch vehicle to accomplish all of the planned science objectives with a full science return. If the Titan IV (SRMU)/Centaur were not available, then the less powerful U.S. Titan IV (SRM)/Centaur would be used to implement the mission. Using the Titan IV (SRM)/Centaur would necessitate a reduction in the mass of the propellant on the spacecraft. This action would limit the amount of maneuvers at Saturn and would therefore result in a reduced science return when compared with a launch on the Titan IV (SRMU)/Centaur. In addition, an opportunity to implement the mission exists in 1999 using the Shuttle. Using the Shuttle, however, would require two launches, separated by a minimum of 21 but no more than 51 days. The science return in this case would be less than that for the 1997 Titan IV (SRMU)/Centaur launch.

2.6.2 Mission Trajectories

Trajectories for interplanetary missions are either ballistic or nonballistic. In a ballistic trajectory, the spacecraft's flight path is shaped only by the gravitational influences of the Sun and planets. A nonballistic trajectory includes, in addition to the gravitational influences, velocity changes produced by the spacecraft's chemical propulsion system. All planetary missions flown to date have used nonballistic trajectories. These trajectories, therefore, consist of two or more ballistic trajectory arcs connected by spacecraft maneuvers. Both ballistic trajectories and nonballistic trajectories can be classified as either direct trajectories (see Section 2.6.2.1) or gravity-assist trajectories (see Section 2.6.2.2). Another class of nonballistic trajectories, low-thrust trajectories, can be used by spacecraft with a low-thrust propulsion system (see Section 2.6.2.3) (JPL 1994a).

2.6.2.1 Direct Trajectory

A direct trajectory typically uses chemical propulsion, does not use planetary gravity-assist swingbys, and can be either ballistic or nonballistic. This trajectory shortens flight time, lowers operational costs, and reduces mission complexity. It also reduces the likelihood of accidental reentry into the Earth's atmosphere. Opportunities to use a given direct trajectory repeat whenever the same relative planetary alignment (phasing) occurs. Currently, none of the available U.S. launch vehicles has the lift capability necessary to launch the Cassini spacecraft into a direct trajectory (JPL 1994a). A U.S. vehicle is not being developed with this capability; therefore, this type of trajectory is infeasible at this time.

2.6.2.2 Gravity-Assist Trajectory

A gravity-assist trajectory depends on chemical propulsion and uses one or more swingbys of planets to significantly reduce requirements for either the mission's launch injection energy, arrival velocity, or flight time. Several past missions have used gravity-assist swingbys. Pioneer 11, launched in 1973, used a Jupiter-Gravity-Assist to allow a flyby of Saturn. Mariner 10, also launched in 1973, used a Venus-Gravity-Assist to swing by Mercury. Voyagers 1 and 2 (launched in 1977) each used a gravity-assist swingby of Jupiter to reach Saturn. Voyager 2 also used a gravity-assist at Saturn to go to Uranus and then a gravity-assist of Uranus to continue to Neptune. The Galileo spacecraft used swingbys of Venus in 1990 and Earth in both 1990 and 1992, which will enable the spacecraft to encounter Jupiter in 1995 (JPL 1994a).

The Proposed Action would use a Venus-Venus-Earth-Jupiter-Gravity-Assist (VVEJGA) involving an Earth-Gravity-Assist swingby. Gravity-assist swingbys of Earth represent a very effective means of increasing the mass that can be delivered to an outer planet, such as Saturn. Trajectories using combinations of Venus and Earth-gravity-assist swingbys, such as Venus-Earth-Gravity-Assist (VEGA), Venus-Venus-Earth-Gravity-Assist (VVEGA), and Venus-Earth-Earth-Gravity-Assist (VEEGA), have an important advantage since their launch energy requirements are low. The addition of a Jupiter-Gravity-Assist after the final Earth or Venus swingby, such as Venus-Venus-Earth-Jupiter-Gravity-Assist (VVEJGA) or Venus-Venus-Venus-Jupiter-Gravity-Assist (VVVJGA), makes it possible to deliver a spacecraft to an outer planet beyond Jupiter with a shorter flight time and/or with lower post-launch propellant requirements (JPL 1994a).

Gravity-assist trajectories that use planets other than the Earth result in a lower level of potential environmental impacts because they eliminate the possibility of an Earth-targeted swingby reentry accident and any associated environmental impacts. For the Cassini mission, JPL performed detailed analyses of numerous trajectories that would not use Earth swingbys, including Jupiter-Gravity-Assist, Mars-Gravity-Assist, Mars-Venus-Gravity-Assist, Mars-Jupiter-Gravity-Assist, Venus-Gravity-Assist, and Venus-Jupiter-Gravity-Assist (JPL 1994a).

A Jupiter-Gravity-Assist (JGA) greatly reduces the required launch injection energy while retaining most of the advantages of a direct trajectory. Even with the reduced injection energy, however, a JGA with launch opportunities in 1997, 1998, and 1999 for Cassini would still require a launch vehicle with performance capabilities (lift and injection) that would exceed that of the Titan IV (SRMU)/Centaur.

The planet Mars was evaluated for a gravity-assist swingby for missions to outer solar system planets. Because of its relatively low mass (about one-tenth that of the Earth), however, Mars is not effective in increasing the energy of a spacecraft trajectory during a single swingby. A Mars swingby would be most useful if the spacecraft returned to Earth for a gravity-assist after the Mars swingby before heading to Saturn. The next launch opportunity using an Earth-Mars combination would be in 2011, which is outside the timefrarne for the Cassini mission. Because the Earth is required, the Earth-Mars combination would not present any environmental advantage over any of the currently identified trajectories to Saturn. In addition, the combinations of Mars with either Venus or Jupiter do not provide any advantages (JPL 1994a).

JPL also identified non-Earth gravity-assist trajectories to Saturn using either Venus alone in a VVVGA or Venus combined with Jupiter for a Venus-Venus-Venus-Jupiter-Gravity-Assist (VVVJGA) (including the March 2002 opportunity noted in Table 2-4) (JPL 1994a). Several VVVGA opportunities to Saturn exist between 1996 and 2004 but only the 1996 and 2001 opportunities might be feasible using the Titan IV (SRMU)/Centaur. For the Venus-Jupiter combination, 1996 is the first opportunity and it becomes available again 17 to 20 years later. Of all these trajectories, three were found to be potentially feasible using the Titan IV (SRMU)/Centaur: 1) a VVVGA with a March 1996 launch date, 2) a VVVJGA with a March 1996 launch date, and 3) a VVVGA with a March 2001 launch date (JPL 1994a).

Preparations for launch are not feasible for either a 1996 VVVGA or a 1996 VVVJGA trajectory due to significant schedule and technical issues that would have to be addressed before launch. Accelerating development of the Cassini spacecraft and Huygens Probe for either of these 1996 launch opportunities, 1.5 years earlier than the Proposed Action, would require an accelerated equipment development schedule from NASA, the European Space Agency (ESA), and the Italian Space Agency (ASI). There would not be enough time to develop and implement the integration design for the spacecraft, launch vehicle, and upper stage without incurring unacceptable development, integration, and schedule risk, consequently increasing developmental risk and the risk of spacecraft component failures in flight.

The March 2001 VVVGA, however, would be a viable trajectory (see Section 2.4 for a detailed discussion). This trajectory would be technically feasible to implement with the Titan IV (SRMU)/Centaur. Without an EGA, it would eliminate the possibility of an inadvertent reentry during an Earth swingby, as well as any of the associated environmental impacts. However, this trajectory would not rule out the possibility of a long-term inadvertent reentry (see Section 4.1.5.3).

2.6.2.3 Low-Thrust Trajectory

A low-thrust trajectory requires the use of low-thrust propulsion systems with thrust acceleration levels less than one ten-thousandth of the Earth's gravity with specific impulses that are 5 to 50 times higher than that of a chemical propulsion system. However, low-thrust trajectories require propulsion systems that are not available or require significant development. Low-thrust propulsion systems that have been studied over the last three decades for unmanned planetary missions include the Solar-Electric Propulsion (SEP), Solar-Thermal Propulsion (STP), Nuclear-Electric Propulsion (NEP), and Solar Sail. The SEP, the most mature and best understood system, uses large solar arrays to provide electrical power to a number of modular electric bombardment thrusters using xenon as a propellant. The STP concept uses large solar concentrators to heat hydrogen or some other working fluid, which is then discharged through a nozzle to produce thrust. The NEP combines a small nuclear reactor with a high-power thruster. Thrust for the Solar

Sail is produced by momentum transfer from sunlight falling on a large, flat, very lightweight membrane. These types of propulsion systems, except for NEP, can generally be operated only near the Sun (JPL 1994a) and, therefore, would not be feasible near Saturn, which is 9.3 AU from the Sun. Spacecraft for these solar-dependent low-thrust missions would need to be augmented by chemical propulsion systems for near-Saturn maneuvers, diminishing or eliminating any payload mass benefit from the otherwise low-thrust system. Moreover, none of these low-thrust technologies, including NEP, is in a state of development that allows commitment to a launch in the 1997 to 2001 timeframe consistent with the mission goals and objectives (JPL 1994a).

2.6.3 Spacecraft Electrical Power Systems

Electrical power generating systems comprise an energy source and an energy conversion subsystem. The available energy sources include the Sun, chemicals in fuel cells or batteries, and heat from either radioactive decay, nuclear fission (reactors), or the combustion of fuels. The energy conversion subsystem transforms energy into electricity using, for example, photovoltaic cells, thermoelectric couples, or dynamic conversion machinery. Only certain combinations of energy source and energy conversion subsystems are inherently compatible.

Other energy sources for the RTGs, if available for Cassini, that could potentially reduce or eliminate the environmental risks associated with the plutonium dioxide used in the RTGs were evaluated. Power systems based on such sources must also satisfy the electrical power system performance criteria discussed in Section 2.2.4.1. The other power systems considered for Cassini include those that: 1) replace the plutonium (mainly Pu-238) dioxide fuel in the RTGs with a less potentially hazardous radioisotope, 2) implement power system designs that require less plutonium dioxide fuel, 3) use a nuclear reactor, or 4) use a power system based on a non-nuclear energy source (JPL 1994a).

2.6.3.1 Other Radioisotope RTGs

The principal concern for using plutonium dioxide fuel in RTGs is the potential radiation health and environmental hazards created if the fuel is released into the environment following an accident. In principle, any radioisotope with a half-life long enough to provide sufficient power throughout the Cassini mission and with a high enough specific activity to provide the required power with a suitably small generator can be used. Two other radioisotopes identified for RTGs are the oxides of strontium-90 (Sr-90) and curium-244 (Cm-244) JPL 1994a). An examination of their properties and production requirements indicates that neither oxide has a significant environmental advantage over plutonium dioxide. Sr-90 emits gamma radiation and Cm-244 emits both gamma and neutron radiation. Therefore, extensive shielding would be required during their production and handling, as well as when the oxide was onboard the spacecraft. Extensive development and safety testing would also be required. In addition, production facilities for sizeable quantities of these radioisotopes are not available. Therefore, Sr-90 and Cm-244 oxides cannot be considered feasible isotopic heat sources for the Cassini power system.

2.6.3.2 Power Systems Requiring Less Plutonium Dioxide

To provide comparable power levels with less plutonium, a more efficient conversion system would be required. The thermoelectric converter on the RTG has an efficiency of 6.8 percent (DOE 1987a). Other conversion technologies considered include static systems (thermionic, thermophotovoltaic, and alkali metal thermoelectric converter [AMTEC]) and dynamic systems (Rankine, Brayton, and Stirling cycles).

The GPHS has a maximum operating temperature of 1,100°C (2,012°F). Thermionic converters are high-temperature systems operating at temperatures above 1,327°C (2,420°F), which make them incompatible with the GPHS. Thermophotovoltaic converters operate at temperatures above 1,227°C (2,240°F), again making them incompatible with the GPHS. With appropriate filters and sufficient development time, however, thermophotovoltaic converters can operate at the limiting GPHS temperatures. The AMTEC is in its developmental phase and requires the resolution of issues regarding performance, degradation, spacecraft integration, launch, lifetime, and zero gravity effects before it can be considered for a spacecraft application. The dynamic conversion systems are not sufficiently developed for use in space at this time (JPL 1994a).

All of these power systems also exhibit serious technology maturity issues that could not be resolved in a timeframe consistent with the Cassini mission requirements and, therefore, are not feasible (JPL 1994a).

An additional approach evaluated for reducing the amount of plutonium dioxide fuel needed for the mission would be to reduce the number of RTGs to two and add batteries. This would be possible on some missions if the electrical power demand is intermittent and a secondary (rechargeable) battery could be added to supply power during peak demand periods and allowed to recharge during low demand times. For the Cassini mission, however, the highest and most continuous power demand would occur during the final years of the mission when the RTG power would be at its lowest output. Current power demand profiles would require partial or total spacecraft power shutdowns to recharge the batteries so that the spacecraft could restart itself again. Not only would this procedure result in the loss of science data, but it would entail extremely high-risk wake-ups from dormant modes that have not been demonstrated for such large numbers of cycles (JPL 1994a).

2.6.3.3 Nuclear Reactors

The environmental advantage of using a nuclear reactor is that it can be launched in a nonoperating mode when the inventory of radioactive fission byproducts is very small. A nuclear reactor of a size and operating lifetime suitable for Cassini, however, does not exist nor is it being developed in the United States (JPL 1994a). A number of technical problems remain to be solved even though nuclear reactors have been launched and operated in space since 1965. Some of the challenges to reactor development and implementation for deep space, long-duration missions, such as Cassini, involve control complexity and excessive mass required for shielding. Therefore, a nuclear reactor is not a feasible power source for the Cassini mission.

2.6.3.4 Non-Nuclear Power Systems

Energy sources other than the heat generated by radioisotopes are available. They include solar energy, fuels and chemicals, and power-beaming (microwave or laser) sources. The feasibility of using any of these non-nuclear power sources in spacecraft electrical power systems is addressed below.

Solar Energy

The use of solar energy for U.S. space applications was initiated in 1958 on the Vanguard I. Since then, solar energy has played a vital role in the U.S. space program by providing electrical power for most spacecraft operating between the orbits of Mercury and Mars (i.e., 0.38 and 1.52 AU [56.6 and 226.3 million km or 35.2 and 140.6 million mi, respectively]). For the Cassini mission, the greatest electrical power requirements would occur when the spacecraft is acquiring scientific data near Saturn, between 9.0 and 9.3 AU (1.34 and 1.38 billion km [840 and 865 million mi]) from the Sun. At these distances from the Sun, the intensity of sunlight is only about 1 percent of that at Earth, and temperatures are quite low.

Solar energy as the source of electrical power for deep-space probes would be desirable were it not for the large size and mass of the resulting power-generating system. Generating spacecraft electrical power from the sun has been and continues to be the subject of several evaluations and studies. A 1981 conceptual study of the system impacts of using a concentrated solar array (CSA) on the Galileo spacecraft, launched in 1989 to Jupiter and powered by two RTGs, concluded that such an effort could be performed but would require an "extensive development effort" and that "..the severe environmental constraints and the embryonic state of CSA development indicates that CSA will not displace the RTG on the Galileo mission" (Rockey et al. 1981). For the same mission, another evaluation of the feasibility of replacing the RTGs on the spacecraft with solar arrays also concluded that the most promising solar technology, the Advanced Photovoltaic Solar Array, would not be feasible due to insurmountable mass and schedule difficulties, and that a completely new solar-powered mission to Jupiter could probably not be launched until the late 1990's or later (JPL 1989). The NASA-JPL study also indicated that "no solar technology could demonstrate any viability for missions more distant than Jupiter." In general, the present level of development of the technology would necessitate the use of large, heavy arrays of solar cells. Although the large mass and dimensions would cause numerous technical problems, such as deploying the arrays, maneuvering the spacecraft, and operating the navigation, communication, and science systems, the resultant mass is the fundamental limitation. The added mass of the solar arrays necessary to power the systems on complex planetary exploration spacecraft, such as Cassini, pushes the total mass of the spacecraft, including its propellants and scientific instruments, above the launch capability of the current generation of U.S. launch vehicles for a launch trajectory to Saturn (JPL 1994a). To accomplish the Cassini mission's science objectives, the spacecraft's size and mass must be within the launch capabilities and capacities of the Titan IV (SRMU)/Centaur and the spacecraft must be sufficiently maneuverable when deployed to acquire the desired science data.

The Sun's energy is typically harnessed by two technologies: 1) a reflective or refractive surface (i.e., an optical lens) concentrator coupled with an appropriate conversion system, such as photovoltaic cells, to convert the Sun's energy into electricity and 2) photovoltaic (solar) cells on flat nonconcentrating arrays that directly convert the Sun's energy to electricity.

Concentrators have not been demonstrated in space, and a number of significant technical problems would have to be solved before a concentrator could be considered feasible for space missions, such as Cassini. The problems include how to regulate the concentrator's temperature for acceptable performance as the spacecraft traverses a Sun-to-spacecraft range from 0.63 AU to 9.3 AU; how to predict the behavior of the optics over the mission lifetime, because small changes in the concentrator condition (e.g., yellowing, aging, and sagging) can lead to significant power losses; and how to improve the alignment of the concentrator elements due to the dependence of the concentrator's power-generating ability on the Sun's incident angle. In addition, concentrator performance depends on clear, unobscured optics, and estimating the buildup of interstellar (and Saturnian) dust on the optics would be difficult. Moreover, vibration testing of any concentrator array would have to be performed to verify post-launch optical alignments and operating characteristics in zero gravity environments. The size of the concentrator arrays that would be needed for the exploration of the Saturnian system would not easily integrate into the Titan IV (SRMU)/Centaur and would not satisfy the launch mass constraints. Furthermore, it is not clear that concentrator arrays would provide any advantage over planar arrays for this mission (JPL 1994a).

For the nonconcentrating photovoltaic arrays, two solar cell technologies, one based on silicon (Si) and the other based on gallium arsenide layered on a germanium substrate (GaAs/Ge), have been considered for the Cassini mission. Silicon solar cells have been used for space power applications since the late 1950s, and improvements in cell performance continue to be made. The development of the GaAs cells began in the 1960s but it was not until the late 1970s that their efficiencies began to equal and then exceed those of silicon. GaAs cells now offer higher efficiency (18 percent) than Si cells (13.8 percent), better efficiency at elevated temperatures, and improved radiation resistance. GaAs-based cells are more brittle than Si cells, however, and have more than twice the mass of typical Si circuits. For scientific and commercial satellites in Earth orbit, Si cells have been the historically preferred technology based on flight experience and cost (JPL 1994a). However, it is expected that the use of GaAs cells in future missions will increase due to technological maturity, flight success, and continuing cost reduction.

The most promising solar array configuration is the Advanced Photovoltaic Solar Array (APSA). The APSA is a lightweight, deployable solar array that may be suitable for long-duration interplanetary missions. The APSA design is a flexible blanket array that uses thin solar cells (Si or GaAs) to minimize mass. Environmental tests of the APSA array (i.e., vibration, acoustics, and temperature cycling) and strength/stiffness tests have been completed. Tests have been conducted to verify the performance of various mechanisms and to demonstrate its mode of deployment; however, no flight testing has been undertaken or planned. An APSA of the size required for Cassini has not been fabricated or tested under the conditions similar to those anticipated for the Cassini mission (JPL 1994a).

The natural radiation environment to which the spacecraft would be exposed on a mission to Saturn will reduce the efficiency of solar cells relative to their performance at Earth. The Cassini mission includes a planned swingby of the planet Jupiter, where the intensity of the radiation is greater than that of the Van Allen belts, due primarily to the presence of a larger magnetic field around Jupiter than around the Earth. If solar cells are to be considered potentially feasible for missions like Cassini, therefore, they must either be adequately shielded from the radiation environment (new technologies that are more radiation resistant be developed) or their significantly lower efficiencies accepted and compensated for in the spacecraft design.

Solar cell performance, particularly for Si cells, is also affected by the combined effects of low (insolation) intensity and low temperature (LILT) in a complex and interactive relationship, the components of which are not completely understood. This interaction results in an anomalous reduction in power output from the cells. The phenomenon is so irregular and random that it is impossible to predict what the actual cell performance distribution would be for any group of cells (Stella and Crotty 1987). LILT effects have been estimated for distances up to 5 AU but there are almost no data to characterize the effects for greater distances. LILT effects at Saturn have been extrapolated from existing data. Unlike Si cells, the GaAs cells are not believed to be affected by LILT to any significant degree, although recent limited testing at JPL suggests that this may not be the case given conditions at Saturn (JPL 1994a).

In addition to environmentally induced cell performance degradation, other real engineering problems, including the size and inertia of the solar array structures and array/spacecraft integration issues, significantly limit the use of solar photovoltaic technologies for long-duration interplanetary space missions such as Cassini. Spacecraft integration issues include field-of-view (FOV) restrictions, extremely long spacecraft turn times, and the potential for interference from electromagnetic and ionizing radiation. FOV difficulties arise from the size of the arrays. The size is large enough to block out significant portions of what the instruments see and necessitates more frequent spacecraft turning. Difficulties are exacerbated by the extremely long turn times associated with using large arrays. The array size (further increasing the initial mass) adds to the spacecraft's inertia, making turning more difficult and propellant-intensive.

Electromagnetic and electrostatic interference can also be generated by large arrays when the current in them fluctuates or a charge builds on nonconductive surfaces. This interference could reduce the performance of scientific and communication equipment. Solar-powered spacecraft also require battery-provided backup power during periods when the solar arrays are not illuminated (eclipsed) and during maneuvers that would require the arrays to be pointed away from the Sun.

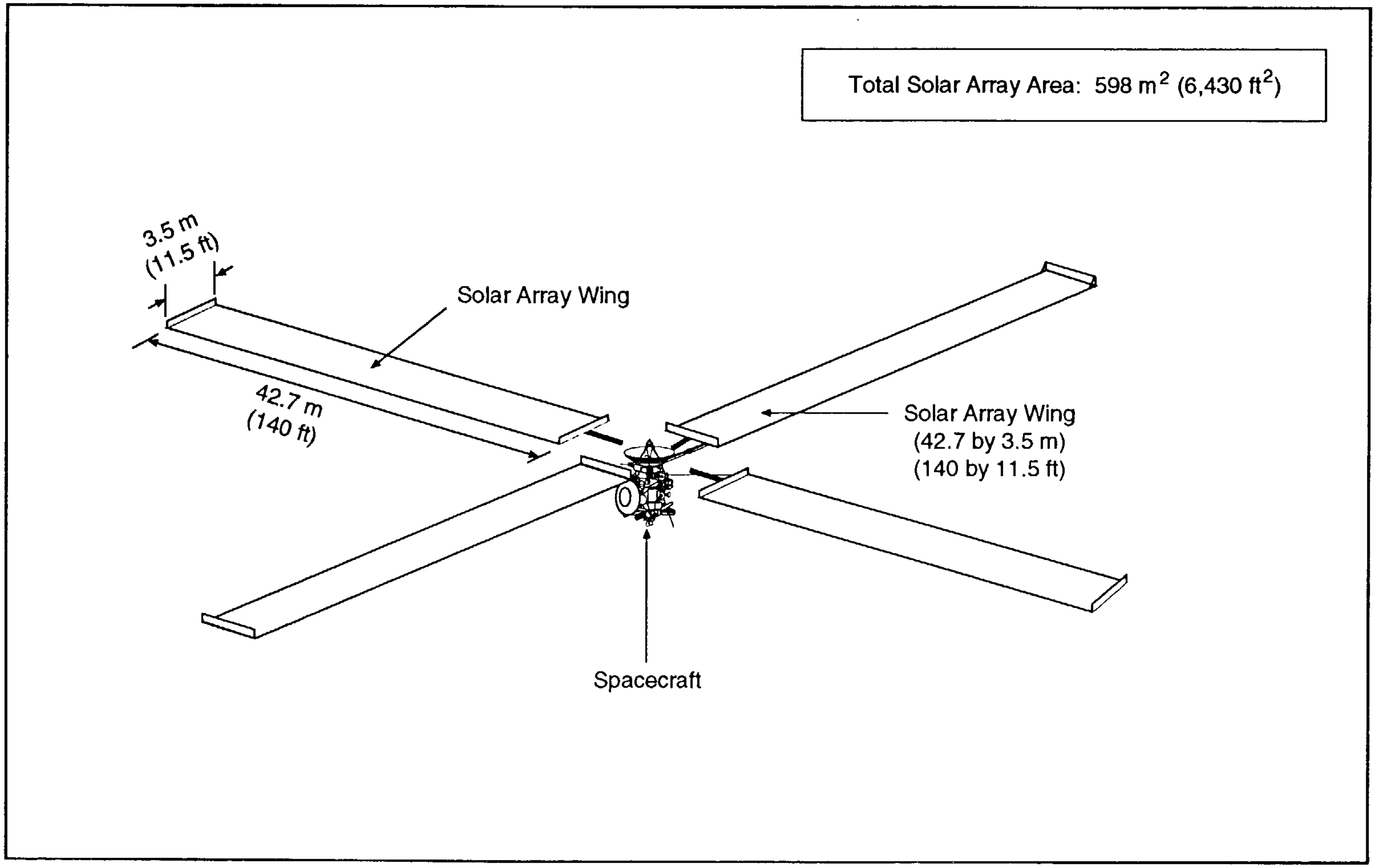
Despite the limiting factors discussed and assuming the use of GaAs APSA technology (for the Cassini mission, GaAs APSA is lighter than Si APSA for the same power output), JPL developed and evaluated several designs of solar-powered spacecraft to ascertain the array compatibility with Cassini's power and operational requirements (JPL 1994a). The designs sought to retain as much science as possible in accordance with the mission's science objectives. In keeping with this goal, two major all-solar designs (and several variants) were evaluated.

In one design, the science instruments were mounted on turntables and scan platforms so that the entire spacecraft would not have to turn to acquire data. The all-solar Cassini design would use four wings, as shown in Figure 2-15, each five times the size of a Hubble Space Telescope wing now operating in low Earth orbit. This design allows the continued acquisition of high-resolution observations during the long (hour to day) maneuvers necessary with large solar arrays. The deployment of such large, massive spacecraft appendages would add considerable risk of failure to the mission compared with using RTGs. In this case, the required solar array size was 598 m^2 (6,430 ft^2). The addition of this size array, in conjunction with the other modifications required to implement solar power, increased the spacecraft dry mass by 1,337 kg (2,948 lb). With the mass of the propellants, the Huygens Probe, and the launch adapter, the total spacecraft mass would increase to 7,228 kg (15,935 lb), far exceeding the launch capacity of the Titan IV (SRMU)/Centaur of 6,234 kg (13,743 lb) for a trajectory to Saturn (JPL 1994a).

Several variations of this design were further investigated in attempts to reduce the mass of the solar arrays. The variants included a combination of RTGs and solar arrays, concentrators to focus sunlight on the arrays, and solar propulsion technologies. All of these designs were very complex and involved increased cost, elevated spacecraft failure risk, and reduced science return. Each of these designs resulted in spacecraft too heavy to be launched in a trajectory to Saturn, given current launch systems.

In the second design, the science instruments were fixed to the body of the spacecraft (as currently designed for use with the RTGs) to reduce the overall mass and the electrical power needed. This design would require that the entire spacecraft be turned to aim the various instruments to acquire data. To further reduce the size of the arrays, the power available to the science instruments was reduced by 50 percent. Because of the large moment of inertia created by the large solar panels (397 m^2 [4,269 ft^2] and 585 kg [1,290 lb]) (JPL 1994a), the time required to turn and maneuver the spacecraft during its exploration of the Saturnian system would increase by a factor of between 4 and 18 compared with the compact RTG-powered spacecraft. The resulting impacts on the mission's science objectives would be serious and include increased times for image mosaics, inadequate turn rates for fields and particles instruments, reduced image resolution due to inadequate target motion compensation, loss of instrument observation time during turns for communicating with Earth, and insufficient turn rates to support radar observation of Titan's cloud-enshrouded surface. These effects on the quality and quantity of the science return raise significant issues about whether the data received would justify the expenditure and risk of this mission design.

This evaluation indicated that the second design would require arrays and other modifications that would increase the spacecraft dry mass by 876 kg (1,931 lb), resulting in a total spacecraft mass (including propellants, the Huygens Probe, and launch adaptor) of 6,293 kg (14,100 lb). This mass would exceed the Titan IV (SRMU)/Centaur launch capability by 59 kg (130 lb). The launch vehicle mass constraints could be satisfied only by disallowing the contingency propellant margins and reserves. Experience, however, has repeatedly demonstrated that the margins and reserves are required for mission success (JPL 1994a). Additionally, there are a number of technical problems associated with this design that would require additional mass to correct, which would further exacerbate the



Source: JPL 1994a

FIGURE 2-15. ALL-SOLAR (GaAs APSA) CONFIGURATION FOR THE CASSINI SPACECRAFT

mass problem. While this design comes close to meeting the launch weight restrictions, the mission would fall short of the minimum science objectives that justify a mission to Saturn.

A wide range of mission opportunities and spacecraft configurations with varying levels of science return would fall between the two major spacecraft designs. The design analyses and evaluations showed that additional science data could only be obtained at the expense of spacecraft electric power subsystem mass.

In summary, an all-solar Cassini spacecraft is considered infeasible at this time because no U.S. launch vehicle exists to launch the mass of even the lightest solar configuration (the GaAs APSA) into the proper trajectory. The large arrays could also generate severe electromagnetic and electrostatic interference, which could adversely affect communications. If a heavy-lift booster and more energetic upper stages were available, severe limitations on spacecraft maneuverability, instrument FOV constraints, and programmatic risks would still make this configuration infeasible (JPL 1994a).

Fuels and Chemicals

Fuels and chemicals are effective means of storing energy that can be converted directly into electricity in a battery, combusted in a fuel cell, or combusted to generate heat to use with a number of energy conversion systems. These types of systems are common on spacecraft. In principle, these power sources would eliminate the concerns regarding a mission accident-related release of radioactive material into the environment. However, the mass requirements of a power system based on fuels and chemicals for an interplanetary mission, such as Cassini, would exceed the launch capacities of a Titan IV (SRMU or SRM)/Centaur. For example, assuming a hydrogen and oxygen power system (with the highest currently available energy-per-unit mass) and a 100 percent efficient conversion system, a mass for the fuel and oxidizer of more than 12,000 kg (26,455 lb) would be required. This is four to six times the allocated dry launch mass of the Cassini spacecraft. No launch vehicle is capable of launching such a large mass. Therefore, power systems based on fuels and chemicals are not feasible for the Cassini mission (JPL 1994a).

Power Beaming (Microwave and Laser) From Earth

Power beaming from Earth consists of generating and transmitting microwaves or laser beams from Earth to a spacecraft, receiving or collecting the energy onboard, and then converting the energy to electricity. The power requirements and the distance from Earth to the spacecraft are primary drivers for this type of system. It is not possible, however, to develop a laser beam power system to satisfy the 1997 Cassini launch date with the current technology and available development time. The ability to deliver a coherent, high-power laser light to such a small target, such as the Cassini spacecraft (out to 9.3 AU), represents a long-term space technology development effort and, therefore, is not available for the Cassini mission.

2.7 COMPARISON OF MISSION ALTERNATIVES, INCLUDING THE PROPOSED ACTION

This section provides a summary comparison of the Proposed Action, including the contingency launch opportunities, and the alternatives. The factors used for this comparison are separated into environmental impacts for both a normal launch and those involving an accident. Table 2-6 compares the environmental impacts from a normal launch for the Proposed Action and the other alternatives. The impacts of the SRM-equipped Titan IV for the Proposed Action are similar to those of the SRMU-equipped Titan IV unless noted. Mission-Specific considerations are summarized for all of the alternatives in Section 2.8.

2.7.1 Environmental Impacts of the Proposed Action

2.7.1.1 Environmental Impacts of Preparing for Launch

Preparing for launch of the Cassini spacecraft involves many activities (e.g., launch vehicle processing; integration of the Huygens Probe, RHUs, and RTGs; and fueling of the spacecraft, the Titan IV launch vehicle, and the Centaur). These preparations would not adversely affect the CCAS/KSC regional area. Section 4.1.1 provides a more detailed discussion of the environmental impacts associated with launch preparations.

2.7.1.2 Environmental Impacts of a Normal Launch

Implementation of the primary launch opportunity in October 1997, secondary launch opportunity in December 1997, or the backup launch opportunity in March 1999 could result in limited short-term air quality, noise, water quality, and biological impacts in the immediate vicinity of the launch site. These impacts have been previously addressed in other NEPA documents (USAF 1986, USAF 1988a, USAF 1988b, USAF 1990, NASA 1994) and would be associated with the routine launch operations of the SRMU or SRM on the Titan IV booster. The potential environmental impacts resulting from a normal launch were determined not to have a substantial adverse effect on CCAS workers or the general population, either individually or cumulatively. Table 2-6 summarizes the environmental impacts of a normal launch, and Section 4.1.2 provides a more detailed discussion.

2.7.1.3 Environmental Impacts of Balance of Mission

The Cassini spacecraft, once injected into its interplanetary VVEJGA trajectory (or a VEEGA for the secondary or backup launch opportunities), would have no adverse impact on the human environment, given a normal trajectory. In addition, the delivery of the Huygens Probe to Titan and the Cassini Saturnian tour would not affect the Earth's environment. The Cassini program complies with NASA policy on planetary protection (JPL 1990).

TABLE 2-6. SUMMARY COMPARISON OF THE IMPACTS OF A NORMAL LAUNCH OF THE CASSINI MISSION

Environmental Impacts	Proposed Action 1997 Primary, Secondary, and Backup	Alternative Actions 1999 Primary/Backup	2001 Primary/Back up	No-Action
Land Use	No substantial adverse impacts on non-launch-related land uses.	Similar to Proposed Action.	Similar to Proposed Action.	No Effect.
Air Quality	High levels of SRMU exhaust products within the exhaust cloud as it leaves the flame trench; cloud would rise and begin to disperse near launch pad; greatest amount of wet HCl deposition within area of about 183 m (600 ft) of launch pad. Exhaust product concentrations expected to drop rapidly with buoyant rise and mixing/dispersal of exhaust cloud; HCl levels expected to decrease to 18.2 mg/m3 within about 5 km (3.1 mi) of launch site. No adverse air quality impacts expected in offsite areas. Not anticipated to adversely affect global climate. Temporary localized decrease in ozone along the flight path, with rapid recovery. Air impacts from SRM-equipped Titan IV similar to Titan IV (SRMU); impacts on troposphere similar; impacts on stratosphere somewhat less than from SRMUs.	Similar to Proposed Action. Greater magnitude of exhaust products due to larger size of the Shuttle solid rocket boosters (SRBs); greater deposition of HCl aerosols and droplets near launch pad due primarily to use of Shuttle main motors at liftoff (main motors emit large quantities of water). Exhaust cloud typically disperses within 14 km (9 mi) of the launch site. Heaviest deposition of HCl droplets and particulates within 1 km (0.6 mi) of launch pad, with substantial short-term air quality degradation within this area. About twice as much solid rocket exhaust products emitted to stratosphere compared to SRM- or SRMU-equipped Titan IV. Short-term decrease in stratospheric ozone levels along flight path, with rapid recovery. Impacts would occur twice during a period of 21 to 51 days.	Similar to Proposed Action.	No Effect.
Noise and Sonic Boom	No sustained adverse impacts on work force or unprotected public.	Similar to Proposed Action. Twice the number of sonic booms occur over ocean; no adverse impacts expected.	Similar to Proposed Action.	No Effect.
Geology and Soils	Short-term acidification of the soils adjacent to launch site. Increased surface concentration of aluminum in the soils. SRM-equipped Titan IV would have similar impacts to Titan IV (SRMU).	Similar to Proposed Action. Greatest impacts in a 1 to 15 ha (2.5 to 37 acre) area north of launch pad. Impacts would occur twice during a period of 21 to 51 days.	Similar to Proposed Action.	No Effect.
Hydrology and Water Quality	No substantial adverse long-term impacts. Short-term increase in the acidity of nearby waters. SRM-equipped Titan IV would have similar but slightly lower magnitude impacts compared to Titan IV (SRMU).	Similar to Proposed Action. Impacts would occur twice during a period of 21 to 51 days.	Similar to Proposed Action.	No Effect.

TABLE 2-6. SUMMARY COMPARISON OF THE IMPACTS OF A NORMAL LAUNCH OF THE CASSINI MISSION (Continued)

	Proposed Action	Alternative Actions		
	1997	1999	2001	No-Action
Environmental Impacts	Primary, Secondary, and Backup	Primary/Backup	Primary/Backup	
Biological Resources	No impact to floodplain. Some acidification of wetlands adjacent to launch site. High-risk zone for wildlife within about 183 m (600 ft) of launch pad; vegetation damage and wildlife mortality in a 20-m (66-ft) area near the flame trench exit. If exhaust cloud is pushed back over land, short-term acidification of nearby surface waters could cause mortality of aquatic biota. No long-term adverse effects expected. No substantial short-term or long-term impact to threatened or endangered species. SRM-equipped Titan IV would have similar but slightly lower magnitude impacts compared to Titan IV (SRMU).	Similar to Proposed Action. In addition, fish kills in nearby impoundments possible with each Shuttle launch due to acidification from exhaust cloud. Impacts would occur twice during a period of 21 to 51 days.	Similar to Proposed Action.	No Effect.
Socioeconomics	Potential short-term economic benefits from tourism.	Some economic benefits associated with tourists coming in region to observe Shuttle launch. Could get influx twice, about 21 to 51 days apart.	Similar to Proposed Action.	No Effect.
Historical/Archaeological Resources	No impact expected.	Similar to Proposed Action.	Similar to Proposed Action.	No Effect.
Radiation Exposures	Occupational doses from handling of RTGs and RHUs will be within regulatory requirements and strictly monitored. No substantial exposure to other workers or the general public.	Similar to Proposed Action.	Similar to Proposed Action.	No Effect.
Balance of Mission	No impact	Similar to Proposed Action.	Similar to Proposed Action.	No Effect.

2.7.1.4 Environmental Impacts and Consequences of Mission Accidents

Possible Nonradiological Impacts of a Mission Accident

The nonradiological impacts of the possible Titan IV accidents were addressed in the Titan IV Environmental Assessments (USAF 1986, USAF 1988a, USAF 1990) and are fundamentally similar to the Spac Shuttle nonradiological accident impacts addressed in the Shuttle Program EIS (NASA 1978), the Tier 1 Galileo and Ulysses missions EIS (NASA 1988b), and the Tier 2 EISs for Galileo (NASA 1989b) and Ulysses (NASA 1990) missions. These accidents include on-pad propellant spills, fires, and explosions. In addition, some accident scenarios could result in the inadvertent reentry of the Cassini spacecraft into the Earth's atmosphere from parking orbit or during an Earth swingby. Should either of these reentry scenarios occur, it is expected that the spacecraft would break up and the remaining liquid propellants onboard would burn and/or disperse in the atmosphere. The propellants would not reach the Earth in concentrations sufficient to affect ambient air quality. Section 4.1.4 provides a more detailed discussion of the potential nonradiological impacts of a mission accident.

Possible Radiological Impacts and Consequences of Mission Accidents

DOE has conducted safety testing and analyses to determine the response of the RTGs and RHUs to postulated accidents, and the consequences of accidents. DOE has designed the GPHS-RTG assembly to ensure that the fuel is contained to the maximum extent practical. RTG and RHU responses to a broad range of accident conditions and estimates of the radiological consequences, if any, of an accident can be predicted. The results of the current accident analyses (Martin Marietta Astro Space 1993, Halliburton NUS 1994a) used to support evaluations for this EIS are presented in Section 4.1.5. DOE will perform additional safety analyses for the Cassini mission and document the results in Final Safety Analysis Reports.

For the Proposed Action (October 1997 primary, December 1997 secondary and the March 1999 backup launch opportunities), analyses indicate that while the consequences of a release could range from small to substantial, the probability of an accident occurring that could release plutonium dioxide fuel is extremely small (see Section 4.1.5.3 of this EIS for more detail).

The regional and global areas of the environment, defined in Section 3, could be affected by a release of plutonium dioxide fuel. The regional area, including the six-county region surrounding CCAS and KSC, could be impacted by a Phase 1 accident. The global area (areas elsewhere around the world) could be potentially impacted by accidents identified in Phases 5, 6, or by an inadvertent reentry during an Earth swingby.

Although most potential accidents would result in the loss of the launch vehicle and/or Cassini spacecraft, most accidents would not result in a release of the plutonium dioxide fuel to the environment (Martin Marietta 1992, Martin Marietta Astro Space 1993). However, four specific accident scenarios were identified as representative of the categories of failures that could cause a release of plutonium dioxide fuel from the GPHS modules at ground level within the post-accident plume during mission launch Phases 1 through 6: 1) Command Shutdown and Destruct, 2) Titan IV (SRMU) Fail-to-Ignite, 3) Centaur Tank Failure/Collapse, and 4) Inadvertent Reentry From Earth Orbit (Martin Marietta Astro Space 1993). Accident scenarios leading to launch vehicle propellant explosions could subject the RTGs to accident environments, such as blast overpressures, solid or liquid propellant fires, and high-velocity fragments. In addition, two postulated accident scenarios associated with the interplanetary cruise portion of the VVEJGA and VEEGA trajectories have been defined. The short-term inadvertent reentry scenario involves the reentry of the spacecraft into the Earth's atmosphere during a planned Earth swingby. The long-term inadvertent reentry scenario involves a spacecraft failure that leaves it drifting in an Earth-crossing orbit around the Sun and potentially reentering the Earth's atmosphere a decade to millennia later. NASA is designing the Cassini mission and spacecraft to ensure, to the maximum extent practical, that an inadvertent reentry accident does not occur.

The potential for radiological impact to the affected area depends on the mission phase/scenario combination, the probability of the accident occurring with a fuel release, the amount of fuel released (i.e., sourc term), and the radiological consequences of the release. Radiological consequences can be expressed as the collective dose, health effects (i.e., excess latent cancer fatalities), individual dose, or land contamination. Healtl effects can be considered with or without de minimis. The concept of de minimis assumes that doses of less thar 1×10^{-5} Sv/yr (1×10^{-3} rem/yr) do not result in any health effects and, therefore, are not included in the health effects calculations.

Although radiological consequences can be used to predict doses to an individual or exposed population, risk is another useful assessment A health effects risk assessment was performed specifically for the Cassini mission to quantify the consequences that could result from a radiological accident (Halliburton NUS 1994a, Halliburton NUS 1994b). Risk is presented in Tables 2-7 and 2-8 as average individual risk and in Table 2-9 as health effects mission risk. The average individual risk due to a given accident scenario represents the probabilit that any given individual within the exposed population group would develop a latent cancer fatality as a result o that accident scenario. The health effects mission risk contribution of a given accident scenario is defined as a probability-weighted health effect as a direct result of that accident scenario. Table 2-7 identifies the impacts from the accident scenario that dominates the average individual risk of acquiring a heath effect within the regional area, assuming that a radiological accident has occurred during a launch phase. Table 2-8 summarizes the radiological impacts for an inadvertent reentry during an Earth swingby.

Table 2-9 compares the health effects mission risk as a result of implementation of each of the alternatives. For clarity, the total heath effects mission risk is separated into the risks associated with the launch, the Earth swingby(s) during the interplanetary trajectory, and the combined risks of the launch phases and swingby portior of the mission. Section 4.1.8 presents a detailed risk assessment. The health and environmental risks associated with plutonium (mainly Pu-238) dioxide are addressed in the Galileo and Ulysses EISs (NASA 1989b, NASA 1990) and in Appendix C of this EIS.

2.7.2 Environmental Impacts of the 1999 Mission Alternative

A Shuttle launch generally results in limited short-term air, water, and biological impacts in the immediate vicinity of the launch site. These impacts have been addressed in detail in other NEPA documents (NASA 1978, NASA 1986, NASA 1989b, NASA 1990)

TABLE 2-7. SUMMARY COMPARISON OF POTENTIAL RADIOLOGICAL IMPACTS ASSOCIATED WITH LAUNCH PHASE ACCIDENTS FOR ALL ALTERNATIVES[a,b]

Potential Radiological Impacts Associated with Launch Phase Accidents	Proposed Action 1997 Primary/Secondary/Backup[c]	Alternative Actions 1999 Primary/Backup[d]	Alternative Actions 2001 Primary/Backup	Alternative Actions No-Action
CCAS/KSC Regional Area-Near Pad				
Source Term, Bq (Ci)	1.38×10^{11} (3.72×10^{0})	2.38×10^{12} (6.44×10^{1})	Same as Proposed Action	No Effect
Total Probability (likelihood)	9.1×10^{-4} (1 in 1,100)	1.0×10^{-5} (1 in 100,000)	Same as Proposed Action	No Effect
Health Effects[e]	3.36×10^{-4}	4.86×10^{-3}	Same as Proposed Action	No Effect
Land Contamination, km^2 (mi^2)[f]	1.86×10^{-1} (7.18×10^{-2})	3.84×10^{0} ($1.4 8 \times 10^{0}$)	Same as Proposed Action	No Effect
Average Individual Risk[e,g] (likelihood)	3.1×10^{-12}(11 in 323 billion)	5.0×10^{-13} (1 in 2 trillion)	Same as Proposed Action	No Effect
Global Area-Africa				
Source Term, Bq (Ci)	5.44×10^{10} (1.47×10^{0})	5.07×10^{10} (1.37×10^{0})	Same as Proposed Action	No Effect
Total Probability (likelihood)	4.6×10^{-4} (1 in 2,170)	5.8×10^{-5} (1 in 17,200)	Same as Proposed Action	No Effect
Health Effects[e]	1.51×10^{-4}	1.41×10^{-4}	Same as Proposed Action	No Effect
Land Contamination, km^2 (mi^2)[f]	2.17×10^{-2} (8.38 x 10-3)	2.02×10^{-2} (7.80×10^{-3})	Same as Proposed Action	No Effect
Average Individual Risk[e,g] (likelihood)	6.9×10^{-11} (1 in 14.5 billion)	8.2×10^{-12} (1 in 121 billion)	Same as Proposed Action	No Effect

Sources: Halliburton NUS 1994a, Halliburton NUS 1994b

a. The detailed consequence analyses are presented in Sections 4.1.6 and 4.2.6.
b. Radiological consequences are identified for the accident scenario that dominates the average individual risk of acquiring a health effect within the affected area.
c. If the Titan IV (SRM)/Centaur configuration is used, no substantial differences would be expected in the radiological consequences.
d. The second launch of each opportunity would have identical radiological consequences. Only the second launch carries the radioactive payload.
e. Estimated without de minimis. The de minimis dose level for the purpose of this EIS is 1.0×10^{-5} Sv (1.0×10^{-3} rem) per year. Health effects are estimated for the population near the launch area or impact area.
f. Land contamination is the estimated land area contaminated above the EPA screening level of 7.4×10^{3} Bq/m^2 ($0.2 \mu Ci/m^2$). See Sections 4.1.6.2 and 4.2.6.1.
g. The average individual risk for a given accident scenario is: (Total Probability of the Scenario x Health Effects for the Scenario)/(Exposed Population at Risk). See Sections 4.1.8 and 4.2.8.

TABLE 2-7. SUMMARY COMPARISON OF POTENTIAL RADIOLOGICAL IMPACTS ASSOCIATED WITH LAUNCH PHASE ACCIDENTS FOR ALL ALTERNATIVES[a,b] (Continued)

Potential Radiological Impacts Associated with Launch Phase Accidents	Proposed Action	Alternative Actions		
	1997	1999	2001	
	Primary/Secondary/Backup[c]	Primary/Backup[d]	Primary/Backup	No-Action
Global Area-Unspecified Locations				
Source Term, Bq (Ci)	5.55×10^{10} (1.50×10^{0})	5.5×10^{10} (1.50×10^{0})	Same as Proposed Action	No Effect
Total Probability (likelihood)	4.4×10^{-4} (1 in 2,300)	1.9×10^{-3} (1 in 526)	Same as Proposed Action	No Effect
Health Effects[e]	6.90×10^{-4}	6.89×10^{-4}	Same as Proposed Action	No Effect
Land Contamination, km^2 (mi^2)[f]	2.22×10^{-2} (8.57×10^{-3})	2.22×10^{-2} (8.57×10^{-3})	Same as Proposed Action	No Effect
Average Individual Risk[e,g] (likelihood)	6.0×10^{-11} (1 in 16 billion)	2.6×10^{-10} (1 in 3.8 billion)	Same as Proposed Action	No Effect

Sources: Halliburton NUS 1994a, Halliburton NUS 1994b

a. The detailed consequence analyses are presented in Sections 4.1.6 and 4.2.6.
b. Radiological consequences are identified for the accident scenario that dominates the average individual risk of acquiring a health effect within the affected area.
c. If the Titan IV (SRM)/Centaur configuration is used, no substantial differences would be expected in the radiological consequences.
d. The second launch of each opportunity would have identical radiological consequences. Only the second launch carries the radioactive payload.
e. Estimated without de minimis. The de minimis dose level for the purpose of this EIS is 1.0×10^{-5} Sv (1.0×10^{-3} rem) per year. Health effects are estimated for the population near the launch area or impact area.
f. Land contamination is the estimated land area contaminated above the EPA screening level of 7.4×10^{3} Bq/m^2 (0.2 $\mu Ci/m^2$). See Sections 4.1.6.2 and 4.2.6.1 .
g. The average individual risk for a given accident scenario is: (Total Probability of the Scenario x Health Effects for the Scenario)/(Exposed Population at Risk). See Sections 4.1.8 and 4.2.8.

TABLE 2-8. SUMMARY COMPARISON OF THE POTENTIAL RADIOLOGICAL IMPACTS ASSOCIATED WITH THE SHORT-TERM INADVERTENT REENTRY ACCIDENT SCENARIO FOR ALL ALTERNATIVES[a,b,c]

	Proposed Action			Alternative Actions		
	1997			1999	2001	
Potential Radiological Impacts	Primary	Secondary/Backup[d]		Primary/Backup	Primary/Backup	No-Action
Reentry Case	VVEJGA	VEEGA El	VEEGA E2	VEEGA	VVVGA/VEEGA	No Effect
Source Term, Bq (Ci)	5.40×10^{15} (1.46×10^{5})	5.44×10^{15} (1.47×10^{5})	5.29 x 1015 (1.43×10^{5})	Same as secondary and backup of Proposed Action	Not applicable/same as secondary and backup of Proposed Action	No Effect
Total Probability (likelihood)	7.6×10^{-7} (1 in 1.3 million)	1.9×10^{-7} (1 in 5.3 million)	2.8×10^{-7} (1 in 3.6 million)	Same as secondary and backup of Proposed Action	Not applicable/same as secondary and backup of Proposed Action	No Effect
Health Effects without de minimis[e]	2.30×10^{3}	2.48×10^{3}	4.56×10^{3}	Same as secondary and backup of Proposed Action	Not applicable/same as secondary and backup of Proposed Action	No Effect
Land Contamination, km^2 (mi^2)[f]	2.04×10^{3} (7.88×10^{2})	2.00×10^{3} (7.72×10^{2})	4.13×10^{3} (1.59×10^{3})	Same as secondary and backup of Proposed Action	Not applicable/same as secondary and backup of Proposed Action	No Effect
Average Individual Risk without de minimis[e,g] (likelihood)	3.4×10^{-13} (1 in 2.9 trillion)	9.4×10^{-14} (1 in 11 trillion)	2.6×10^{-13} (1 in 3.8 trillion)	Same as secondary and backup of Proposed Action	Not applicable/same as secondary and backup of Proposed Action	No Effect

Source: Halliburton NUS 1994a, Halliburton NUS 1994b, JPL 1993f

a. The detailed consequence analyses are presented in Sections 4.1.6 and 4.2.6.
b. Radiological consequences and risk for the long-term inadvertent reentry case for all alternatives cannot be estimated and, therefore, are not included.
c. Expectation case or "expected" source terms are used.
d. The impacts were estimated for the backup launch opportunity only. The impacts of the secondary launch opportunity are assumed to be similar.
e. Health effects, or excess latent cancer fatalities, for the short-term inadvertent reentry accident are evaluated based on collective exposure of approximately 5 billion persons worldwide. Most of the persons exposed would receive an individual radiation dose of less than 1.0×10^{-5} Sv (1.0×10^{-3} rem) per year (the de minimis dose level). If only those individuals worldwide receiving higher than de minimis dose level are considered, the estimated health effects would be approximately 10 (excess latent cancer fatalities) with the VVEJGA, and about 15 with either the VEEGA E1 or E2.
f. Land contamination is the estimated land area contaminated above the EPA screening level of 7.4×10^{3} Bq/m^2 (0.2 $\mu Ci/m^2$). See Sections 4.1.6.2 and 4.2.6.1 .
g. The average individual risk for a given accident scenario is: (Total Probability of the Scenario x Health Effects for the Scenario)/(Exposed Population at Risk). See Sections 4.1 .8 and 4.2.8.

TABLE 2-9. SUMMARY COMPARISON OF THE HEALTH EFFECTS MISSION RISK FOR ALL CASSINI MISSION ALTERNATIVES

	Health Effects Mission Risk					
	Proposed Action[a]		Alternative Actions			
	1997		1999	2001		
Mission Phase	Primary	Secondary/Backup	Primary/Backup	Primary	Backup	No-Action
• Launch Phases	8.4×10^{-7}	8.4×10^{-7}	2.1×10^{-6}	8.4×10^{-7}	8.4×10^{-7}	No Effect
• Short-Term Inadvertent Reentry	1.7×10^{-3}	1.8×10^{-3}	1.8×10^{-3}	No Short-Term Reentry	1.8×10^{-3}	No Effect
• Long-Term Inadvertent Reentry	b	b	b	b	b	b
• Total Mission(without long-term inadvertent reentry health effects mission risks)	1.7×10^{-3}	1.8×10^{-3}	1.8×10^{-3}	c	1.8×10^{-3}	No Effect

Sources: Halliburton NUS 1994a, Halliburton NUS 1994b

a. If the Titan IV (SRM)/Centaur configuration is used, no substantial differences would be expected in the mission risk.

b. It should be noted that all alternatives have the potential for interplanetary trajectory accidents that could result in a long-term inadvertent Earth reentry. The mean probability of a long-term reentry for each alternative has been estimated as very small (6.0×10^{-8} over the first 100-years for the 1997 primary VVEJGA mission; 4×10^{-7} for the longer backup 1997 and the 1999 VEEGA missions; and on the order of 10^{-7} for the 2001 VVVGA alternative). Since the timing of a long-term reentry cannot be predicted, it is not possible to estimate the world population or the extent of radioactive decay of the fuel for such a reentry, thus it would not be possible to estimate the resulting consequences (i.e., health effects and land contamination) and risks. It would be reasonable to assume, ignoring the uncertainties noted above, that the radiological releases, and in turn the consequences and risks would be on the same order of magnitude as those for a short-term reentry for an Earth-gravity-assist trajectory.

c. Following the assumptions in "b" above, and assuming the risk associated with long-term inadvertent reentry would be on the same order of magnitude as that for a short-term reentry, the long-term reentry then becomes the principal determinant of total mission risk for the 2001 VVVGA alternative. Thus, the total mission risk for this alternative is assumed to be on the same order of magnitude as noted for the other alternatives.

and are associated with the routine launch operations of the Shuttle. Since this mission alternative would involve two Shuttle launches during a period of 21 and 51 days, the associated launch impacts would occur twice.

2.7.2.1 Environmental Impacts of Preparing for Launch

Preparing for a Shuttle launch of the Cassini spacecraft would involve several activities, including the integration of the Huygens Probe, RTGs, and RHUs and fueling of the spacecraft, the Shuttle, and its external tank. These preparations would not adversely affect the CCAS/KSC regional area.

2.7.2.2 Environmental Impacts of a Normal Launch

The environmental impacts associated with a normal launch of the Shuttle, addressed in detail in several NEPA documents (NASA 1978, NASA 1986, NASA 1989b, NASA 1990) and the *KSC Environment Resources Document* (NASA 1994) are summarized in Table 2-6, given previously.

2.7.2.3 Environmental Impacts of Balance of Mission

The environmental impacts of balance of mission for the 1999 mission alternative would be similar to those of the Proposed Action. See Section 2.7.1.3.

2.7.2.4 Environmental Impacts and Consequences of Mission Accidents

Possible Nonradiological Impacts of a Mission Accident

The nonradiological impacts of the possible Space Shuttle accidents were addressed in the Shuttle Program EIS (NASA 1978), the Tier 1 Galileo and Ulysses missions EIS (NASA 1988b), and the Tier 2 EISs for Galileo (NASA 1989b) and Ulysses (NASA 1990) missions. These accidents include on-pad propellant spills, fires, and explosions. In addition, some accident scenarios could result in the inadvertent reentry of the Cassini spacecraft into the Earth's atmosphere from parking orbit or during an Earth swingby. Should either of these reentry scenarios occur, it is expected that the spacecraft would break up and the remaining liquid propellants onboard would burn and/or disperse in the atmosphere. The propellants would not reach the Earth in concentrations sufficient to affect ambient air quality. Section 4.2.4 provides a more detailed discussion of the potential nonradiological impacts to the environment from a mission accident.

Possible Radiological Impacts and Consequences of Mission Accidents

The results of the current accident analyses (Halliburton NUS 1994b, Martin Marietta Astro Space 1994b) used to support evaluations for this EIS are presented in Section 4.2.5. DOE will perform additional analyses for the Cassini mission, and will document the results in Final Safety Analysis Reports.

For the 1999 mission alternative, while the consequences of a release could range from small to substantial, analyses indicate that the likelihood of an accident occurring that could release plutonium dioxide fuel is extremely small. In the event of an accident, the regional and global areas of the environment, defined in Section 3, could be affected by a release of plutonium dioxide fuel. The regional area, including the six-county region surrounding CCAS/KSC, could be impacted by a Phase 1 accident. Indeterminate locations within the global area could be potentially impacted by accidents in the higher altitude portions of Phase 1 and Phases 2 through 4 as well as by an accident occurring as a result of an inadvertent reentry.

Although most potential accidents would result in the loss of the launch vehicle and/or Cassini spacecraft, most accidents would not result in a release of the plutonium dioxide fuel to the environment. However, several specific accident scenarios were identified from these previous analyses that could cause a release of plutonium dioxide fuel from the GPHS modules during Shuttle launch Phases 1 through 4. In addition, the two inadvertent reentry scenarios identified for the Proposed Action would also be associated with the 1999 mission alternative.

Using the same methodology as for the Titan IV (SRMU), the potential extent of radiological impact to the affected area was estimated for the 1999 mission alternative. Details of the radiological impacts are given in Section 4.2.5. Table 2-7, given previously, identifies the impacts that would dominate the average individual risk of acquiring a health effect within the affected area, assuming that a radiological accident had occurred during a launch phase. Table 2-8, also given previously, summarizes the radiological impacts for the short-term inadvertent reentry. As with the Proposed Action, a risk assessment was specifically conducted for the 1999 mission alternative to quantify the degree of consequence that could result from a radiological accident (Halliburton NUS 1994b, Martin Marietta Astro Space 1994b) Table 2-9, given previously, presents the health effects mission risk.

2.7.3 Environmental Impacts of the 2001 Mission Alternative

The impacts from preparation for and implementation of a normal launch of the Cassini spacecraft during the 2001 mission alternative on a VVVGA trajectory would be similar to those of the Proposed Action using the Titan IV (SRMU)/Centaur. The SRM-equipped Titan IV does not have the capability to insert the spacecraft into a VVVGA trajectory. The 2001 primary launch opportunity would essentially use identical components (i.e., Titan IV (SRMU)/Centaur and electrical power source) as those used for the Proposed Action, described in Section 4.1. The spacecraft would use larger propellant tanks, however, to accommodate the additional propellant required to complete the mission using the VVVGA trajectory and a specially designed rhenium engine for spacecraft propulsion. This alternative would require the spacecraft to execute three gravity-assist swingbys of the planet Venus.

Launch accident scenarios, environments, and radiological consequences for the primary opportunity would be similar to those for Phases 1 through 6 of the Proposed Action (see Sections 4.1.5 through 4.1.9). The accident environments created by a potential explosion of the additional propellant would be no more severe than those estimated for the representative accident scenarios discussed previously (JPL 1993b). In addition, the mission trajectory would not use the Earth for a gravity-assist, thereby eliminating the potential for an inadvertent short-term reentry. However, should the spacecraft become uncommandable anytime after injection into its interplanetary trajectory and before the SOI, the probability of a long-term reentry would exist. The long-term reentry conditions would be assumed to be similar to the short-term inadvertent reentry conditions described for the Proposed Action.

Because there is no non-Earth-Gravity-Assist backup launch opportunity for the 2001 non-EGA trajectory (i.e., VVVGA), the backup opportunity would employ a VEEGA trajectory. Tables 2-7 through 2-9, previously given, present the potential radiological impacts associated with launch phase accidents, potential radiological impacts associated with the short-term inadvertent reentry accident scenario (for the backup opportunity), and heath effects mission risk, respectively.

As with all launch opportunities, a long-term inadvertent reentry could also result in heath effects risks. These risks are not known but the probability is expected to be very low and the risks are expected to be similar to those for the short-term inadvertent reentry for the Proposed Action.

2.7.4 Environmental Impacts of the No-Action Alternative

The No-Action alternative would not cause any adverse health or environmental impacts.

2.7.5 Scope and Timing of Mission Science Return

The Proposed Action would accomplish NASA's scientific objectives for the Cassini mission's study of Saturn, its atmosphere, moons, rings, and magnetosphere. Launch of Cassini in October 1997 (the Proposed Action primary launch opportunity) would result in the earliest collection of these scientific data at a most optimum time (in 2004) because the spacecraft would arrive at Saturn when the rings would have a scientifically favorable tilt toward the Sun and the Earth. The secondary launch opportunity would afford a similar science profile but would be limited by poorer ring geometry. The science return would be delayed 2 years compared with the primary launch. The 1999 backup launch opportunity would accomplish essentially the same science objectives, with some reduction of ring science. The backup launch opportunity would delay the science return by 4 years.

The 1999 mission alternative using the dual Shuttle launches would be able to obtain similar levels of science objectives and science return as either the secondary or backup launch opportunities of the Proposed Action.

The 2001 mission alternative would result in a later arrival date at Saturn, when Saturn's rings would be seen nearly edge-on from Earth and with lower solar illumination, thereby limiting the use of radio and optical science experiments during the Saturnian tour. The number of Titan flybys would have to be reduced significantly from 35 to 21, and the SOI burn delay would have to be eliminated, substantially decreasing the close-in ring science. Therefore, the overall science return would be reduced from the return of the Proposed Action. In addition, the spacecraft would be older at the onset of the science phase, increasing the probability of spacecraft failure due to aged components. Limitations with propellant and electrical power margins would reduce the science return associated with this alternative compared with the Proposed Action.

The No-Action alternative would not yield any of the anticipated science data on Saturn and its environment, thereby effectively preventing NASA, ESA, and ASI from achieving their solar system exploration objectives. Although new technological advances (e.g., solid-state recorded, an innovative solid-state power switch, and gyros) have been made during the development of Cassini the scientific investigations of the American and international scientists who have contributed to the development of the Cassini spacecraft and its experiments would be terminated. In addition, this alternative would terminate the international agreements to develop Cassini, disrupt and strain the relationships for other space-related projects, and hinder the future formation of other international science and engineering teams. Cassini's U.S.- European partnership is an example of an undertaking whose scope and cost would not likely be borne by any single nation, but is made possible through the shared investment and participation. Failure to undertake the mission would discourage other similar international partnerships for larg peaceful efforts.

2.7.6 Launch Schedules and Availability of Launch Vehicle

Consistent with planning for the Proposed Action, the Cassini mission would be scheduled for flight to Saturn and its environs using the Titan IV (SRMU or SRM)/Centaur in October 1997 from Launch Complex 40 or 41 at CCAS. If NASA could not launch Cassini in October of 1997, the contingency launch opportunities (secondary in December 1997 and the backup in March 1999) would then be considered. Depending on the nature of the delay, launch facility schedules, mission budgets, and the cooperation of the foreign partners, the Cassini mission would then be rescheduled and launched on either the secondary or backup opportunities on the Titan IV (SRMU or SRM)/Centaur. Similarly, schedules would be developed, if necessary, for the 1999 and 2001 mission alternatives.

2.7.7 Availability of Facility and Personnel

To implement the Proposed Action, NASA anticipates that all required NASA, ESA, and ASI scientific anc engineering facilities and personnel (including contractors and subcontractors) would be available to support the mission's launch in October 1997 from CCAS. NASA's Deep Space Network is preparing to meet the tracking and data relay requirements of the mission.

The 1999 mission alternative on the Shuttle would require retaining the program personnel and facilities fo approximately 2 years, as well as securing new personnel and launch services for the Shuttle. Moreover, the delay of this mission could disrupt and possibly strain the international partnerships formed to develop the Cassi Orbiter, the Huygens Probe, and other space-related projects.

A launch of the 2001 mission alternative would require retaining the program personnel and facilities for approximately 4 years. Some of the mission's expert personnel could be lost during this period. Moreover, the delay of this mission would disrupt and possibly strain the international partnerships.

The selection of the No-Action alternative would terminate the existing U.S. and foreign engineering and scientific services, and important expertise could be irretrievably lost.

2.8 SUMMARY

This section summarizes the mission-specific considerations for the Proposed Action and alternatives. Table 2-10 provides a summary comparison of these considerations.

TABLE 2-10. SUMMARY COMPARISON OF THE MISSION-SPECIFIC CONSIDERATIONS FOR THE CASSINI MISSION ALTERNATIVES

Considerations for Comparison	Proposed Action		
	Primary[a]	Secondary[a]	Backup[a]
Launch Vehicle	Titan IV (SRMU or SRM)/Centaur	Titan IV (SRMU or SRM)/Centaur	Titan IV (SRMU or SRM)/Centaur
Trajectory	VVEJGA	VEEGA	VEEGA
Launch Opportunity	October 1997	December 1997	March 1999
Cruise Time to Saturn	6.7 Years	8.8 Years	9.8 Years
Mission Margins			
• Power	Adequate	Slightly Reduced	Slightly Reduced
• Propellant	Adequate	Adequate	Adequate
Timing of Science Return			
• Saturn Arrival Date	June 2004	October 2006	December 2008
• End of Saturnian Tour	2008	2010	2012
Completion of Science Return	Full Return	Reduced Science Return	Reduced Science Return
Continuity of Support			
• Facilities Available	Firm Commitment	Firm Commitment	Firm Commitment
• Mission Personnel	Firm Commitment	Firm Commitment	Firm Commitment
International Agreements	Cooperative	Cooperative	Possibly Disrupted

Source: JPL 1994a

a. Should the Titan IV (SRM) be needed, differences exist in cruise time and Saturn arrival date. The science return would be less than the Titan IV (SRMU)/Centaur primary launch opportunity.

TABLE 2-10. SUMMARY COMPARISON OF THE MISSION-SPECIFIC CONSIDERATIONS FOR THE CASSINI MISSION ALTERNATIVES (Continued)

Considerations for Comparison	Alternative Actions				
	1999 Dual Shuttle		2001 Non-EGA		
	Primary	Backup	Primary	Backup	No-Action
Launch Vehicle	STS/Upper Stage(s)	STS/Upper Stage(s)	Titan IV (SRMU)/Centaur	Titan IV (SRMU)/Centaur	Not Applicable
Trajectory	VEEGA	VEEGA	VVVGA	VEEGA	Not Applicable
Launch Opportunity	March 1999	August 2000	March 2001	May 2002	Not Applicable
Cruise Time to Saturn	9.8 Years	9.4 Years	10.3 Years	11.4 Years	Not Applicable
Mission Margins • Power • Propellant	 Slightly Reduced Adequate	 Slightly Reduced Adequate	 Slightly Reduced Marginal	 Slightly Reduced Marginal	 Not Applicable
Timing of Science Return • Saturn Arrival Date • End of Saturnian Tour	 December 2008 2012	 January 2010 2014	 June 2011 2015	 September 2013 2017	 Not Applicable
Completion of Science Return	Reduced Science Return	Reduced Science Return	Reduced Science Return	Reduced Science Return	No Science Return
Continuity of Support • Facilities Available • Mission Personnel	 Probably Available Probably Available	 Probably Available Probably Available	 Probably Available Probable Loss of Mission Personnel	 Probably Available Probable Loss of Mission Personnel	 Not Applicable
International Agreements	Possibly Disrupt	Possibly Disrupt	Possibly Disrupt	Possibly Disrupt	Disrupt

Source: JPL 1994a

Launching the Cassini spacecraft and delivery of the Huygens Probe during the Proposed Action's October 1997 primary launch opportunity or during one of the contingency opportunities would enable scientists to acquire extensive new information about the planet Saturn, its atmosphere, moons, rings and magnetosphere. This 4-year science tour of Saturn and its environs would be an opportunity to gain significant insights, both planned and unplanned, into major scientific questions about the creation of the solar system and the conditions that led to life on Earth, in addition to a host of questions specific to the Saturnian system.

Among the scientific goals, the Huygens Probe would collect data on Saturn's largest moon, Titan. The Probe, developed by ESA specifically for the Cassini mission, would descend by parachute through Titan's atmosphere. The instruments mounted on the Probe would directly sample the atmosphere to determine its composition. Once on the surface, the Probe would gather data on the surface composition and landscape and transmit the information to the Cassini Orbiter.

Cassini would also study Saturn's rings, continuing the science efforts begun by the Voyager mission. Long-term closeup observations of the rings by Cassini could help resolve unexplained phenomena, such as the various wave patterns, small and large gaps clumping of material, and small "moonlets" embedded in the rings.

The Proposed Action represents the combination of spacecraft power system, launch vehicle configuration, and trajectory that would best satisfy all of the mission science objectives. The spacecraft's electrical power would be provided by three RTGs. For the primary opportunity, a Titan IV (SRMU or SRM)/Centaur would launch the Cassini spacecraft from CCAS into a VVEJGA trajectory to arrive at Saturn in 2004. If NASA could not launch Cassini during the primary opportunity, contingency launch opportunities in either December 1997 or March 1999 have been identified. The December 1997 secondary launch opportunity would place Cassini on an 8.8-year VEEGA trajectory, arriving at Saturn in 2006; the March 1999 backup launch opportunity would place Cassini on a 9.8-year VEEGA trajectory, arriving at Saturn in 2008. In the event that the Titan IV (SRMU)/Centaur were not available, a Titan IV (SRM)/Centaur would be used. The primary, secondary and backup opportunities would remain the same.

The Titan IV (SRMU)/Centaur and the Titan IV (SRM)/Centaur are the only feasible U.S. launch vehicle configurations to meet the October 1997 launch opportunity. The RTG technology is the only power system currently available that meets all the requirements for the mission. Although the potential use of solar power was evaluated, it was not considered feasible at this time because of the large mass that would be required for the solar array. The large mass and dimensions of the array combined with the mass of the Cassini spacecraft would exceed the launch capabilities of the Titan IV (SRMU)/Centaur. In addition, the array size would impose severe limitations on spacecraft maneuverability, constrain instruments field-of-view, and increase the risk of electromagnetic and electrostatic interference impeding the performance of communications equipment.

Therefore, the Proposed Action has the greatest potential to accomplish the mission and its scientific objectives. It could be accomplished in a timely manner without a major disruption of the NASA, ESA, and ASI scientific programs. A launch during the Proposed

Action's contingency opportunities would result in reduced science return compared with the primary launch opportunity. In addition, the contingency opportunities would delay the science return (i.e., 2 years for the secondary opportunity and 4 years for the backup opportunity). This would entail additional costs to NASA and its international partners.

The alternatives to the Proposed Action are the 1999 mission alternative, the 2001 mission alternative, and the No-Action alternative. The 1999 mission alternative would involve dual Shuttle launches from KSC to deliver the upper stages and the Cassini spacecraft into low Earth orbit. An on-orbit mating of the spacecraft and upper stages would be performed by astronauts, followed by spacecraft injection into its VEEGA interplanetary trajectory. The configuration of the upper wages is currently undefined. The backup launch opportunity for this mission would occur in August 2000 using a VEEGA trajectory. The 1999 mission alternative would obtain less science return than the 1997 primary launch opportunity of the Proposed Action.

The 2001 mission alternative would involve a launch from CCAS on a Titan IV (SRMU)/Centaur in March 2001 using a VVVGA (non-Earth gravity-assist) trajectory. The backup launch opportunity would insert the spacecraft into a VEEGA trajectory using a Titan IV (SRMU)/Centaur launch from CCAS in May 2002. This alternative would result in a reduced science return from the primary launch opportunity of the Proposed Action due to the measures that would be taken to enable a launch on the Titan IV (SRMU)/Centaur (i.e.,- a reduction in the number of Titan flybys from 35 to 21, elimination of the SOI burn delay, and an extension of the initial orbit period). The spacecraft's data gathering activities at Saturn would be restricted to conserve the marginal quantity of propellant available.

The No-Action alternative would cancel the mission, forfeiting the opportunity to acquire significant mission-specific scientific data, which cannot be obtained by any other means. Cancellation of the Cassini mission would also terminate the international partnerships formed to develop the Cassini Orbiter and the Huygens Probe and would disrupt agreements made for other space-related projects.

Executive Summary

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Chapter 8

Appendix A

Appendix B

Appendix C

Appendix D

Appendix E