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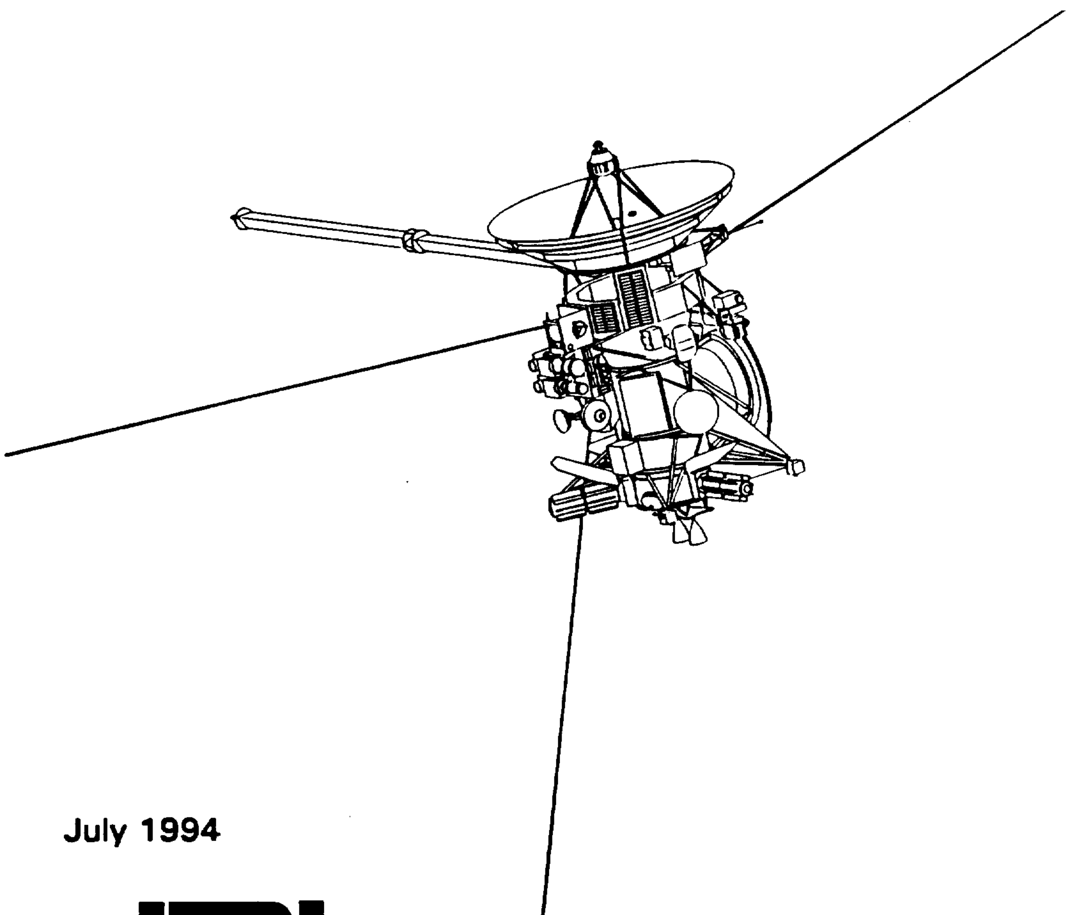
Volume 2: Alternate Mission and Power Study



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GLOSSARY OF ACRONYMS AND ABBREVIATIONS

AMTEC	Alkali Metal Thermoelectric Converter
AMO	Air Mass Zero
AP	Auxiliary Propulsion
APSA	Advanced Photovoltaic Solar Array
ASI	Italian Space Agency
BCA	Battery Control Assembly
BOL	Beginning of Life
BOM	Beginning of Mission
BTK	Block for Transfer and Correction
C_3	Launch energy (in km^2/s^2)
CC	Cargo Container
CEQ	Council on Environmental Quality
Cm-244	Curium-244
CPC	Compound Parabolic Concentrator
CR	Concentration Ratio
CTS	Communications Technology Satellite
d	Heliocentric range (in AU)
Δ(delta)	Change (i.e., difference between two values)
ΔV	Change in velocity
DIPS	Dynamic Isotope Power System
DoD	U.S. Department of Defense
DOE	U.S. Department of Energy
DOD	Depth of Discharge
DSN	Deep Space Network
EGA	Earth Gravity Assist
EIS	Environmental Impact Statement
EOL	End of Life
EOM	End of Mission
EOS-AM	Earth Observing System-Morning
ESA	European Space Agency
ETR	Eastern Test Range
EUS	Energia Upper Stage
FOV	Field-of-View
FPR	Flight Performance Reserve
FSAR	Final Safety Analysis Report
FY	Fiscal Year

g	Acceleration due to gravity on Earth's surface ($9.8\ m/s^2$ or $32\ ft/s^2$)
GaAs	Gallium Arsenide
Ge	Germanium
GLL	Galileo
GPHS	General Purpose Heat Source
GRS	Gamma Ray Spectrometer
GSO	Geo-Stationary Orbit
GTO	Geo-Transfer Orbit
HEO	High Earth Orbit
HGA	High Gain Antenna
IC	Internal Combustion
I_{sp}	Specific impulse (in seconds)
IUS	Inertial Upper Stage
JGA	Jupiter Gravity Assist
JPL	Jet Propulsion Laboratory
LEO	Low Earth Orbit
LeRC	Lewis Research Center
LGA	Low Gain Antenna
LILT	Low Intensity, Low Temperature
LV	Launch Vehicle
LVC	Launch Vehicle Contingency
MAG	Magnetometer Experiment
MHD	Magneto-Hydrodynamic
MHW	Multi-Hundred Watt
MO & DA	Mission Operations and Data Analysis
MOSO	Mission Operations Support Office (at JPL)
MSTI	Miniature Seeker Technology Integration
NASA	National Aeronautics and Space Administration
NEP	Nuclear-Electric Propulsion
NEPA	National Environmental Policy Act
NLS	National Launch System
PAM	Payload Assist Module
PAM-S	Payload Assist Module-Special
PLF	Payload Fairing
Pu-238	Plutonium-238
RCS	Reaction Control System
RDM	Radiation Design Margin
RHU	Radioisotope Heater Unit
RORSAT	Radar Ocean Reconnaissance Satellite

RTG	Radioisotope Thermoelectric Generator
SAR	Safety Analysis Report
SEEGA	Solar-Electric Earth Gravity Assist
SEP	Solar-Electric Propulsion
Si	Silicon
SLATS	Solar Linear Array Technology System
SOI	Saturn Orbital Injection
Sr-90	Strontium-9O
SRM	Solid Rocket Motor
SRMU	Solid Rocket Motor Upgrade
SSO	Sun-Synchronous Orbit
STP	Solar Thermal Propulsion
STS	Space Transportation System (i.e., the Space Shuttle)
t_{max}	Maximum off-Sun time (in hours)
TPV	Thermophotovoltaic
UARS	Upper Atmospheric Research Satellite
ULS	Ulysses
USAF	U.S. Air Force
$V_\propto$	Excess velocity at infinity on a hyperbolic orbit of departure
$V_\propto A$	Arrival velocity with respect to Saturn, just before Saturn has a significant gravitational effect on the spacecraft
V_s	Velocity of Saturn
V_E	Velocity of Earth
V_{sc}	Velocity of the spacecraft
VEEGA	Venus-Earth-Earth-Gravity-Assist
VEGA	Venus-Earth-Gravity-Assist
VIMS	Visual Infrared Mapping Spectrometer
VVEJGA	Venus-Venus-Earth-Jupiter-Gravity-Assist
VVVGA	Venus-Venus-Venus-Gravity-Assist
VVVJGA	Venus-Venus-Venus-Jupiter-Gravity-Assist
VVEGA	Venus-Venus-Earth-Gravity-Assist

Units of Measure

AU	astronomical unit
ci	curie
deg	degree
ft	foot
hr	hour
kg	kilogram
km	kilometer
krad	kilorad
kW	kilowatt
lb	pound
m	meter
nmi	nautical miles
s	second
V	Volts
W	Watts
W_e	Watts Electric
W_t	Watts Thermal

SECTION 1

INTRODUCTION

1.1 PURPOSE OF THE STUDY

Volume 2 is one of four documents that the Cassini Project at the Jet Propulsion Laboratory (JPL) has compiled to support the preparation of an Environmental Impact Statement (EIS) for the Cassini Program by the National Aeronautics and Space Administration (NASA). Its focus is on identifying and characterizing potential mission and power alternatives, and then comparing those alternatives with the Cassini baseline mission (i.e., the planned action). The major mission and spacecraft power options emerging from this process appear in Sections 3 and 4, respectively. This updated version of Volume 2 supersedes all previous versions of Volume 2, and characterizes additional potential launch vehicle alternatives for the Cassini mission. As described in Volume 1 of the EIS Supporting Studies, the baseline design includes a Titan IV Solid Rocket Motor Upgrade (SRMU)/Centaur launch system, one Earth gravity-assist (EGA) swingby and relies on radioisotope thermoelectric generators (RTGs) for electrical power generation on the spacecraft.

1.2 BACKGROUND

1.2.1 National Environmental Policy Act (NEPA)

As mandated by the National Environmental Policy Act (NEPA) of 1969 (42 U.S.C. 4321 et seq.), as amended, federal agency action significantly affecting the quality of the human environment requires an Environmental Impact Statement. NASA's regulations for implementing NEPA (NASA, 1980) require an EIS for development or operation of nuclear systems and requires a detailed look at alternatives. This document supports that effort.

1.2.2 Document History

The previous version of this document was published in June of 1993 and addressed a number of potential launch vehicle and power alternatives. However, the Office of Science and Technology Policy (OSTP) and the Office of Management and Budget (OMB) requested that NASA investigate for the Cassini mission additional potential launch vehicle alternatives, the possibility of reducing mission operations and data analysis costs, and the possibility of increasing mission success. To comply with this request, early in 1994, the JPL and Lewis Research Center (LeRC) provided NASA with feasibility evaluations of a number of additional potential launch system alternatives for the Cassini mission. The potential launch system alternatives that were investigated included:

- Two launches of the Space Transportation System (STS) with an upper stage and spacecraft/upper stage assembly on-orbit.

- Ariane 5 with a Centaur IIA upper stage.
- Proton-M with Block 'D' + Star 63F upper stages.
- Split mission using two Proton launches, sending two smaller orbiters (one with the Huygens probe) to Saturn.

The results of this study were presented to OSTP and OMB on February 2, 1994. In mid-February 1994, after NASA dismissed the Ariane and Proton options for technical and programmatic reasons, OSTP and OMB requested NASA to further consider launching Cassini via two STS missions, and compare that potential alternative with the Titan IV SRMU baseline. Specifically, NASA was asked to address relative benefits, costs, and possibilities of failure.[1] In late March 1994, an independent review panel compared the dual STS potential alternative with the baseline and concurred with the Cassini Project Office that the Titan IV (SRMU)/Centaur should be maintained as the baseline launch system for the 1997 Cassini mission. In addition to the information addressed in the June 1993 version, this current version addresses the above mentioned potential launch system alternatives, as well as updating information on the Russian Energia launch vehicle.

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[1] Letter to NASA Administrator, Daniel P. Goldin from OSTP, February 16, 1994.

SECTION 2

METHODOLOGY

2.1 STUDY SCOPE

NASA has established a set of scientific goals and objectives for Saturn's exploration and, together with the Cassini Project and the science community, developed a technical approach for achieving the most immediate and highest priority of those goals. This approach is embodied in the baseline Cassini spacecraft, mission, and power concepts, as discussed in Volume 1.

The purpose of Section 2 is to explain how potential alternatives to the baseline are identified, characterized, and compared. Comparisons are limited to areas that JPL is technically most qualified to address, such as certain aspects of solar system science and spacecraft engineering. No attempt is made to provide detailed hazard or environmental risk assessments, for example. Evaluations of that type will be performed in the Safety Analysis Reports (SARs) and the EIS, as will consideration of the “no-action” alternative and any policy compatibility issues.

2.2 STUDY APPROACH

In broad terms, the approach followed in performing this study consists of three principal steps:

(1) Identification. Identification of potential alternatives to the baseline capable of achieving the Cassini Program's scientific goals and objectives.

(2) Characterization. Definition of potential alternatives' physical and performance characteristics (i.e., to facilitate comparison with the baseline).

(3) Comparison. Comparison of potential alternatives with the baseline in terms of scientific and programmatic factors pertinent to decision makers.

The following subsections describe each of these steps in detail. Throughout this volume, emphasis has been placed on documenting the major considerations and findings associated with the performance of these steps, as opposed to presenting an actual progression through them.

2.2.1 Step 1: Identification of Potential Alternatives

In this study, a potentially feasible alternative is defined as a technical approach, different from the baseline, that would satisfy all three of the following criteria:

- Result in a smaller level of possible environmental impact;

- Be technically feasible to implement;
- Fulfill the Cassini science requirements.

2.2.1.1 Alternative Identification Criterion 1: Smaller Environmental Impact. In the unlikely event of a severe launch accident or inadvertent reentry[1] into the Earth's atmosphere during an Earth swingby, the principal source of environmental impact would be the potential release of plutonium into the environment by the baselined RTGs and RHUs (radioisotope heater units). Thus, the types of potential alternatives that might lead to satisfaction of this criterion would have a changed trajectory or power source, as follows:

- Utilize RTGs, but not use the Earth for a gravity assist;
- Replace the plutonium fuel in the RTGs with another radioisotope that is environmentally preferable in some way;
- Develop RTG designs that use less plutonium;
- Replace the RTGs with a nuclear reactor; or
- Replace the RTGs with non-nuclear power sources, such as solar arrays.

2.2.1.2 Alternative Identification Criterion 2: Technical Feasibility. Assessment of technical feasibility involves determining whether or not a particular technology is appropriate to the approach required for meeting Program objectives, and involves two primary considerations, availability and applicability.

A. Availability

Those systems that are technically mature and “space-qualified,” or are under development on a timetable that is consistent with the schedule for the Cassini Program, would be considered technically available. To base technical availability on the Program phase when the science data is obtained, rather than on the development/launch schedule, relaxes the constraints on technologies that may be “almost ready.” For example, an almost-ready launch vehicle large enough to permit a direct trajectory of shorter duration would allow a later launch date and still return the science data at the desired time. This would permit the project planners to consider waiting for a new technology that could offer significant advantages to the Program.

On the other hand, this study specifically excludes as feasible alternatives those systems that are under consideration but not on a firm development schedule. This approach is consistent with the standard practice of program planners and managers to consider the level of technical risk associated with various system design options, and to limit their options to those that offer reasonable prospects for success.

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[1] An inadvertent reentry could occur in two different ways. A short-term reentry is an inadvertent reentry occurring during a targeted Earth swingby. A long-term reentry could occur if the spacecraft became disabled, then went into an orbit about the Sun, and then reencountered the Earth potentially many years into the future.

B. Applicability

Equal in importance to technical availability is the applicability of the technology to the spacecraft/mission configuration under consideration. The technology must provide the performance and operating characteristics required by the remainder of the system without imposing new requirements of its own that cannot be accommodated by the system. For example, a power technology must be able to provide sufficient power to the science instruments when required without being so massive that the lift capabilities of the potential launch vehicle alternatives are exceeded.

2.2.1.3 Alternative Identification Criterion 3: Fulfill Science Requirements. A potential alternative's eligibility for further consideration is first determined by whether or not it satisfies all of the Cassini science requirements—requirements imposed to accomplish the overall mission science goals and objectives (Volume 1, Section 3). The study does, however, perform a brief sensitivity analysis (see “B” below) to determine the extent to which science requirements would have to be relaxed to potentially enable new alternatives.

A. Rationale for Mandating Compatibility with Science Requirements

Space programs are not initiated for their own sake; they must have a purpose. The science objectives provide a basis for deciding whether or not a program is worth the associated expenditure of public resources. Science objectives play a key role in determining costs because they impose constraints and requirements on the spacecraft, mission, and power designs. The importance of adhering to a specific set of science objectives in designing the Cassini mission and spacecraft cannot be overstated.

To make a Cassini-type mission alternative worthwhile, for example, the spacecraft would have to be an orbiter capable of looking at a wide range of phenomena over a period of time. Flybys of Saturn have already been done. Because of this, in searching for potential spacecraft, mission, and power alternatives to the Cassini baseline, this study's investigation has been restricted to those alternatives satisfying the science objectives enumerated in Section 3 of Volume 1. Table 2-1 summarizes the criteria used to select potential alternatives.

B. Sensitivity to Science Requirements

While this investigation targets those potential alternatives satisfying all the Cassini science objectives and requirements, it also considers the possibility of relaxing this criterion. Can the technical feasibility of existing alternatives be enhanced by reducing the science return? Are new alternatives possible? If the answer to one or both of these questions turns out to be yes, at what cost, in terms of lost science, are these gains acquired? Subsections 3.4 and 4.4 attempt to provide qualitative answers to these questions.

Table 2-1. Criteria Used to Select Potential Alternatives to the Planned Action

1. REDUCTION OF POSSIBLE ENVIRONMENTAL HAZARDS Missions/Spacecraft have one or more of the following characteristics: • Do not require Earth swingbys for gravity assist • Replace the plutonium fuel in the RTGs with another radioisotope • Utilize RTG designs that use less Pu-238 • Replace the RTGs with a nuclear reactor • Utilize non-nuclear spacecraft power sources, such as solar arrays
2. TECHNICAL FEASIBILITY Technology must be <u>available:</u> • Technology must be sufficiently mature • Hardware/systems must be available to the Project when needed • The design must offer reasonable prospects for mission success Technology must be <u>applicable</u>: • Technology must be compatible with system elements • Technology must not exceed physical constraints on flight systems
3. ABILITY TO MEET CASSINI PROGRAM SCIENCE GOALS

It should be noted that reducing the number of science objectives does not necessarily simplify the design in any direct proportion. Most science requirements are addressed by more than one instrument; each instrument generally contributes to multiple objectives. These factors are interdependent and not easily separable.

2.2.2 Step2: Characterization of Potential Alternatives

After identifying all of the potentially feasible alternatives to the proposed action, the next step is to characterize them. Characterization utilizes a variety of physical and performance parameters to highlight measurable differences between the alternative and the baseline in areas of interest to Program decision makers. The main spacecraft and mission parameters used to characterize alternatives are summarized in Tables 2-2 and 2-3, respectively. The tables are organized to reflect whether a parameter's primary effect would be science- or Program-related.

2.2.3 Step 3: Comparison of Potential Alternatives with the Baseline

The final step is to assess the impacts on science and Program performance, relative to the baseline, that would result from implementing a specific alternative. The science and Program parameters delineated in Tables 2-2 and 2-3, and the satisfaction of science objectives (Volume 1, Section 3) provide the basis for this comparison. Throughout the volume, content is focused on documenting the major considerations and findings as opposed to presenting a mechanical, step-wise progression through all of the characterizations and comparisons.

2.3 STUDY METHODS

It is the practice of this study to consider a potential alternative only up to the point at which it ceased to be at least marginally viable. In many cases, qualitative considerations are adequate to arrive at a conclusion regarding a particular alternative's viability. Quantitative analysis is only applied in situations where the alternative's technical feasibility or capability of satisfying the science objectives cannot be clearly established by inspection. For instance, if a potential alternative depends on one major device or piece of equipment that is still in the concept or design stage, with no plans to build the first one until after the baseline date for arrival at Saturn, that alternative is rejected.

Table 2-2. Example Parameters/Factors Used to Characterize Potential Spacecraft Power Alternatives

Science-Related	Program-Related
Mass Instrument Field of View Pointing Accuracy Turn Times Hydrazine Requirements Navigation Capability Electromagnetic (EM)/Thermal Interference Generation Power Requirements	Spacecraft Development Cost Safety Hazards Reliability/Risk (Space environment tolerance, technical development risk, operational reliability)

Table 2-3. Example Parameters/Factors Used to Characterize Potential Mission Alternatives

Science-Related	Program-Related
Launch Period Bi-Propellant Requirements Saturn Initial Orbit Period Tour Duration	Launch Period Launch Vehicle Margins Flight Time Mission Duration Operations Cost Backup Mission Availability

SECTION 3

CASSINI MISSION ALTERNATIVES

This section discusses the potential trajectories and the potential launch system alternatives for the Cassini mission. Potential Cassini launch system alternatives are only summarized in this section, but are fully discussed in Appendix A. Additional information regarding the types of possible trajectories to Saturn is also provided in Appendix A, while this section focuses on the actual potential trajectory opportunities for the Cassini mission. The potential launch vehicle alternatives, trajectory alternatives, and mission alternatives are evaluated using Section 2 criteria and compared to the Titan IV (SRMU)/Centaur baseline, with respect to trajectory opportunities, mission science return, backup launch opportunities, and general concerns. The launch vehicles and trajectories examined here represent the results of a search for the optimum launch opportunities for the Cassini mission.

3.1 POTENTIAL LAUNCH SYSTEM ALTERNATIVES

As discussed in Appendix A, feasibility and performance were the overriding considerations when selecting the potential launch system alternatives examined in this study. The baseline Titan IV (SRMU)/Centaur is examined in Appendix A and described in Volume 1. The potential launch system alternatives evaluated for the Cassini mission included:

- Titan IV (SRMU) with Titan Centaur.
- Titan IV (SRM) with Titan Centaur.
- Space Transportation System (STS), also known as the Space Shuttle, with either an Inertial Upper Stage (IUS) or with an IUS and Payload Assist Module-Special (PAM-S).
- Two launches of the Space Transportation System (STS) with an as yet undeveloped powerful upper stage followed by upper stage/spacecraft assembly while in low Earth orbit.
- Ariane 5 with Centaur IIA upper stage.
- Energia with the Energia Upper Stage (EUS) and/or the Retro and Correction Stage (RCS).
- Energia-M with Centaur IIA or Block 'DM' + Star 63F.
- Proton-M with Block 'D' + Star 63F.
- Split mission using two Proton launches, sending two smaller orbiters (one with the Huygens probe) to Saturn.

These potential launch system alternatives are fully discussed in Appendix A and the major elements of that discussion are presented in Tables 3-1a and 3-1b.

Table 3-1a. Summary of Potential U.S. Launch System Alternatives

	Titan IV SRM with Centaur	**Titan IV SRMU with Centaur**	**Single STS with IUS/PAM-S**	**Dual STS with Assembly on-orbit**
Qualifying Launch Opportunities				
Oct'97 VVEJGA	•	•	X	⊗
Dec '97 VEEGA	•	•	X	⊗
Mar '99 VEEGA	•	•	X	•
Aug '00 VEEGA	•	•	X	•
Jan '01 VEEGA	•	•	X	•
Mar '01 VVVGA	X	•	X	X
Mar '02 VVVGA	X	X	X	X
May '02 VEEGA	•	•	X	•
'97,'98, '99 JGA	X	X	X	X
Science Return for Equivalent Launch Dates*	Would return less science.	Would return the maximum science for the launch opportunity.	Zero science since no qualifying launch opportunities.	Would return less science.
Concerns	• Less performance than baseline. • No non-EGA opportunities.	• Spacecraft needs rhenium engine for non-EGA opportunity.≠ • No non-EGA backup opportunities for March 2001.	• Not technically feasible.	• Requires development of new upper stage. • Assembly on-orbit increases technical complexity. • No non-EGA opportunities.

- • The launch system has the energy to inject the Cassini spacecraft into the trajectory. T
- ⊗ There is 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.
- X The launch system does not have the energy to inject the Cassini spacecraft into the trajectory.
- * Amount of science return expected compared to using a Titan IV (SRMU)/Centaur for the same launch date if it is a qualifying launch opportunity.
- ≠ The rhenium engine is a spacecraft engine with a rocket chamber fabricated from rhenium and has an internal oxidation-resistant iridium coating. A version of this engine that uses mono-methyl hydrazine fuel has been in development for NASA missions. Another version of the engine, using nitrogen tetroxide, is being developed for commercial spacecraft use. Neither engine has been f flight-tested, and NASA currently has no funds for the qualification or testing of the engine.

Table 3-1b. Summary of Potential Foreign Launch System Alternatives

	Ariane 5 with Centaur I IA	Energia with EUS/RCS	Energia-M with Centaur I IA	Proton-M with Block D'+Star63F	Dual Proton-M with Block 'D'+Star63
Qualifying Launch Opportunities					
Oct '97 VVEJGA	†	†	†	†	†
Dec '97 VEEGA	†	†	†	†	†
Mar '99 VEEGA	•	•	•	•	•
Aug '00 VEEGA	•	•	•	•	•
Jan '01 VEEGA	•	•	•	•	•
Mar '01 VVVGA	•	•	•	X	X
Mar '02 VVVGA	•	•	•	X	X
May '02 VEEGA	•	•	•	•	•
'97, '98, '99 JGA	X	•	X	X	X
Science Return for Equivalent Launch Dates*	Would return the maximum science for the launch opportunity.	Would return the maximum science for the launch opportunity.	Would return the maximum science for the launch opportunity.	Would return less science.	Would return the maximum science for the launch opportunity.
Concerns	• Spacecraft needs rhenium engine for non-EGA opportunities.≠ Would need idealized rhenium engine performance for March 2002. • Needs development of new upper stage configuration and interfaces. Including substituting an Ariane 52nd stage with a Centaur IIA. • PLF would be too small for the Cassini spacecraft.	• Development of upper stages is uncertain. • Lack of information available with respect to the booster and upper stages to assess technical feasibility. • Only 2 Energia launches to date and there have not been any Energia launches since 1988.	• Development of Energia-M is highly uncertain. • Very long cruise time of 11.7 years for the March 2002 opportunity. • Needs development of new upper stage configuration. • March 2002 requires substantially larger bipropellant tank.	• Needs development of new upper stage configuration. • Evaluation based on maximum possible performance. • Needs larger PLF, which most likely will make mission technically infeasible due to increased mass. • No flight termination system. • No non-EGA opportunities.	• Would use a total of 5 RTGs. • Twice the number of gravity assists. • Needs development of new upper stage configuration • No non-EGA opportunities • Need new payload fairing.

• The launch system has the energy to inject the Cassini spacecraft into the trajectory.
† Even if this launch system was determined to be available and 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.
X The launch system does not have the energy to inject the Cassini spacecraft into the trajectory.
* Amount of science return expected compared to using a Titan IV (SRMU)/Centaur for the same launch date if it is a qualifying launch opportunity.
≠ See equivalent note in Table 3-1a.

3.2 POTENTIAL TRAJECTORY ALTERNATIVES

Appendix A, Subsection A.1 introduces various trajectory concepts, as well as three general types of interplanetary trajectories. In the subsection below, direct and gravity-assist trajectories are investigated as potential alternatives to the Cassini baseline trajectory. Using a direct trajectory or a single Jupiter gravity assist instead of multiple Venus or Earth swingbys would have many advantages. Trajectories dependent upon low-thrust or other exotic propulsion modes (see Appendix A.1.3), speculative technologies, and prohibitively large spacecraft masses are excluded from this analysis.

To qualify as an alternative to the baseline, a trajectory must have mass performance characteristics that enable the Cassini science objectives to be achieved, as prescribed in Section 2. For example, an established minimum criterion (JPL, 1992b, Subsection 4.1.5) is that the satellite tour must provide at least 21 Titan flybys.

Only the March 1996 Venus-Venus-Venus-Jupiter Gravity Assist (VVVJGA) and the March 2001 Venus-Venus-Venus Gravity Assist (VVVGA) missions can be considered as potential primary trajectory alternatives to the baseline. The scientific and programmatic implications of switching to the March 1996 VVVJGA or the March 2001 VVVGA trajectories using the Titan IV (SRMU)/Centaur are discussed in Subsection 3.3. Any other non-EGA trajectories would require a more powerful launch system, even with extended flight times.

3.2.1 Direct and Jupiter Gravity-Assist Trajectories

Direct trajectories are the simplest interplanetary transfers and are characterized by very high launch energies and short flight times. They impose no special requirements on spacecraft design, and would always be the preferred transfer mode if sufficient launch vehicle performance were available.

A Jupiter Gravity Assist (JGA) greatly reduces the required injection energy while retaining most of the advantages of a direct trajectory. Even with injection energy reduced, however, the required performance for a JGA would still exceed the performance of a Titan IV (SRMU)/Centaur.

3.2.1.1 Opportunities. Because Jupiter is so massive, a relatively large deviation from optimal phasing can be tolerated. JGAs to Saturn are typically available for three out of every 20 years. In this decade, the launch opportunities occur in 1997, 1998, and 1999. Required launch energies vary greatly as a function of launch year and flight time. For example, a launch in 1997 with a flight time of 6.5 years requires a launch energy, or C_3, of 76 km^2/s^2; a 1999 launch with a 4.5-year flight time has a C_3 of 104 km^2/s^2. Trajectory optimization sometimes results in a plane-change maneuver being inserted between launch and Jupiter, typically on the order of 300 m/s. The size of the Saturn orbit insertion (SOI) maneuver varies significantly with flight time, arrival speed, launch year, and the period of the initial orbit around

Saturn. The maneuver varies from a low of 290 m/s (6.5-year flight time, launching in 1998) to a high of 1020 m/s (4.5-year flight time, launching in 1997).

3.2.1.2 Performance. Using a mass value of 2100 kg for the dry orbiter (baseline mass minus 50 kg for sunshades and other thermal protection not needed for JGA trajectories), 352 kg for the Probe and its support equipment, and 190 kg for the launch vehicle adapter, JGA performance was computed for the 1998 opportunity. The resulting injected mass requirement is about 4600 kg at a C_3 of 87 km^2/s^2, for a flight time of five years. This far exceeds the Titan IV (SRMU)/Centaur's capability. Changes to the flight time, launch year, or mission profile will not result in a significant improvement.

3.2.1.3 Comparison of Energia-Enabled JGA Trajectories with the Baseline. The advantages of a JGA trajectory over a more intricate, gravity-assist trajectory using the inner planets include reduced flight time, less opportunity for spacecraft failure, reduced thermal loads, and earlier science return. For Cassini, flight times would be reduced by roughly two years. Savings from lower mission operations costs could be used to defray launch vehicle costs.

The primary disadvantage of using the JGA trajectories in this case involves the reliance on the Energia booster, along with untested upper stages. The various technical risks associated with Energia use are described in Appendix A.2.3.

3.2.2 Non-Qualifying Gravity-Assist Trajectories

Some trajectories use phasing orbits as a way of incorporating Earth, Venus, and sometimes Mars gravity-assist swingbys. The period of a phasing orbit is approximately a multiple of the planet's orbit period. Near the point of greatest distance from the Sun (i.e., aphelion) of the phasing orbit, a large ΔV (i.e., velocity change) maneuver is performed to slow the spacecraft and target it for the 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. Trajectories using combinations of Venus and Earth gravity-assist swingbys, such as VEGA, VVEGA, and VEEGA, have an important advantage: their launch energy requirement is low, typically between 10 and 20 km^2/s^2 for launch C_3. Non-optimal encounter sequences are tightly coupled to penalties in launch energy and post-launch ΔV requirements. VEEGA trajectories provide more phasing flexibility. The baseline backup VEEGA trajectory would have a post-launch ΔV(deterministic maneuvers through SOI) in the range of 900 m/s, for example. The addition of a Jupiter gravity-assist swingby after the final Earth or Venus swingby, such as VVEJGA or VVVJGA, makes it possible to deliver a spacecraft to an outer planet beyond Jupiter, either with a shorter flight time and/or for a lower post-launch ΔV. The baseline primary VVEJGA has a ΔV in the neighborhood of 1000 m/s.

Some backup opportunities for a mission may entail EGAs. Missions that have a non-EGA primary and an EGA backup (rather than a non-EGA backup) exist and will be noted. Among the other planets, the following analysis indicates that the phasing is not proper for Mars-Jupiter gravity-assist trajectories, and a Venus-Mars or Mars-only gravity-assist trajectory would provide no significant benefit.

3.2.2.1 Mars-Only Gravity-Assist Trajectories. The planet Mars would appear to be an obvious candidate for a gravity-assist swingby for missions to comets or outer solar system planets. Adding a Mars swingby to a direct trajectory reduces the required C_3 from about 110 to about 90 km^2/s^2. However, due to its relatively low mass (about 1/10 that of Earth), Mars is not very effective in increasing the energy of a spacecraft trajectory during a fast swingby. The most useful potential application of a Mars gravity assist would be to reduce or eliminate the ΔV at aphelion required in a ΔV-EGA-type trajectory. The Mars encounter would have to occur near aphelion of the Earth-to-Earth loop, and the orientation of this trajectory segment would have to fit with any subsequent gravity assists and, ultimately, the transfer to Saturn. The correct phasing of Earth, Mars, and Saturn occurs roughly every 36 years; the last opportunity was in 1975, and the next will not be until 2011.

Mars can also improve performance when it happens to be fairly near a trajectory that can reach the target body without considering Mars. Such was the case for the 1982 and 1984 Galileo launch opportunities to Jupiter, which planned to use Mars swingbys for performance enhancement. Although never flown, these were essentially direct Earth-Jupiter trajectories that gained some performance benefit because Mars was “along the way.”

3.2.2.2 Mars-Venus Gravity-Assist Trajectories. As noted, Mars' small mass makes it a relatively ineffective gravity-assist body if the swingby takes place when the spacecraft is outbound from the Sun. Mars swingbys can sometimes be of value, however, if they take place during an intermediate phase of the trajectory when the spacecraft velocity is lower. In particular, adding a Mars swingby to a multiple-Venus swingby trajectory might be of substantial benefit, provided that the phasing between Venus and Mars were favorable. The two poorer performing Venus gravity-assist flight paths identified in 1997 and 2001 were studied (see Subsection 3.2.3.1) to determine whether a Mars swingby could provide any performance improvement. Investigation showed that the Venus/Mars phasing was not favorable for these cases.

3.2.2.3 Mars-Jupiter Gravity-Assist Trajectories. Combining Mars and Jupiter gravity assists can bring the C_3 down to the 60 to 70 km^2/s^2 range, again requiring roughly the same post-launch ΔVs as the direct trajectories. The Earth-Mars phasing to Jupiter occurs only once every 50 years, however, and Saturn will not be in the proper place to take advantage of the 2032 alignment. The most recent opportunity was in 1982. Including both Mars and Jupiter into the phasing scheme serves to make this trajectory extremely rare.

3.2.3 Qualifying Gravity-Assist Trajectories

This subsection will address the possibility that non-Earth gravity-assist trajectories may result in viable Cassini missions. The addition of one or more planetary swingbys can greatly improve mission performance by reducing C_3 and, in some cases, ΔV. There is usually a flight time penalty and there may be other impacts on spacecraft design or mission operations. Nonetheless, the reduction in launch vehicle performance requirements can be the determining factor in mission feasibility studies. Qualifying non-Earth gravity-assist trajectories to Saturn using Venus alone (VVVGA) and Venus-Jupiter (VVVJGA) have been identified.

3.2.3.1 Venus-Only Gravity-Assist Trajectories. Mars and/or Jupiter gravity-assist trajectories are essentially direct trajectories that have been phased to take advantage of the intervening bodies; trajectories that include Venus swingbys are fundamentally different. The spacecraft must first be sent toward the inner solar system, and must generally make more than one complete loop around the Sun before being directed toward the outer planets. This results in a significant flight time penalty, but the improvement in performance can be substantial. Note that trajectories using Venus flybys place substantially greater thermal loads on the spacecraft because of much closer approaches to the Sun, resulting in likely mass penalties for added thermal protection and imposing additional operational complexity on the mission.

In order to reach Saturn with a launch energy and post-launch ΔV that are sufficiently low, at least three Venus gravity assists are needed (VVVGA). There are several types of VVVGAs available, all characterized by the ratio of Venus revolutions to spacecraft revolutions about the Sun. Analysis has shown that the most favorable combination for VVVGA trajectories to Saturn yields the time between the first two Venus swingbys to be two Venus years (i.e., about 450 Earth days) and between the second and third swingbys to be five Venus years (Bender, 1991). Several VWGA opportunities exist between 1996 and 2004, but only the 1996 and 2001 launches might be possible using the Titan IV (SRMU)/Centaur (see below under Summary for further evaluation). Additionally, there is a March 2002 VVVGA opportunity that possibly could satisfy the minimum science return criterion if Cassini was launched using an Energia-M (if it ever is developed). This potential alternative is discussed more in Subsection 3.3.3.3.

3.2.3.2 Venus-Jupiter Gravity-Assist Trajectories. Just as the baseline 1997 VVEJGA trajectory is enhanced by a Jupiter/Saturn opportunity, the VVVGA launch in 1996 also can be enhanced by a Jupiter swingby. The time between Venus swingbys for the resulting VVVJGA trajectory must be reduced to a two to four Venus-year phasing scheme to enable the launch in 1996. The VVVJGA trajectory has a substantially lower injection energy and slightly less post-launch ΔV than the VVVGA. For launches later than 1996, there is insufficient time to complete the required three Venus swingbys and reach Jupiter before the phasing is lost. Thus, the 1996 VVVJGA is the last such trajectory until the Jupiter/Saturn phasing repeats 17 to 20 years later.

3.2.3.3 Summary. Table 3-2 summarizes the best identified, non-Earth gravity-assist trajectories, along with their required injection energies and masses. Mass assumptions for the various missions are shown as notes in Table 3-2. Also, the baseline Cassini spacecraft mass was used, except that 10 kg was added for every 100 kg of bipropellant required beyond the capacity of the baseline tank size of 3000 kg (as an estimate of the increase in mass required for larger tanks). The

10 kg is a mass that has been used by the Cassini Project for small propellant changes around the nominal 3000 kg. An issue that would need to be addressed is whether or not the increase in size of the tanks, and possibly other structures, would still allow the spacecraft to fit within the currently planned payload fairing (PLF) of the launch vehicle.

Figures 3-1 and 3-2 are plots of injected mass capability versus injection energy for the potential launch vehicle altematives considered. These plots reflect the launch vehicle performance data (see Appendix A.2) and show the investigated trajectories to Saturn. Each of the trajectories identified in Table 3-2, as well as direct trajectories, is represented by a shaded box on the figures. The horizontal extent of each box shows the range of injection energies required by the trajectory across a nominal launch period. The vertical extent shows the range of the injected mass requirement for each trajectory. This mass uncertainty is primarily a function of flight time. The upper and lower bounds of these ranges correspond to the shortest and longest flight times considered reasonable for this trajectory, respectively.

Table 3-2. Gravity-Assist Cassini Mission Opportunities (No Earth Swingbys)

Trajectory Type	**Launch Date**	**Flight Time (yrs)**	**Launch Energy (km^2/s^2)**	**Post-Launch Delta-V (m/s)**	**Biprop Load (kg)**	**Injected Mass (kg)**
JGA	6/98	5.0	87.4	573	1790	4560
VVVGA	3/96	9.7	28.8	1743	4310	7280
	12/97	10.2	17.7	2004	5280	8330
	7/99	10.0	58.5	1095	2790	5610
	3/99	9.7	43.5	1807	4620	7620
	3/01*	10.3	16.5	1764	4365	7325
	3/02*	11.7	10.9	2130	5175	8215
VVVJGA	3/96	9.4	17.2	1486	3590	6470

NOTES: (1) Post-launch ΔV includes deterministic interplanetary maneuvers and the SOI maneuver only (SOI includes gravity losses due to bum offset).

(2) Calculations assume: Dry S/C mass = 2150 kg (Venus swingbys) or 2100 kg (JGA) + 10 kg tank mass per 100 kg of bipropellant required over 3000 kg, Probe =306 kg (343 kg), Probe support h/w = 46 kg (30 kg), launch vehicle adapter = 190 kg (165 kg), hydrazine =132 kg, interplanetary nav = 191 m/s (161 m/s), tour biprop ΔV=471 m/s, initial orbit = 362 m/s (345 m/s), specific impulse of biprop system = 308 s, 2.7% of propellant unusable.

*Masses assumed for evaluating the 2001 and 2002 VVVGA missions are in parentheses (in note 2). They reflect the most recent analysis and up to date design information. This mass change is not significant enough to alter the conclusions that should be drawn from the earlier analysis done for the other trajectories.

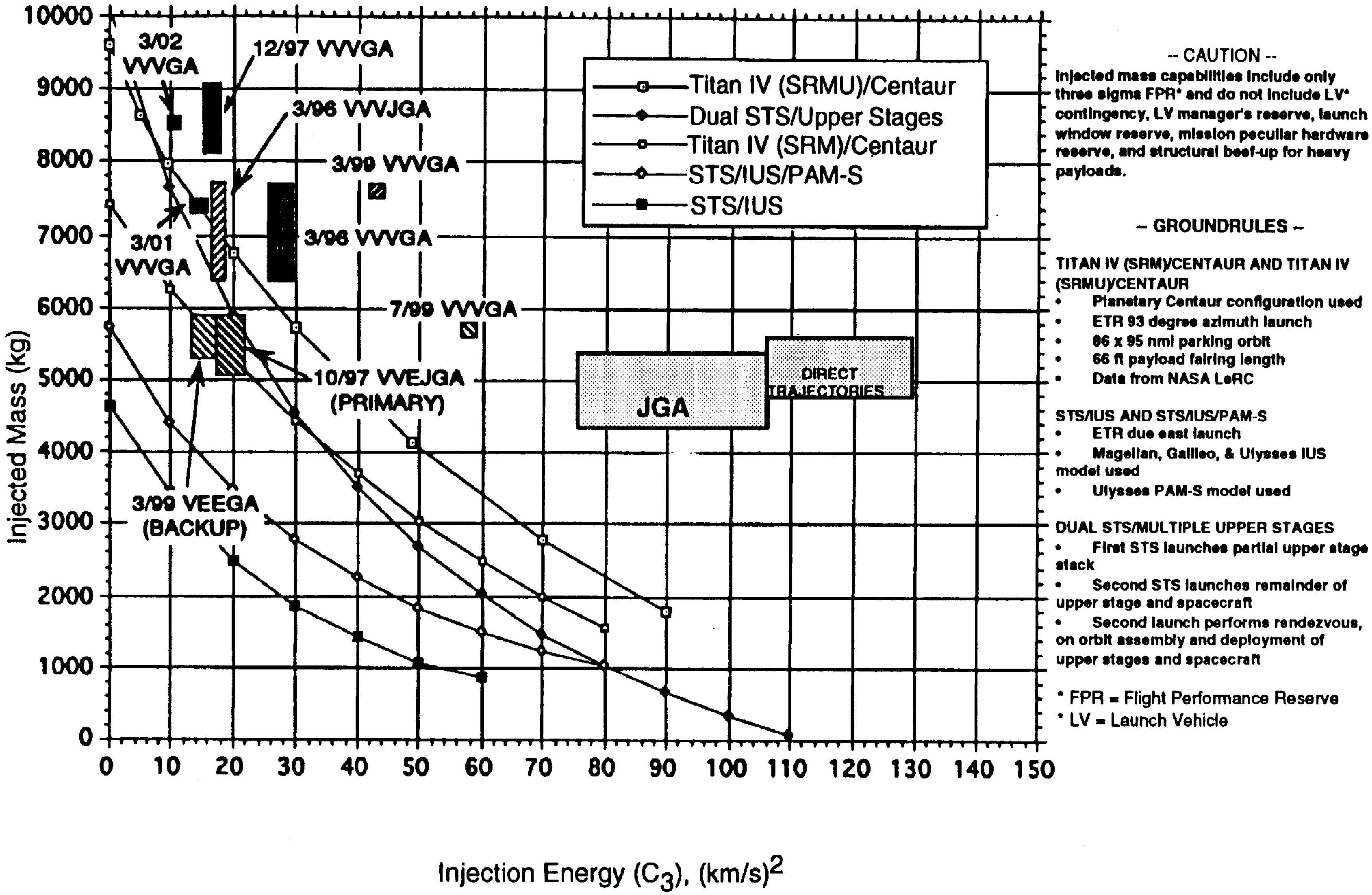


Figure 3-1. Cassini Alternate Trajectory Performance with U.S. Launch Vehicles

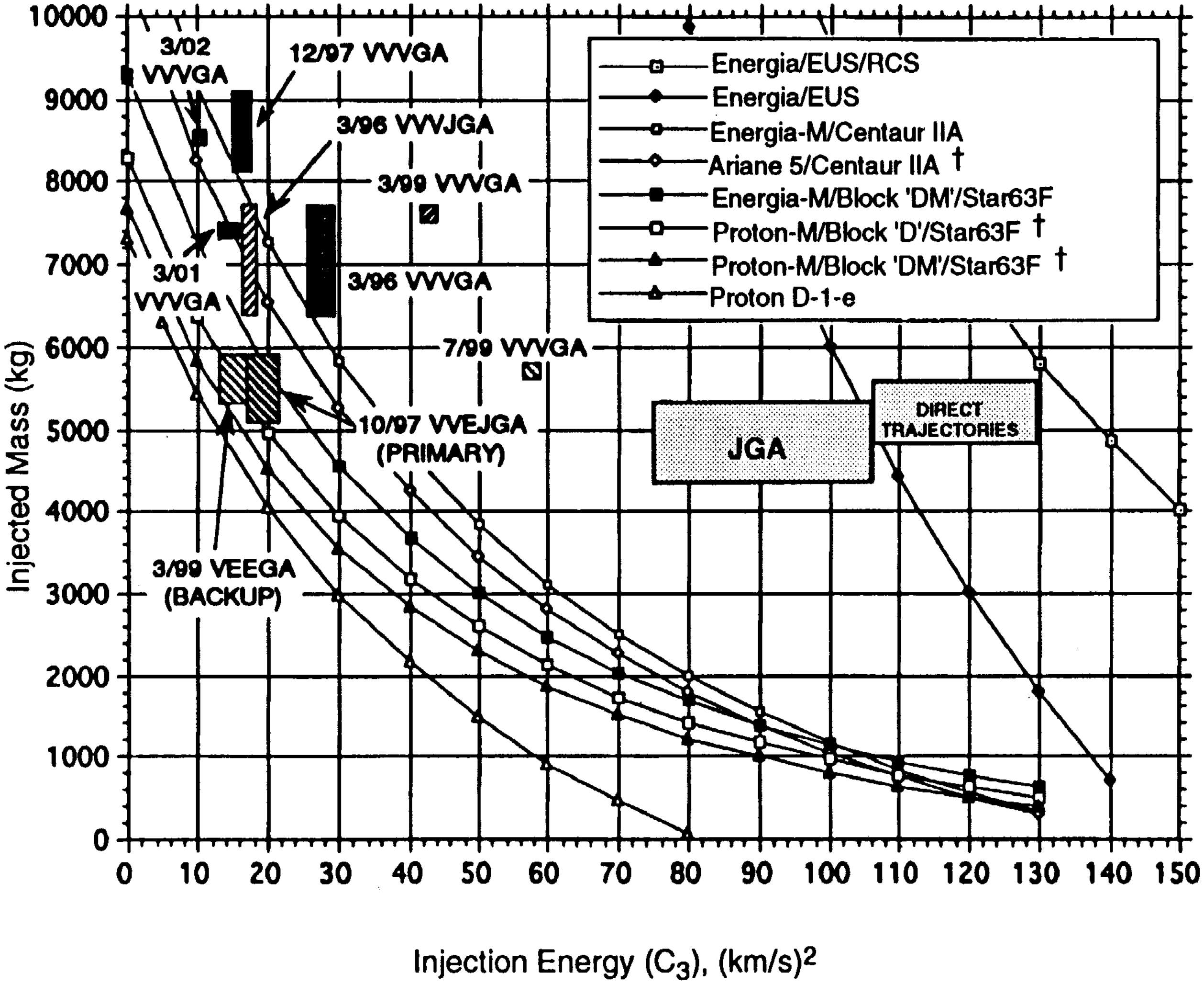


Figure 3-2. Cassini Alternate Trajectory Performance with Foreign Launch Vehicles

-- CAUTION --

Injected mass capabilities include only three sigma FPR* and do not include LV* contingency, LV manager's reserve, launch window reserve, mission peculiar hardware reserve, and structural beef-up for heavy payloads.

-- GROUNDRULES --

ENERGIA/EUS AND ENERGIA/EUS/RCS
- Energia launch from Baikonur to 200 km circular, 51.6 degree parking orbit
- Energia uses four strap-ons for EUS and six or eight for EUS/RCS upper stages
- First EUS burn used for parking orbit insertion
- 25% performance reduction for uncertainty in EUS parameters

ENERGIA-M/CENTAUR IIA AND ENERGIA-M/BLOCK 'DM'/STAR63F
- Energia-M launch from Baikonur to 200 km circular, 51.6 degree parking orbit with Block 'DM' or Centaur IIA first burn
- Energia-M smaller version of Energia
- Block 'DM' is Proton current 4th stage
- Centaur IIA is Atlas IIA upper stage

ARIANE 5/CENTAUR IIA
- Ariane 5 launched from Kourou in French Guiana
- Ariane 5 second stage replaced by Centaur IIA
- Ariane 5 injection to 75 by 200 km, 5 degree inclination parking orbit
- Centaur IIA off-loaded by about 3500 kg

PROTON-M/BLOCK 'D' or 'DM'/STAR63F AND PROTON D-1-e
- Proton D-1-e is current configuration
- Proton-M will be modernized version of three stage Proton
- All Proton launches from Baikonur
- For heavy payloads a first burn of Block 'DM' or 'D' needed to insert into 200 km, 51.6 degree inclination parking orbit

* FPR = Flight Performance Reserve
* LV = Launch Vehicle
† Does not account for extra mass penalty from required larger payload fairing.

The baseline primary and backup trajectories are also shown in Figures 3-1 and 3-2. For these, the bounds of the payload mass ranges were computed slightly differently than for the rest of the trajectories. For the upper bounds, mission designs (including flight time, number of Titan flybys, etc.) were selected that would require full propellant loading, which can be used only if SRMUs are available for launch. The lower bounds correspond to propellant loading that is consistent with an SRM-based mission. When using these figures to evaluate potential alternatives to the baseline missions, one must consider the fact that the launch vehicle curves shown depict idealized performance, as described in Appendix A.2.1. In reality, the holding of several required reserves results in a reduced level of performance that is actually available for designing a mission. For example, the currently recommended reserves for the Titan IV (SRMU)/Centaur add up to about 14% of the maximum performance capability.

The injection capabilities of several potential launch vehicles are represented by the curves in Figures 3-1 and 3-2. If a trajectory's box lies well above a given curve, this indicates that the trajectory is not compatible with this launch vehicle. Conversely, if a box lies well below a given curve, then the trajectory falls within that launch vehicle's capabilities. Trajectories whose boxes are close to a curve would require further analysis to determine whether they are marginally feasible for that launch vehicle.

As illustrated in Figure 3-1, none of the non-Earth gravity-assist trajectory opportunities fall within the Titan IV (SRM)/Centaur's capability. The March 1996 VVVJGA trajectory is the only opportunity, other than the baseline missions, falling at least partly below the curve for the Titan IV (SRMU)/Centaur, the most capable qualifying launch vehicle. It is therefore selected for more study in Subsection 3.3 as a potentially feasible alternative. The March 2001 and 2002 VVVGA trajectories were also selected for further evaluation. The March 1996 VVVGA is farther above the curve, but close enough that it, too, could conceivably be evaluated further. However, it is not selected for additional study because the VVVJGA opportunity is available for the same launch date and offers much better performance. The December 1997 VVVGA, while falling well above the curve, will be addressed in Subsection 3.4. The remaining trajectories are too far from the curve to warrant further examination.

3.3 CASSINI MISSION ALTERNATIVES: COMPARISONS WITH THE BASELINE

3.3.1 Overview

Both the March 1996 VVVJGA and the March 2001 VVVGA alternative missions require substantially more propellant than the baseline primary and backup missions, but, based on preliminary analysis, cannot be eliminated from further consideration. Therefore, in this subsection, these two potential missions are examined further and compared to the baseline primary mission. Before either could be considered as a serious alternative, however, a more rigorous study of the mission and the required modifications would need to be carried out.

Use of either alternate trajectory would have some impact on the overall mission science return. This has not been studied in detail, but is likely to be moderate for the 1996 VVVJGA, as long as all the science instruments could be ready in time, and more substantial for the 2001 VVVGA, although still meeting minimal criteria.

A more significant disadvantage to using the March 1996 trajectory is, however, the lack of a timely, non-EGA backup; the only known candidate is the March 2001 VVVGA alternate primary mission (Subsection 3.3.2.3). As discussed in Subsection 3.3.3, utilizing the March 2001 VVVGA would result in an increased risk of spacecraft malfunction. A further drawback to either of the alternate missions would be the extra program costs resulting from longer flight times and mission durations. The 2001 date would entail substantial costs due to holding the development team together, storage costs, post-storage testing, RTG power degradation, and spacecraft requalification (see Subsections 3.3.2.4 and 3.3.3.4).

In comparing each of the technically feasible alternatives with the baseline, many factors, such as the examples found in Subsection 2.2.2, were considered. The factors which were of greatest importance are:

- <u>Mission Science Return</u> (e.g., ring observations, Saturn radio occultations, Titan flybys)
- <u>Operational Reliability</u> (e.g., increased cruise duration; the possibility of malfunctions)
- <u>Backup to the Alternate Mission</u> (e.g., programmatic risk; the lack of a backup trajectory)
- <u>Program Cost</u> (e.g., costs of accelerated or delayed development/ launch; operations costs)

These factors are discussed in detail for each of the potentially feasible alternatives appearing below in Subsections 3.3.2 and 3.3.3.

3.3.2 March 1996 VVVJGA Alternate Mission

The VVVJGA trajectory launches in March 1996 and arrives at Saturn about 9.4 years later. Preliminary analysis has shown that the launch period runs from March 12 through March 30, 1996. The first Venus swingby occurs in June of 1996 and places the spacecraft on course for the second Venus swingby in September 1997, exactly two Venus years later. Both of the first two Venus swingbys occur with a hyperbolic velocity, or V_∞. of about 8.8 km/s. A deep space maneuver near the following aphelion results in a V_∞ of about 13.8 km/s at the third Venus swingby in July 1999, and a maneuver about 40 days after the swingby adjusts the orbital geometry and targets the spacecraft to Jupiter. The Jupiter swingby takes place in February 2001 at a distance of about 75 Jupiter radii. Saturn arrival is in July 2005. The Saturn encounter phase follows the same sequence as the baseline mission: Probe release and data relay occur near the end of the initial orbit, and the Orbiter then continues on its orbital tour until about July 2009.

In order to further compare this mission with the baseline, the end-of-mission (EOM) ΔV is estimated by calculating the propellant load achievable with respect to a performance estimate of the Titan IV (SRMU)/Centaur (with 14% propellant reserves). Assuming that no injection margin is held beyond the recommended reserves of 875 kg for the launch vehicle (performance quotation from Lewis Research Center, November 1992), the propellant load for the 9.4-year flight time mission would be limited to 3425 kg. These calculations assume the same dry mass allocations described in Note 2 of Table 3-2, including the tank-sizing factor. The propellant mass is 165 kg less than is shown in Table 3-2, which was computed without taking into account the launch vehicle capability. This reduction in propellant results in a slightly negative EOM ΔV margin, as shown in Table 3-3, necessitating some combination of mission design changes. Examples of such potential changes include reducing the number of Titan flybys, extending the initial orbit period after SOI, and reducing the magnitude of the SOI burn offset.

As the table shows, extending the flight time results in improved margins, although the benefit per unit increase in flight time is rather small. This flight time flexibility could be used to offset future spacecraft mass increases or other factors that might detract from mission performance. For example, the trajectory design includes a maneuver of over 400 m/s at a heliocentric distance of about 0.85 AU, or five months after the final Venus swingby. Thermal constraints may require that this maneuver be delayed or done in smaller segments, which would result in a performance penalty. Such details have not yet been addressed, and are likely to have an effect on the performance data shown in Table 3-3.

Table 3-3. Alternate Mission Duration and ΔV Margin Summary (1996 VVVJGA)

Cruise Duration (yrs)	Total Mission Duration (yrs)	Arrival Date	EOM ΔV Margin (m/s)
9.4	13.4	7/05	-70
9.8	13.8	12/05	7
10.2	14.2	5/06	68

3.3.2.1 Mission Science Return. The main difference between the baseline primary mission and the alternate VVVJGA mission lies in the nature of the interplanetary trajectories. It is likely that the Saturn-encounter phases are qualitatively very similar. Detailed orbital tour designs would have to be developed for both missions before an accurate comparison could be made. However, some general comments on likely qualitative differences between the two missions are provided below.

To a first order, Saturn system science return from the baseline and alternate missions is probably similar. The most obvious difference stems from the later arrival date of the alternate mission (July 2005, compared to June 2004 for the baseline). During the first decade of the 21st century, Saturn's rings as seen from Earth will “close” rapidly until about 2010, when they will be seen nearly edge-on. Lower ring surface brightness will result from the decreased solar illumination, making optical observations of the rings more difficult. Radio science occultation observations of the rings must also penetrate through longer slant paths in the rings, rendering measurements in denser portions of the rings difficult or impossible. These problems will affect the baseline primary mission as well, but will be worsened by the later arrival date of the alternate mission.

A second difference is due to the longer mission duration, which means that power output from the RTGs will be smaller than that available for the baseline primary mission, making simultaneous instrument observations by several instruments difficult. Also, as mentioned previously, the negative EOM ΔV margin for the July 2005 arrival date would necessitate some minor mission reduction, which would have a corresponding impact on mission science return. Alternatively, a longer flight time could be used, which would exacerbate the ring closing problem. Further analysis would be required in order to select a minimal impact ΔV reduction strategy.

Finally, the earlier launch date (3/96 instead of 10/97) would necessitate accelerating the spacecraft's development. It is likely that many of the instruments would have some difficulty meeting this schedule. As a result, some instruments might have to be dropped or descoped. Although difficult to quantify at this time, it is probable that the accelerated schedule would have a significant impact on science return.

3.3.2.2 Operational Reliability. The alternate mission probably carries with it an elevated risk of mission failure. This is due to the longer cruise time, which increases the possibility of spacecraft malfunction before arrival at Saturn. Similarly, the increased mission duration of 13.4 years implies a greater possibility of malfunction before the end of the mission. These increased risks have not been quantified.

3.3.2.3 Backup to the Alternate Mission. The most serious element of increased mission risk associated with the March 1996 alternative is the lack of a timely non-EGA backup mission. As shown in Figure 3-1, aside from the VVVJGA, the March 2001 VVVGA is the non-EGA trajectory most nearly compatible with the performance of the Titan IV (SRMU)/Centaur. This is the same mission that is discussed below as a second alternate mission (see Subsection 3.3.3). All of the drawbacks associated with the March 2001 VVVGA as an alternate primary would also apply, of course, if it were considered as a backup. None of the other potential alternative non-EGA missions shown in Figure 3-1 can be considered Titan IV (SRMU)/Centaur-enabled backups to the VVVJGA, due to their high injected mass requirements. By contrast, there are a number of VEEGA trajectories (all VEEGAs contain two Earth swingbys) available to the baseline launch system, including opportunities in December 1997 and March 1999.

3.3.2.4 Program Cost. The VVVJGA mission identified above would launch in March 1996, more than 1-1/2 years earlier than the baseline mission, and would arrive at Saturn in July 2005, more than a year later than the baseline. Accelerating the spacecraft's development consistent with this earlier launch date would require the support of the European Space Agency (ESA) and the Italian Space Agency (ASI). The spacecraft would also have less testing time, which would increase development risk. Accelerated spacecraft hardware deliveries would result in contract cost increases; these cost increases have not been quantified. The Cassini Project has concluded that the earlier launch would not reduce development costs below the total development costs for the 1997 mission. In particular, significant funding increases in FY95 through FY96 would be required. Also, the 9.4-year flight time (compared to 6.7 years for the baseline missions) would increase mission operations costs.

3.3.3 March 2001 VVVGA Alternate Mission

The VVVGA trajectory identified as a potential alternate to the baseline primary launches in March 2001 and arrives at Saturn about 10.3 years later. Preliminary analysis has shown that the launch period occupies about the first 2-1/2 weeks of March 2001. The first Venus swingby occurs in August 2001 and places the spacecraft on course for the second Venus swingby in late September 2002, somewhat less than two Venus years later. The first Venus swingby occurs with a V_∞ of about 8.3 km/s, while the second, owing to the effect of a ~500 m/s maneuver between the two swingbys, is at about 11.3 km/s. The second Venus swingby is powered, with a maneuver near Venus of just over 300 m/s. A deep-space maneuver near the following aphelion results in a V_∞ of about 14.0 km/s at the third Venus swingby in late November 2005. This final gravity assist augments the heliocentric energy sufficiently to reach Saturn with no further maneuvers. Saturn arrival is in June 2011. The Saturn encounter phase follows the same sequence as the baseline mission: Probe release and data relay occur near the end of the initial orbit, and the Orbiter then continues on the orbital tour until about June 2015.

As described in Subsection 3.3.2 for the VVVJGA mission, a realistic performance estimate for the Titan IV (SRMU)/Centaur for the 2001 VVVGA mission is used to calculate the maximum propellant load achievable for the spacecraft and the resulting EOM ΔV. As before, no injection margin is held beyond the recommended reserves of 875 kg for the launch vehicle (performance quotation from LeRC, November 1992). Also, these calculations assume the same dry mass allocations described in Note 2 of Table 3-2, including the tank-sizing factor. The propellant load corresponding to the trajectory with a cruise duration (i.e., flight time) of 10.3 years is limited to 3670 kg. This is 980 kg less than that shown in Table 3-2, which was computed without taking into account the launch vehicle capability. The resulting EOM ΔV margin is -417 m/s, as shown in Table 3-4. This table includes results for trajectories with other cruise durations as well.

All of the end-of-mission delta-V margins are negative. The spacecraft's propellant would be depleted before Cassini's complete science objectives and investigations are altogether fulfilled. As Table 3-4 shows, flight time changes have a small effect on the margins, with the best value still negative at -410 m/s for an 11.3-year flight time. These facts emphasize the marginal nature of this mission as a potential alternative to the baseline.

While this analysis is preliminary, and further optimization of the trajectory could result in a slight improvement in mission performance, the performance results clearly indicate that a combination of measures would have to be taken to reduce mission delta-V requirements to make this a usable mission. For the 10.3-year flight time mission, the satellite tour might be reduced from 35 to 21 Titan flybys (an established minimum requirement; see JPL, 1992b, Subsection 4.1.5), the Saturn orbit insertion burn offset eliminated, the Initial Orbit Period (IOP) extended to 232 days, and the more powerful rhenium engine utilized. This combination, while not necessarily the best one, would raise the EOM ΔV margin from about -417 m/s to approximately zero. With such reductions, the 2001 VVVGA mission might be viable with SRMU performance. In addition to the reduced science return, the mission would have little resiliency in the face of unexpected spacecraft mass increases or launch vehicle performance degradations. By comparison, the baseline mission has considerable resiliency, and if it were necessary the SRM could be accommodated on the launch vehicle instead of the SRMU. As shown earlier, the March 2001 VVVGA alternative requires the SRMU.

Table 3-4. March 2001 Alternate Mission Performance Summary

Cruise Duration (yrs)	Total Mission Duration (yrs)	Arrival Date	EOM delta-V Margin (m/s)
9.8	13.8	12/2010	-493
10.3	14.3	6/2011	-417
10.8	14.8	12/2011	-396
11.3	15.3	6/2012	-410
11.8	15.7	12/2012	-462

Note: EOM ΔV values were computed for an optimum launch date. To compensate for the penalty of including an 18 day launch period, 100 m/s was subtracted from the optimum value.

3.3.3.1 Mission Science Return. The main difference in science return between the baseline primary mission and the March 2001 alternate mission would result from the measures that would have to be taken to sufficiently reduce the ΔV to meet the launch vehicle capability. Then detailed orbital tour designs would have to be developed for both missions before an accurate comparison could be made. However, some general comments on likely qualitative differences between the two missions are provided below.

Due to the necessity of reducing ΔV requirements, there would be significant degradations in science return in this mission phase, although the resulting mission would likely remain within accepted minimum criteria. Examples described earlier for the 10.3-year flight time mission are: (1) reducing the number of Titan flybys from 35 to 21, (2) eliminating the SOI burn offset (thus giving up much of the close-in ring science), and (3) increasing the IOP from 152 to 232 days. Additionally, the remarks in Subsection 3.3.2.1 concerning the “closing” of Saturn's rings as seen from the Earth apply with greater force for the March 2001 mission (arrival in June 2011), when the rings would be seen nearly edge-on. After arrival, however, the rings would begin to open up to the point where the latter stages of the mission will once again have ring science comparable to the end of the primary mission. Finally, the later launch and longer mission duration means that the power output from the RTGs will be smaller than that available for the baseline primary mission.

The possible science impacts of switching to the March 2001 alternate mission can be summarized as follows:

- Minimal impact for interplanetary phase;
- A significant general degradation of science return in the satellite tour phase due to a reduced number of Titan flybys;
- RTG power would be down 60-80 W due to late arrival date, resulting in reduced observation time per instrument;
- Poorer quality ring observations, both optical and radio.

3.3.3.2 Operational Reliability. The March 2001 alternate mission carries with it an elevated risk of mission failure. The fact that the launch is not until 2001 means that the spacecraft would be in storage for three and a half years. With a flight time of greater than 10 years, the spacecraft would be almost 14 years old before the primary science phase could begin. This could increase the possibility of a spacecraft malfunction. This increased risk has not been quantified.

3.3.3.3 Backup to the Alternate Mission. At this time, after a reasonable search for opportunities through 2004, only one potential non-EGA backup to the March 2001 VVVGA alternate primary has been identified. This potential alternative is a March 2002 VVVGA opportunity that possibly could satisfy the minimum science return criterion if Cassini was launched using an Energia-M with a Centaur IIA upper stage. Use of an Energia-M as a launch vehicle is addressed in Appendix A, Subsection A.2.3.3. The 2002 alternative would also require for the spacecraft an increased propellant load of about 1200 kg and use of the non-space qualified rhenium engine. More detailed study would be required to determine whether the larger propellant tanks would be compatible with the Cassini spacecraft, launch vehicle payload fairing, and upper stage configuration. Additionally, there exist EGA opportunities that could be used as backups to the non-EGA March 2001 primary alternative. These opportunities would occur in mid to late 2002. If additional analysis should discover any such potential backups, then all the reliability problems mentioned above for the alternate primary itself would apply with even stronger force to the backup (assuming the trajectory would result in a later Saturn arrival date compared to the primary). Similarly, the power from the RTGs would have decreased even further with the additional delay. Analysis would have to be done to determine whether the remaining power was sufficient to perform the minimum science.

3.3.3.4 Program Cost. The delayed launch (more than three years) and greatly extended duration (10.3 years) of this mission alternative would have a significant impact on overall mission operations costs. Additional costs would also include storage costs and additional testing costs which would occur after removing the spacecraft from storage.

3.4 SENSITIVITY ANALYSIS: ADDITIONAL MEASURES FOR ENABLING CASSINI MISSION ALTERNATIVES AND HOW THEY IMPACT MISSION SCIENCE OBJECTIVES

As discussed in Subsection 3.3.2.3, the only marginally feasible, non-EGA backup mission identified for the March 1996 VVVJGA alternate mission is the March 2001 VVVGA, which is itself a possible second alternate mission. While none of the science would be significantly impacted by employing the March 1996 alternate, the March 2001 alternative is clearly not as attractive, due to its late launch date and 10-year flight time.

However, as suggested in Subsection 2.2.1.3B, it might be conceivable to pursue an earlier backup trajectory with a shorter flight time if major sacrifices were made in the mission science objectives. For demonstration purposes only, two such sacrifices are: (1) reducing the length of the Saturn tour and (2) deleting the Huygens probe.

3.4.1 Reducing the Length of the Saturn Tour

Various possible reductions in the Saturn tour would reduce mission propellant requirements. The resulting decrease in injected mass could, if substantive enough, enable additional alternate trajectories. Possible approaches to reducing the tour include:

- Reducing the number of Titan swingbys;
- Eliminating other icy satellite swingbys;
- Reducing Titan radar coverage;
- Eliminating the high inclination tour of Saturn's rings and high-latitude magnetosphere.

These delta-V reduction measures would conflict with virtually all of the Saturn-related science objectives. For instance, a large reduction in the number of Titan swingbys would severely constrain the degree to which the time variability of Titan's clouds and hazes could be observed, the structure and chemical composition of its atmosphere analyzed, the exchange and deposition of energy within its atmosphere measured, and its surface mapped. The elimination of icy satellite swingbys would render determination of the surface composition and geological history of Saturn's icy satellites virtually impossible, and it would clearly impede determination of the nature and origin of the dark material located throughout the Saturnian ring and satellite system. The reduction of Titan radar coverage would preclude detailed mapping of Titan's cloud-enshrouded surface, and the elimination of the high inclination tour of Saturn's rings and polar magnetosphere would prevent a determination of the three-dimensional structure and dynamic behavior of Saturn's rings and magnetosphere.

Among candidate alternate trajectories that were ruled out in Section 3.2, the one that comes closest to feasibility is the December 1997 VVVGA. The following changes to the mission plan were first evaluated in an attempt to make this mission feasible:

(1) Increasing the period of the initial orbit after SOI to 232 days (baseline is 152 days), thus delaying the Huygens Probe mission and the start of the tour;

(2) Performing the SOI burn at the optimum time, rather than offsetting it slightly to acquire ring science data; and

(3) Reducing the number of Titan flybys during the tour from the nominal number of 35 to the minimum of 21.

Lengthening the flight time was not included because the result would violate the Project policy that the second Titan flyby be accomplished within 12 years of launch.

With the above measures, the mission was still infeasible. Even if it were assumed that the Project could take increased risk by reducing the launch vehicle reserves by half, from 875 kg to 438 kg, the EOM ΔV was still short by 125 m/s.

The only way to raise that negative ΔV to zero would be to take rather drastic measures, such as limiting the tour to only nine Titan flybys. A delta-V reduction of this magnitude would clearly have a major impact on the science return; all Cassini tour objectives would be seriously affected. The resulting mission science return would begin to resemble the science yield that might be expected from a swingby mission; an objective which has already been accomplished with the Voyager spacecraft.

3.4.2 Deletion of the Huygens Probe[1]

Another drastic measure for enabling alternate trajectories would be deleting the Huygens probe, which would save 352 kg (775 lb) of dry mass. For the March 2001 VVVGA every kilogram of dry mass on the Probe necessitates nearly one kilogram of propellant mass. Therefore, the overall mass savings resulting from eliminating the Probe (i.e., for this trajectory) would be on the order of 600 kg

[1] Note: this exercise was done only for demonstration purposes.

(1320 lb). This substantial mass reduction would translate into a considerable reduction in injected mass, which could enable other non-EGA trajectories.

However, deletion of the Huygens probe would seriously damage the NASA/ESA working relationship. In fact, if this action were taken, other European contributions to Cassini would be at risk of being withdrawn, jeopardizing the entire Cassini mission. Even if the mission survived, the loss of the Probe would sacrifice the heart of two of the five Titan-specific mission science objectives (see Volume 1, Table 3-1): (1) determination of the structure and chemical composition of Titan's atmosphere and (2) measurement of the exchange and deposition of energy within its atmosphere. The accomplishment of these objectives is key to answering questions pertinent to understanding the origins of life, such as, “What chemical processes produced the atmosphere of hydrocarbons and other organic molecules unique to Titan?” and “Do these hydrocarbons exist in liquid form on Titan’s surface?” Sacrificing the above objectives would mean foregoing the possibility of answering these questions.

SECTION 4

SPACECRAFT POWER ALTERNATIVES

This section summarizes the process of selecting alternatives to the baseline Cassini spacecraft power design—plutonium-fueled RTGs providing electrical energy, and RHUs providing heat for the spacecraft and science payload. (RHUs are used with the all-RTG design because they reduce the total amount of plutonium required on the spacecraft. It would be extremely inefficient to carry additional plutonium for the production of heat by the RTGs, which would in turn generate electricity, which would then be converted back to heat.) Subsection 4.1 surveys the potential alternatives. Subsections 4.2, 4.3, and 4.4 focus on specific solar designs, which are given a more detailed treatment in Appendices D, E, and F, respectively.

The study's approach to evaluating alternatives, as detailed in Section 2, consists of: (1) identifying alternative power technologies that enable fulfillment of the Cassini mission's science goals and objectives; (2) characterizing each technology's physical and performance parameters to enable comparison with the baseline; and (3) comparing the alternative power technology with the baseline in terms of scientific and programmatic factors pertinent to Program decision makers.

Among a wide range of potential power alternatives considered here, only solar power warrants detailed consideration. After further examination, this assessment concludes that an all-solar Cassini must be deemed infeasible at this time because: (1) no U.S. launch vehicles exist to launch the large mass of even the lightest solar configuration required, and (2) even if a heavy lift booster and suitable upper stages were available, severe limitations on spacecraft motion, instrument field-of-view, and programmatic risk would make this option scientifically untenable.

4.1 IDENTIFYING FEASIBLE POWER ALTERNATIVES

The study team adopted the approach that a meaningful power alternative is one that reduces or eliminates the risk of plutonium dioxide release. In principle, this could be accomplished in four ways:

- Substitute a less potentially hazardous radioisotope for the Pu-238 currently used in the RTGs.
- Develop RTG designs that use less Pu-238.
- Use a nuclear reactor instead of RTGs.
- Replace the RTGs with a non-nuclear power source.

Several isotopes were investigated by the Department of Energy (DOE) as alternatives to Pu-238, and two were considered potential substitutes; DOE's analysis appears in Subsection 4.1.1 and as Appendix B. The feasibility of using nuclear reactors for Cassini is examined in Subsection 4.1.3. Subsections 4.1.2 and

4.1.4 examine both alternative conversion technologies for radioisotope heat sources, as well as other alternative energy sources. Various power technologies that are presently being—or potentially could be—used in spaceflight were surveyed, and the key technical and design issues associated with their use for Cassini are summarized.

4.1.1 Non-Plutonium RTGs

Principal concerns raised in the past regarding the use of RTGs were based on the potential for release of respirable-sized particles of plutonium fuel in the event of an accident. The assessment of the human and environmental risks associated with those RTGs used on recent missions can be found in U.S. Department of Energy, 1989a and 1990b. Several isotopes were investigated as alternatives to Pu-238 by DOE; their analysis appears as Appendix B.

In principle, any radioisotope with an appropriate half-life could be used. Radioisotopes other than Pu-238 have been used in RTGs for ground-based power applications, although none have ever been used for U.S. space missions. In general, the isotope must have a half-life long enough to assure a sufficient quantity of heat throughout the mission and a sufficiently high power density (power/unit mass or volume of isotope) to provide the required power with a suitably small generator. Also, the radiation levels produced by the isotope must be low enough not to interfere with the science instruments, damage spacecraft electronics, significantly complicate installation procedures, and/or significantly increase risk to ground personnel working in proximity to the RTG.

As detailed in Appendix B, DOE identified strontium-90 (Sr-90) and curium-244 (Cm-244) as possible alternatives to Pu-238. Each isotope has a half-life less than one-third that of Pu-238, and a variety of other characteristics that offer advantages and disadvantages compared to Pu-238.

4.1.1.1 <u>Strontium-90</u>. Sr-90 is a beta emitter with a 28-year half-life and produces, in fuel form (SrO), approximately 0.007 thermal watts (W_t) per Curie (Ci) of activity[1] (about 2.1% of the Pu-238 fuel form's heat production). Its lower capacity for heat production and shorter half-life means that, to produce the desired electrical power at the end of a 15-year mission, the amount of Sr-90 fuel used for each RTG at the beginning of the mission must be increased over the amount of Pu-238 fuel that would be required. This amounts to a total Sr-90 activity per RTG of 8.5×10^5 curies, 6.2 times greater than that of the Pu-238 RTG fuel activity. In addition, the fuel would be 22.5% heavier and take up 2.65 times the volume of the Pu-238 fuel, causing the Sr-90 RTG to be larger and heavier than a Pu-238 RTG. The Sr-90 RTG would also produce a dose rate of approximately 10,000 rem/hour at one meter from the RTG center line, about two million times that of the Pu-238 RTG dose rate at 1 meter (approximately 5×10^{-3} rem/hour). This increase in dose rate would require extensive shielding during production, shipment, and ground handling, as well as requiring massive spacecraft shielding to protect sensitive electronics.

[1] The activity of a quantity of radioactive nuclide is the quotient of the number of nuclear transformations occurring in this quantity during a given time period (1 Curie of activity = 3.7×10^{10} transformations/sec).

4.1.1.2 Curium-244. Cm-244 is an alpha emitter with an 18.1-year half-life and produces, in fuel form (Cm_2O_3), around 0.34 W_t per Ci of activity (roughly equivalent to the Pu-238 fuel form's heat production). The shorter half-life of the material means that, to produce the desired electrical power at the end of a 15-year mission, the amount of Cm-244 fuel used for each RTG at the beginning of the mission must be increased over the amount of Pu-238 fuel that would be required. This amounts to a total Cm-244 activity per RTG of 2.06 x 10^5 curies (1.5 times that of the Pu-238 RTG fuel activity). However, the fuel would be 26% as heavy and take up only 24% of the volume of the Pu-238 fuel. The Cm-244 RTG would produce a gamma dose rate of about 0.9 rem/hour at 1 meter from the RTG center line, about 180 times that of the Pu-238 RTG dose rate at 1 meter (about 5 x 10^{-3} rem/hour), as well as a fast neutron flux of approximately 116,000 n/second-cm^2, about 450 times that of the Pu-238 RTG fast neutron flux. The increase in gamma dose rate and neutron flux would also require extensive shielding during production, shipment, and ground handling, as well as massive spacecraft component shielding to protect sensitive electronics.

The United States lacks processing facilities for both Sr-90 and Cm-244. There is no capability to recover, process, and encapsulate large quantities of Sr-90 or Cm-244. It is doubtful that, even with immediate funding, sufficient capability could be ready in time for the Cassini mission. Moreover, as a means of reducing the level of environmental risk, the gamma and neutron emissions of these alternative radioisotopes are not likely to provide any significant advantages over the plutonium baseline. (given the lack of a clear environmental advantage, as well as the technical difficulties surrounding the production and application of non-plutonium RTGs, this option was pursued no further.

4.1.2 Higher Efficiency Power Conversion Systems Requiring Less Plutonium

From an environmental risk standpoint, if a mission can be accomplished using less plutonium, the potential environmental hazard associated with a launch accident is reduced. This concept of providing the same level of power with less plutonium was addressed by evaluating alternative energy conversion technologies with higher efficiencies than the thermoelectric converters currently used in RTGs. Essentially, all electrical generating systems can be viewed as a combination of an energy source and an energy conversion system. The energy source may be a beam of photons from the Sun or from an artificial source such as a laser, chemicals in batteries or fuel cells, or heat from combustion, radioisotope decay, or fission sources. The higher the efficiency of a conversion technology, the lower the amount of plutonium required to generate the same electrical power output. Subsection 4.1.4.1 and Appendix C evaluate alternative conversion technologies potentially compatible with radioisotopes in general and the GPHS in particular. All of the alternatives to thermoelectric conversion exhibit serious technology issues that cannot be resolved in a time frame compatible with the mission baseline. In view of these technology issues, the study has eliminated these higher conversion efficiency options from further consideration.

4.1.3 Nuclear Reactors

From an environmental risk standpoint, nuclear reactors offer an advantage in that they can be launched “cold” (i.e., in a non-operating mode). Prior to reactor operation, the inventory of environmentally detrimental fission by-products is very small.

Reactors of a size and operating lifetime suitable for Cassini, however, do not exist nor are they presently under development in the U.S. The relative immaturity of the technology and the long estimated lead time for development, if such were undertaken, mean that a reactor power source would present unacceptably high levels of technical and schedule risk to the Project. There are also a number of technical issues yet to be resolved with space reactor designs, even though reactors have been launched and operated in space since 1965 (almost exclusively by the Soviet Union). Space reactors (both U.S. and Soviet) of these earlier designs so far have demonstrated only modest lifetimes in space, namely, up to about one year. Control complexity and mass, the latter both inherent in the reactor and due to required shielding, are additional challenges to reactor development and implementation for deep space, long-duration missions. For all of these reasons, reactors do not constitute a feasible power alternative for Cassini.

4.1.4 Feasibility Assessment of Non-Nuclear Power Sources

A wide range of power-generating technologies exist that use energy sources other than the heat provided by radioisotopes. As such, all of these would eliminate the concerns regarding a launch or reentry, accident-induced release of radioactive materials into the environment.

Identification of non-nuclear power generation alternatives requires determining which technologies are mature enough to be applied to the spacecraft in the baseline time frame. Also, each of the candidates must be consistent with the Cassini power level and operating requirements in order to qualify as legitimate alternatives to the baseline. The feasibility of these non-nuclear technologies is addressed in the following section.

4.1.4.1 <u>Power Technologies Addressed in the Study.</u> A number of energy conversion subsystem concepts have been developed for converting energy from the source into electrical energy. (Only certain conversion approaches are compatible with a given energy source.) The range of energy sources and energy conversion technologies considered is shown in Table 4-1. Each column represents a current or potential near-term energy source and each row represents a current or potential near-term conversion technology. An “X” has been placed in each element of the matrix to indicate which energy sources and conversion technologies might be compatible for the Cassini mission (only certain combinations of energy sources and energy conversion subsystems are inherently compatible).

Table 4-1. Alternative Power Systems Considered

Conversion Technologies	Energy Sources: Radio-isotope	Solar Non-Conc./Conc.	Reactor	Fuels and Chemicals	Power Microwave/Laser
STATIC:					
Photovoltaic		X/X			/X
Rectenna					X/
Thermoelectric (TE)	X	/X	X	X	
Thermionic	X	/X	X	X	
Alkali Metal Thermo-electric Converter (AMTEC)	X	/X	X	X	
ThermoPhotovoltaic (TPV)	X	/X	X	X	
Fuel Cells				X	
Primary Battery				X	
DYNAMIC:					
Rankine	X	/X	X	X	
Brayton	X	/X	X	X	
Stirling	X	/X	X	X	
Magneto-Hydrodynamic (MHD)				X	
Internal Combustion				X	

“X” denotes compatible source/conversion technology pairs

4.1.4.2 Feasible Technologies. The Table 4-1 technologies are examined in greater detail in Appendix C. Of the non-nuclear power technologies investigated, only non-concentrating solar photovoltaic arrays, low-concentration solar photovoltaic arrays, fuel cells, and primary batteries are sufficiently well-developed to warrant consideration as candidate power technologies for near-term spaceflight applications. This list is further shortened by determining which of these remaining candidates (solar arrays, fuel cells, or batteries) are compatible with the power and operational needs specific to the planned Cassini mission. Their possible applicability can be summarized as follows:

A. Non-concentrating solar arrays: Possibly applicable

A proven technology that has been applied on a range of spacecraft, including some involving missions of long duration.

B. Concentrating solar arrays: Possibly applicable

Space-qualified concentrators have not been demonstrated nor are they on a firm development schedule. Plans exist to conduct a brief STS spaceflight experiment (early in FY96) using an inflatable paraboloid antenna: this will demonstrate the inflatable paraboloid antenna's deployment and surface accuracy. This will not demonstrate a functioning solar concentrator over the required operating lifetime. Non-concentrating solar arrays are a proven technology that could be enhanced with low-concentration reflectors.

C. Fuel cells: Not applicable

Space-qualified fuel cells have not demonstrated the required lifetime over the planned duration of the mission, and the fuel needed would have an unacceptable mass impact.

D. Batteries: Not applicable

As discussed in Appendix C, the mass of primary (non-rechargeable) batteries or fuel cells needed to power the Cassini spacecraft over the entire mission duration is incompatible with launch vehicle capabilities. Solar array and secondary (rechargeable) battery combinations are discussed in Appendixes D, E, and F, and generally entail prohibitive mass and/or unacceptable science return.

Since neither primary nor secondary batteries prove suitable for powering the entire spacecraft over the mission duration, the next logical question is whether or not some combination of batteries and RTGs is capable of doing so. Another possible option, one that was considered in the early stages of the Cassini spacecraft design, is to replace one of the RTGs with secondary cells that would be periodically recharged by the remaining RTGs. However, analysis of the subsystem power requirements has shown that the power demand from Saturn orbital insertion to the end of the mission exceeds what the RTGs can deliver without even accounting for the power required to recharge the secondary batteries. Hence, the only way to attempt the mission with secondary cells would be to periodically shut down the spacecraft, let the batteries recharge, and then restart the spacecraft. Not only would such a procedure be inconsistent with the science objectives, it would entail extremely high-risk wake-ups from dormant modes that have not yet been demonstrated for spacecraft.

Hence, non-concentrating and low-concentration solar arrays are the only non-nuclear power systems technology that were not eliminated in the Appendix C evaluation, thus warranting further examination in Appendices D, E, and F.

4.2 APPLICABILITY OF SOLAR POWER TO CASSINI

This subsection, together with the following two subsections, summarize the solar Cassini design studies that are detailed in Appendices D, E, and F. They examine the applicability of solar power to the Cassini mission and, where solar power is applicable, characterize the implications of solar power on spacecraft design and performance.

Extensive studies were made of two point designs intended to bound the problem of science return as a function of spacecraft mass. This subsection and Appendix D present a spacecraft design that attempts to maintain maximum science performance at the expense of a large size and mass. The original all-solar design is based on, and is compared to, the original Cassini baseline design. Both the nominal all-solar design and the pre-redesign Cassini have a strong emphasis on science return. They carry articulated scan platforms that allow the science instruments to be pointed in a desired direction without having to turn the spacecraft. This increases the amount and accuracy of data acquired. The scan platforms, however, require deployable booms and platform actuators, which draw power and add mass.

To facilitate sensitivity analysis, a second, reduced science solar point design is considered in Subsection 4.4 and Appendix F. The second study reduces science performance in an attempt to fit an all-solar Cassini spacecraft within Titan IV (SRMU)/Centaur mass constraints. The reduced science all-solar design is based on, and is compared to, the current baseline Cassini, also called the redesigned Cassini. These designs have the instruments fixed to the body of the spacecraft, thus necessitating turning the entire spacecraft to point the instruments in a particular direction. This procedure both slows down the amount of data acquired and makes it less accurate.

Each all-solar design is compared to its corresponding baseline Cassini design, and mass differences are calculated. The mass differences reported for the two solar designs are thus not directly comparable to each other. Between these two design points lies a broad spectrum of mission opportunities and spacecraft configurations which yield varying levels of science return at spacecraft masses and sizes between the two design points. Neither of the two point designs is shown to be a feasible alternative; therefore, neither are any of the designs in between.

In addition, some other modifications of the original all-solar design and hybrids with other sources of electrical power are discussed in Subsection 4.3 and Appendix E.

4.2.1 Approach

To assess the applicability of solar power to Cassini, the following approach was employed:

- Calculation of the required solar array area. The area depends on the output power needed by the spacecraft and science instruments (including the battery recharge energy), the sunlight-to-electricity conversion efficiency of the arrays, distance from the Sun, orientation relative to the Sun, and any adjustments made to compensate for array performance degradation over time.

- Definition of an array configuration appropriate to the spacecraft, and identification of other design changes (e.g., batteries and reaction wheels). Once identified, the design changes can be used to determine whether basic launch vehicle constraints on spacecraft mass or volume can be met.

- Identification of any other differences between the solar-powered spacecraft and the RTG/RHU baseline regarding specific physical and operational characteristics that could affect the spacecraft's capability to meet science objectives and other programmatic goals.

4.2.2 Array Sizing

4.2.2.1 Mission Power Requirements. Throughout the original study, baseline Program science objectives were assumed for the mission. The solar power alternative would need to supply the same minimum level of power during critical portions of the missions as that supplied by the baseline RTG power source, plus waste heat from the RTGs used in the spacecraft, plus the amount required to supplant the RHUs with electrical heaters. In order to calculate the size of the arrays required to supply the minimum power levels (and thereby allow calculation of their mass, operability, cost, etc.), the baseline power demand, operating distance from the Sun, efficiency of the arrays, and several other factors had to be specified. The first two factors were derived from baseline mission requirements. The design points for the mission were established as follows.

The period of largest power demand for Cassini will be during the tour of the Saturnian system, when Saturn is at a distance of between 9.0 and 9.3 AU from the Sun. (While the Orbiter is expected to continue on with Saturn, out to its greatest distance of 10.1 AU from the Sun, this point is past the end of the planned mission.) Array design must, therefore, satisfy end-of-mission demands, at 9.3 AU, when power requirements are high, insolation is at a minimum, and array degradation is at a maximum.

Replacing the electrical power output of the RTGs and the thermal output of the RHUs would require an output of 837 W from the solar panel at Saturn (see Appendix D.2.3.1). This power level constitutes the mission-specific basis for sizing the Cassini solar arrays. This basis for array sizing is somewhat optimistic, in that it does not account for the additional power required to replace the heat radiated to the propulsion module by the RTGs (i.e., array size would increase if waste RTG heat were included).

The amount of sunlight falling on a spacecraft decreases rapidly as it moves away from the Sun. The intensity of sunlight decreases as $1/r^2$, where r is the distance to the Sun. At Saturn the Sun appears to be dim because such a small fraction of its light reaches the spacecraft. At its closest approach, Saturn is nine times as far away from the Sun as the Earth, and the sunlight it receives is only about 1% of that at Earth. The spacecraft's solar arrays would have to be extremely large to gather enough energy from this small, dim, distant Sun. Figure 4-1 illustrates how the collector area must increase in order to offset the $1/r^2$ decrease in sunlight. The filled circles show the $1/r^2$ decrease in insolation; the other line shows a corresponding r^2 increase in the area of the solar arrays. This figure understates the size of the arrays both because they are not as efficient at lower intensities (see the discussion of LILT below), and also because they suffer degradation in the decade-long trip from Earth to Saturn.

4.2.2.2 <u>Additional Sizing Factors.</u> In addition to power and operating distance from the Sun, several other factors must be specified that will affect subsequent array sizing calculations.

A. Array Conversion Efficiency

The efficiency at which available sunlight is converted to electrical power is primarily a function of the type of photovoltaic technology used, and the conditions in which it operates—namely, the array temperature and the insolation level.

Two photovoltaic technologies were considered mature enough to warrant evaluation in this study—the silicon (Si) and gallium arsenide active layer on germanium substrate (GaAs/Ge) solar cells used in an Advanced Photovoltaic Solar Array (APSA). At Earth (1 AU) at the end of ten years, an Si-APSA system would be expected to provide a power output of 93 W/m^2 (watts per square meter), for a 6.9% end-of-life (EOL) efficiency assuming a 10.2% beginning-of-life (BOL) efficiency, based on total array area. For GaAs/Ge-APSA, the corresponding figure would be 143 W/m^2, for a 10.6% EOL efficiency assuming a 15.7% BOL efficiency. (Cell efficiencies at standard test conditions and before assembly are 13.8% for Si and 18% for GaAs/Ge.) These values must be adjusted to reflect the reduction in actual insolation experienced at the average operating distance for the spacecraft at Saturn (9.3 AU vs. 1 AU), as well as the increase in actual operating lifetime required.

B. Radiation Design Margin

A radiation design margin (RDM) of two (minimum) is applied by the Cassini Project to ensure the protection of all electronic devices on the spacecraft. Consistent with this policy, an RDM is applied to the photovoltaic array by doubling the expected radiation fluence when calculating the allowance to be made for radiation-induced array degradation effects.

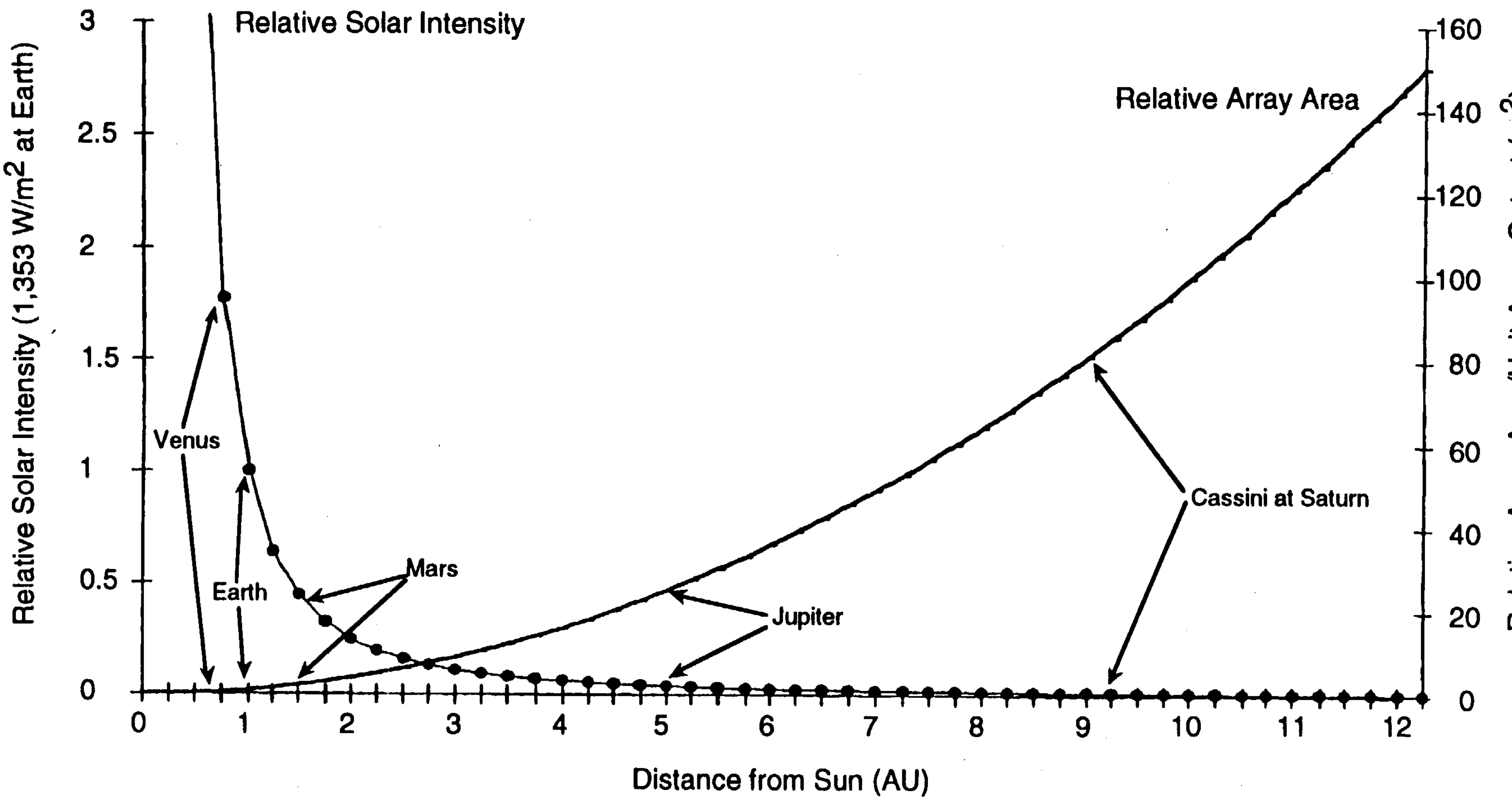


Figure 4-1. Decrease In Solar Intensity and Increase In Solar Array Area With Increased Distance From the Sun

C. Low Intensity, Low Temperature (LILT) Effects

Silicon solar cells have been observed to change their conversion efficiencies under conditions of low light levels and low temperature—conditions present in the regions where Cassini will operate. Ideally, the conversion efficiency of silicon solar cells will improve with reduced operating temperatures. However, a random amount of performance degradation has been observed. The impact of the degradation is exacerbated at low intensities, due to the low current output of the cell, mitigating much, or in some cases, all of the gains from the initial temperature reduction. The degradation impact is greater with increasing distance from source. There has only been limited laboratory data obtained for low intensity, low temperature cell performance, since no photovoltaic powered spacecraft have been sent beyond 2 AU. The LILT factor estimates used in calculating the solar array sizes were 0.72 for the Cassini silicon-APSA design (Appendix D.2.2.4) and 0.95 for the Cassini GaAs/Ge-APSA design (Appendix D.3.2.3). This means that LILT for silicon arrays tends to significantly penalize array performance at the Cassini tour distance; the array sizing needs to account for these effects.

For GaAs/Ge, negligible improvement or degradation in efficiency has been observed under LILT conditions. However, it should be noted that LILT measurements have been performed only for conditions simulating up to 5.2 AU. It is difficult to accurately reproduce conditions found in Saturn orbit using existing simulation methods/equipment.

Other factors that would degrade the output of the solar arrays are additional coatings placed on the arrays to reduce electrical charging effects, damage to the arrays from Saturn ring particle impacts, etc. These factors were estimated to require relatively small increases in the size of the arrays and therefore were not included in the sizing calculations.

4.2.3 Array Configuration and Initial Conclusions

This subsection summarizes the results of calculating the solar array area for each of the two solar cell options, defining an array configuration appropriate to each spacecraft, and then calculating the basic physical properties (i.e., mass and stowed volume). According to the approach discussed in Subsection 4.2.1, this information is then compared with the launch vehicle mass and volume constraints, allowing an initial assessment of the all-solar Cassini spacecraft feasibility.

4.2.3.1 <u>Required Array Configuration and Stowed Volume.</u> Tables 4-2a and 4-2b and Figures 4-2a and 4-2b summarize the required array area, critical envelope dimensions, and payload fairing parameters defined for both the silicon and gallium arsenide designs associated with Cassini. The results apply both to the articulated and nonarticulated array design concepts. Although the study (Appendix D) addressed both articulated and nonarticulated design approaches, neither was shown to have a clear advantage. Articulation simplified some operational aspects, but at the expense of increased spacecraft mass and design complexity. While some of the stow volume difficulties enumerated in Table 4-2b might be ameliorated through array redesign or use of a larger payload fairing, such measures would entail additional costs and integration difficulties not quantified in Appendix D.

4.2.3.2 Array Impact on Spacecraft Mass. Table 4-3 summarizes the net mass increases to the original spacecraft calculated for Cassini, including both Si- and GaAs/Ge-based APSA technologies.[2] The principal RTG-related mass elements that would be replaced by solar arrays include the RTGs, RTG support structure, and RHUs. Other mass change constituents are detailed in Appendix D.

The primary contributor to the mass increase would be the mass added by the solar arrays themselves and their support structure. The next major mass contributor would be the batteries included to provide backup power. The backup power would be necessary during periods of eclipse and during maneuvers that would cause the solar arrays to point away from the Sun. The eclipse periods in which the spacecraft will be in Satum's shadow are significant, and will determine the size of the batteries. For the tour, an eclipse period of up to 14.5 hours is representative and is used for this study.

[2] These mass deltas actually underestimate the impact of using solar arrays. An explanation for this underestimation appears in Appendix D.

Table 4-2a. Solar Array Dimensions*

Array Technology	Cassini (four wings) Wing Dimensions (width x length, m)	Total Area** (m^2)
Si-APSA	3.5 x 81.9	1152
GaAs/Ge-APSA	3.5 x 42.4	598

* Dimensions shown reflect non-articulated solar array areas
** Including attachment panels

Table 4-2b. Integration Aspects

Array Technology	APSA Critical Envelope Dimensions (m)(L x S)*	HGA Raise Distance (m)(H)**	Exceeds PLF Dynamic Envelope?	Fits 66 ft PLF?
Si-APSA	3.6 x 0.76	0.54	Yes	No
GaAs/Ge-APSA	2.4x 0.74	0.52	Yes	No

* Dimensions taken from Figure 4-2a
** Mast canister and sizing estimated per TRW, 1990, and modified for Cassini

Table 4-3. Net Mass Delta Impact: Solar Alternative vs. RTG Baseline (kg)

	Spacecraft/Array Technology (fixed array option)	
	Si*	GaAs/Ge**
Solar Array and Associated Hardware minus RTGs (baseline) and Associated Hardware	+2190 -272	+1609 -272
Net Mass Impact to Spacecraft	+1918	+1337

* Mass numbers summarized from Table D-3.
** Mass numbers summarized from Table D-8.

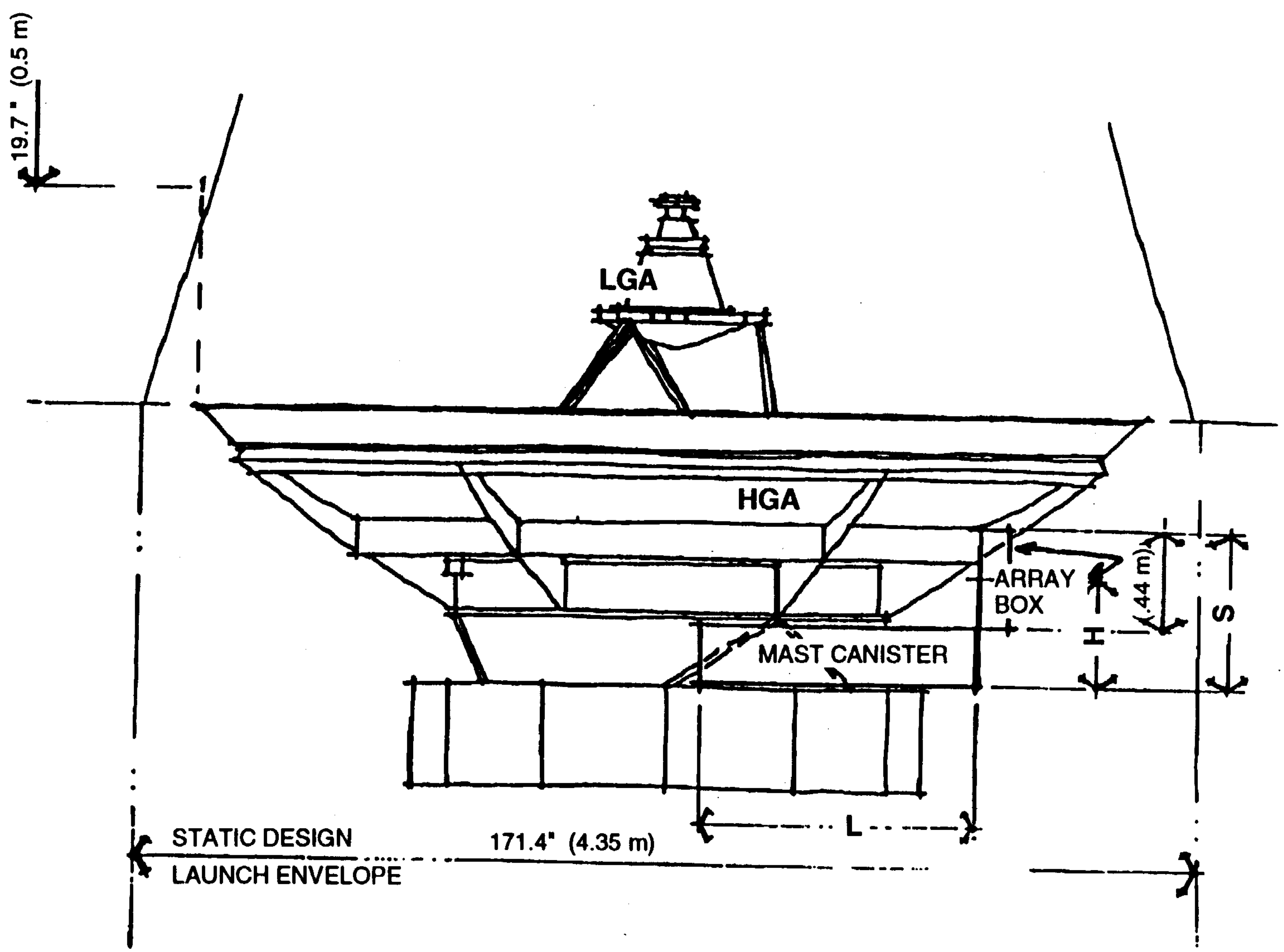


Figure 4-2a. Sketch of Stowed Solar Array Launch Configuration

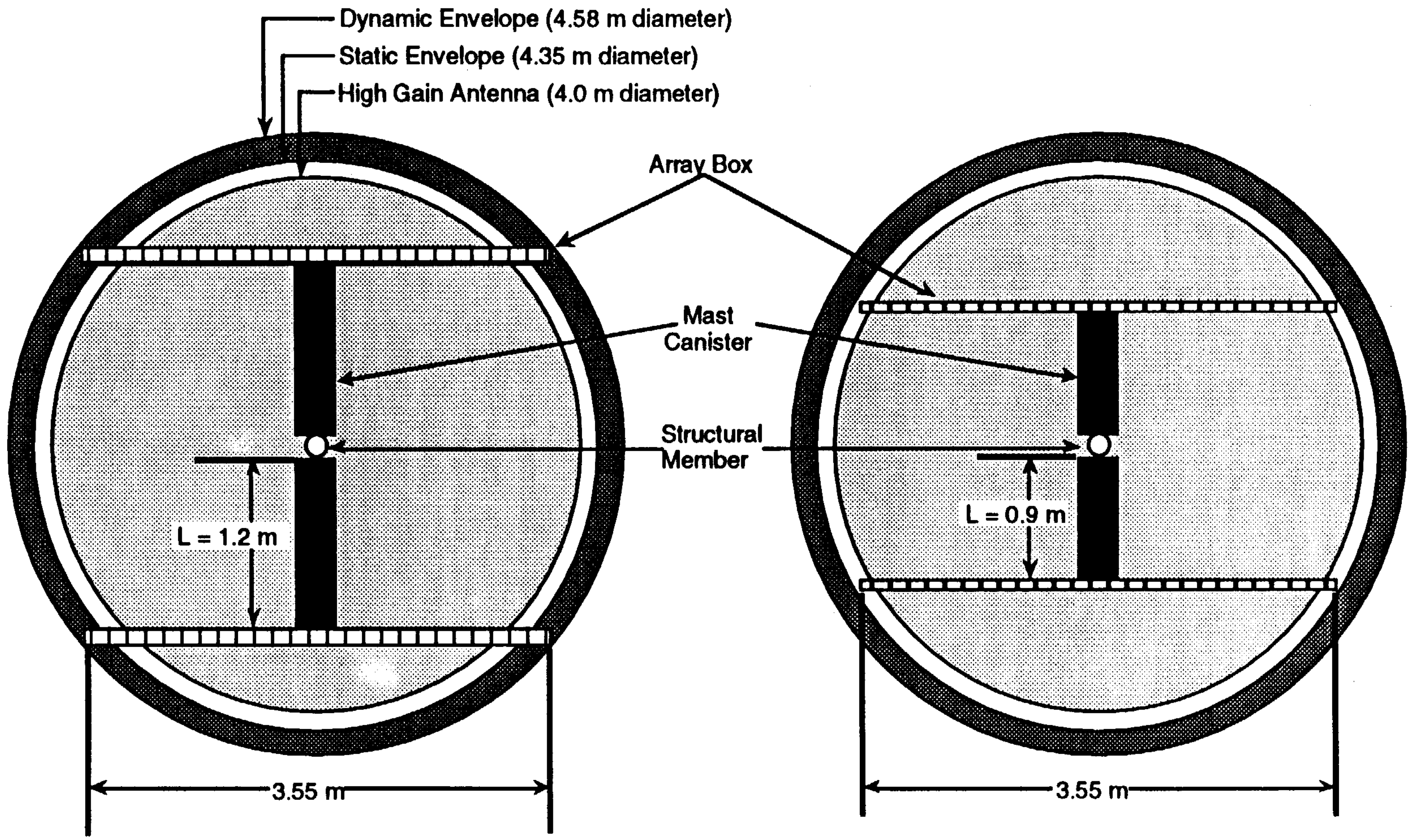


Figure 4-2b. Stowed Configuration Representation
(For clarity of presentation, only two masts are shown here)

4.2.4 Mission and Science Impacts

4.2.4.1 Implications of Mass Increase. The smallest dry mass increase is 1337 kg (2948 lb) for the unarticulated GaAs configuration. This amounts to an increase in the dry mass of approximately 63% over the original baseline Orbiter allocation of 2175 kg (4796 lb). This increase, along with the propellant required to execute the baseline mission plan, is well beyond the launch capability of the Titan IV (SRMU)/Centaur. As a result of this, significant alteration of the mission plan was attempted to determine the feasibility of an all-solar design under conditions of increased programmatic risk and reduced science goals. These alterations are summarized in Table 4-4.

Table 4-5 summarizes the total masses of the RTG and all-solar designs. The propellant masses for all of these designs are based upon the propellant required for the modified mission tour. As shown in the table, even with the “modified” mission tour, the all-solar options are substantially more massive than allowed by the baseline launch vehicle.

4.2.4.2 Implications of Spacecraft Design. In addition to mass, some other issues related to the spacecraft design itself for the all-solar option are addressed here. The list is not comprehensive; if the all-solar option were to be considered further, other issues would likely be identified.

4.2.4.2.1 Payload Effectiveness. Even by assuming that many of the high-precision pointing instruments would remain on a scan platform and some of the fields and particles instruments would remain on a turntable, the all-solar design would still impact payload effectiveness. Some of these implications are addressed here.

A. Field-of-view (FOV) and Pointing Restrictions

The total area of the four solar array panels is 1152 m^2 (12,380 ft^2) for the silicon cell option or 598 m^2 (6430 ft^2) for the gallium arsenide cell option, which is equivalent to 5.3 or 2.8 tennis courts, respectively. Clearly, these panels will restrict the instruments on the scan platform from pointing in certain elevation and azimuth ranges, and may constrict the FOV of other instruments. For observations by instruments on the scan platform, more spacecraft turns will be required during observations due to the increased blockage (requiring additional hydrazine due to more frequent off-loading of the reaction wheels), or else some observations will be cut short or eliminated entirely.

Table 4-4. Mission Plan for Original All-Solar Cassini

Event	Modified Plan	Baseline Plan
Cruise Duration	9.3 years	6.7 years
Saturn Orbit Insertion Period	300 days	216 days
Interplanetary ΔV Allocation	95% confidence	99% confidence
Saturn Tour Duration	3 years	4 years

Table 4-5. Launch Masses for the Original RTG Baseline and All-Solar Cassini (kg)

	Original RTG Baseline****	Original All-Solar Option (Si)	Original All-Solar Option (GaAs)
Orbiter Allocation	2175	2175	2175
Probe Allocation*	346	346	346
Monopropellant	20	368	219
All-solar Mass Δ**	0	1918††	1337†††
Launch Adapter	240	400	350
Bipropellant***	1740	3300	2801
Total Spacecraft Mass:	4521	8507	7228
Titan IV (SRMU)/Centaur Performance (including reserves*****)	6234	6234	6234
Margin (performance minus spacecraft mass)	1713	-2273	-994
Launch Vehicle Margins and Reserves†	875	875	875

* Including Probe support equipment
** Non-articulating arrays
*** Bipropellant mass estimates are for modified mission described in Table 4-4 (ΔV=1576 m/s)
**** JPL, 1991b
***** Allowable spacecraft mass
† Amount withheld from allowable spacecraft mass, due to launch vehicle and development uncertainties
†† From Table D-3
††† From Table D-8

B. Impact of Turn Times

A major objective of the Cassini mission is to acquire radar data of Titan's unseen surface. The increased turn times of the all-solar design (relative to the baseline design) would result in some combination of less radar data returned, radar data resumed at lower resolution, and taking time away from other (non-radar) near-

Titan activities. For example, during the far-approach to Titan, radiometry data are acquired by having the spacecraft antenna trace a conical path, achieving as much global coverage of the surface of Titan as possible. With the lower turn rate, significantly less coverage would be acquired during the same allotted time on a given pass. Achieving the same coverage on a given pass would either mean starting farther out, thus accepting even lower resolution data, or reducing the time allocated for the next activity, i.e., acquiring altimetry data. Alternatively, the lost coverage could be regained by assigning additional Titan flybys to radar activities, but this would reduce the time allocated for other near-Titan activities, such as radio science. It is likely that the result would be reduced global coverage on each pass. Another radar mode, the spotlighting mode, would be significantly affected as well. During this mode with the baseline spacecraft design, a spot of particular interest on the surface of Titan is illuminated by the radar throughout the pass. Turn rates of up to 0.4 deg/sec are required at the nominal closest approach distances. The lower turn rates of the all-solar design would mean that a point could not be followed at the closer distances, thus resulting in less data and lower resolution information on these interesting locations, such as the landing site of the Huygens probe.

A similar problem exists for the radio science occultation experiment at Titan. By receiving signals from the spacecraft through Titan's atmosphere, ground-based instruments can analyze the Doppler-shifted frequencies and reveal characteristics of that atmosphere. At the start of this experiment, the spacecraft is pointing and transmitting to Earth. As Titan approaches between the spacecraft and Earth, the signal first passes through Titan's atmosphere (the ingress) and then is blocked by the disk itself. Within a few minutes, as the disk continues to move, the signal is again reestablished, passing through the atmosphere on the other side (the egress). Since the atmosphere acts as a lens and bends the signal, the spacecraft is directed to turn to track the edge of the disk for a short time at the ingress, keeping in contact with Earth, until the bending is no longer adequate. Then the spacecraft must rotate rapidly across the disk of Titan and prepare to transmit to Earth, through a bent signal at first, at the egress point. Again, the spacecraft slowly tracks the edge of the disk, until the outer edge of the atmosphere has completely passed and the signal to Earth is a straight line once more. For the baseline RTG-powered spacecraft design, pointing rates and accelerations of up to 0.4 deg/sec and .004 deg/sec^2, respectively, are required to rotate across the disk in the required time, in order to be ready to transmit to Earth at the egress point. If these cannot be achieved, the experiment can be conducted at only one of the two points, ingress or egress, on each Titan flyby. This either reduces the return of this unique type of atmospheric data by about half or necessitates that more Titan flybys be reserved for radio science, at the expense of another experiment (e.g., radar). In addition, the advantage of acquiring these atmospheric data at two different longitudes close to the same point in time would be lost. When data at different longitudes are acquired at nearly the same time, differences can be attributed primarily to the longitudinal variations. However, when they are acquired months apart, it is questionable whether the observed differences are due to the longitudinal variation or other changes that have occurred with time. (Due to the greatly increased settling time required at the end of turns for the all-solar design, the pointing rate and acceleration values would need to be even greater than those quoted above, unless the reduced pointing accuracy would be acceptable.)

Time required for spacecraft turns during the tour can be thought of as overhead—it takes away from the time available for science. Since the original all-solar spacecraft design described in Appendix D includes a scan platform for science, the two turns per day required in the current baseline mission can be eliminated. This capability reduces the overhead turn time. On the other hand, since other kinds of turns are still required (e.g., for some science observations described above, and to keep constraints from being violated with changing geometries), the lower turn rates of an all-solar design, coupled with the longer settling times, would have the effect of increasing the overhead turn time. Whether the result is a net increase or decrease in time available for science would need more detailed study.

C. Electromagnetic and Ionizing Radiation Interference

Experience with fields and particles instruments indicates that solar-powered spacecraft generate much higher levels of interference than RTG-powered spacecraft. The effect of the presence of four large, charged panels would need to be evaluated for each instrument. A conductive coating might keep the charge from building up, thus minimizing electrostatic interference. However, this procedure has only been demonstrated to work well on rigid arrays and could prove difficult on flexible or foldup arrays, as required for a Cassini all-solar spacecraft. This technique, nevertheless, still does not help with the fluctuations in current that cause the dynamic component of the electromagnetic interference.

D. Impact of Later Saturn Arrival Date

With the longer cruise duration assumed (see Table 4-4), arrival at Saturn would be in March 2007. Even with a shortened tour of only three years, the negative effects of the closing of Saturn's rings as seen from Earth apply more strongly to this tour than to that of the baseline mission (see Subsection 3.3.2.1).

4.2.4.2.2 Spacecraft Operability.

A. Reliability

Deployment of appendages is one of the riskier operations of a spacecraft mission. The all-solar Cassini design, even with GaAs/Ge solar arrays, has four solar array wings, each of which is five times the size of one Hubble Space Telescope wing. The deployment of these arrays adds considerable risk to the mission as compared to the baseline, for which deployment of the RTGs is not required.

There is also inherent risk in an all-solar design of the spacecraft turning off-Sun and, through a command (or other) error, not turning back before the battery is depleted. This situation could prove catastrophic to the mission. It is the probable cause for the demise of the Soviet Phobos mission, for example. To maintain this risk at an acceptable level, the operational software and procedures would have to be increased in sophistication and complexity. This is particularly important with the round-trip communication time at Saturn being up to 2.8 hours and the batteries being sized for 14.5 hours, a representative value for the maximum expected eclipse time. More batteries could be added, but at a fairly significant mass cost.

B. Torque

Control of the all-solar design spacecraft would be made difficult with the four very large solar array wings. This could be a particular problem at the lower closest approach distances to Titan, down to 950 km for the baseline mission. To keep the risk of uncontrollable torque on the spacecraft sufficiently low, a higher minimum altitude might be selected. If the value chosen were high enough, significant impact to the science return and to the mission design could result. For example, a minimum altitude greater than 1600 km would eliminate the high-resolution imaging radar data. Mission design impacts, along with the ensuing science impacts, could include not reaching as high an inclination at the end of the tour and not covering as full a range of solar orientation angles.

C. Early Cruise Maneuvers

Cassini must execute some (typically) small maneuvers during the early, interplanetary cruise phase of the mission. A maneuver shortly after launch must correct for slight errors in the Centaur injection burn that places the spacecraft on its interplanetary trajectory. Furthermore, small correction maneuvers are needed before and/or after each Earth and Venus swingby in order to stay very close to the trajectory designed to achieve the precise change in energy required of the swingby. For thermal reasons, the baseline spacecraft design constrains the durations of these maneuvers by the following equation:

$t_{max} <= 0.5(d/0.6)^2$ (where t_{max} is the maximum off-Sun time in hours and d is the heliocentric range in AU).

The closest distance to the Sun at which the spacecraft must be able to execute a maneuver is 0.6 AU. At that distance, the off-Sun limitation is 30 minutes. The actual burn times of these maneuvers, as opposed to some larger deep space maneuvers or the Saturn orbit-insertion maneuver, are typically very short in duration (seconds) when executed with the main engine. Before each burn, however, a spacecraft turn is required for proper orientation of the main engine. A typical turn is 90°, but it can be close to 180°. After the burn, the spacecraft must turn back to the original, thermally safe cruise orientation. These turn times account for most of the off-Sun time counted in the equation above for a given maneuver. To make the turns, the reaction wheels do not provide sufficiently short turn times. Hence, the thrusters are used. For the baseline spacecraft design, a large turn, the required spacecraft settling, the maneuver burn itself, and the second turn can all be accomplished within the 30-minute requirement.

However, for the all-solar design, the moment of inertia is greatly increased on all axes. The combination of the longer turn times and the longer settling times make it nearly impossible for the 30 minute constraint to be met. Larger thrusters and longer mounting booms could reduce the actual turn time. If the all-solar option were to be considered further, an analysis would need to be done to see whether a viable solution could be found. In any case, more hydrazine would be required (in addition to the sizable increase in hydrazine needed to counteract solar torques, as identified above in the discussion on mass as well as in Appendix D).

Perhaps a better solution would be to add several more thrusters in the proper configuration so that turns would not be required at all. By choosing the proper combination of thrusters, any maneuver could be performed in a lateral fashion, while the spacecraft maintained its thermally safe attitude. The main engine would not be used. The additional thrusters and required supporting structure, however, would still increase the spacecraft mass and add complexity and risk to the spacecraft design. (This option was used as the baseline design for some period of time, but abandoned due to the increased complexity and risk to the spacecraft.) In addition, due to the inefficiency of the lateral burns and lower specific impulse of the thrusters compared to the main engine (about 210 s vs. 308 s), increased propellant would be required.

4.2.5 Tethered Design

A more radical step would be to utilize a tethered design, which would decouple a power module containing the solar arrays from an instrument module containing the science instruments. Although this decoupling could greatly improve stability for the science instruments, it is likely that the programmatic risks, operational complexity, difficulty in doing trajectory correction maneuvers, mass, and costs involved in such a bifurcated spacecraft design would be greater than the nominal solar design.

4.2.6 All-Solar Design Conclusions

In summary, the extremely large mass of the lightest solar configuration is beyond the capabilities of the Titan IV (SRMU)/Centaur launch vehicle, even assuming a truncated satellite tour involving substantial reduction in science. An all-solar Cassini must therefore be considered infeasible at this time. Even if a heavy lift booster and a suitable upper stage were made available, the severe field-of-view problems, greatly increased turn times, and greater operational complexity and programmatic risk associated with the all-solar Cassini design makes this option scientifically untenable. For these reasons, the nominal all-solar Cassini design is not considered further in this study.

4.3 SUBOPTIONS TO THE NOMINAL ALL-SOLAR DESIGN

A number of suboptions to the nominal all-solar design (Appendix D and Subsection 4.2) have been developed, and design details for some of them are discussed further in this subsection and Appendix E. The purpose of these suboptions is to reduce the mass of the solar array. Mass is a primary concern because of launch vehicle and trajectory constraints.

The suboptions being evaluated in this section include the RTG/solar hybrid, concentrator photovoltaic array, Solar-Electric Propulsion (SEP), and Solar Thermal Propulsion (STP). Each represents a trade-off in cost, development time, science return and reliability compared to the baseline and the nominal all solar design. Unfortunately, predicting the cost, development time and reliability for many years operating lifetime becomes a very qualitative and uncertain task.

4.3.1 RTG/Solar Hybrid

The hybrid design represents a compromise between the nominal all-solar and baseline designs: Use of solar energy reduces the amount of Pu-238 needed, but the larger the fraction of power that is supplied by solar energy, the greater the power system mass. Reliability of the hybrid design will be lower than the baseline since a successful mission will depend on the successful simultaneous operation of two different technologies. This combines failure mechanisms of each power supply into the system and increases the design complexity. Science return from the hybrid design will be reduced compared to the baseline because of increased turn times, reduced field-of-view, and greater susceptibility to solar pressure due to larger exposed surface area.

Another hybrid option would include the use of radioisotope heaters instead of electrical resistance heaters on an otherwise all-solar power system design. Analysis indicates, however, that the relatively high efficiency of the solar arrays and large areas required for electrical power make it unreasonable to introduce the complication of launching Pu-238 heating sources for the small amount of heating power required. (On the other hand, the relatively low-efficiency RTG thermoelectric converters in the baseline design should not use electrical heaters, because converting Pu-238 heat to electricity, then converting the electricity back to heat, simply means launching more plutonium than would be required if the heat were used directly.) A third hybrid option, using solar arrays to supply heat for an RTG-powered spacecraft, would create a significant complication with a negligible environmental improvement.

4.3.2 Concentrator Photovoltaic Suboption

As discussed in Appendix E, concentrator photovoltaic arrays represent a form of photovoltaic converter system in which a percentage of the solar cells are effectively replaced by refractive or reflective optics. These optics increase the output of the remaining solar cells by increasing the effective incident solar illumination, thus providing a higher output for a given cell area relative to standard (nonconcentrating) photovoltaic systems. There are two reasons for examining the use of concentrator arrays: (1) the possibility of lowering overall array cost by replacing expensive solar cells with “low cost.” optics while still providing an equivalent array output, and (2) an additional reduction in the number of solar cells required to provide a given output due to the shielding provided by the concentrator optics (to protect solar cells from damaging particulate space radiation). In practice, the former has yet to be demonstrated, since space qualified optics and system assembly costs are not yet “low cost.” The utility of the latter increases with increasing concentration ratio, although the large mass of the concentrator elements has generally resulted in system masses much higher than equivalent area planar arrays. This is partly due to the development of those systems by the U.S. Department of Defense (DoD) to withstand a wide variety of natural and man-made threats.

4.3.2.1 Concentrator Photovoltaic Array Design. For the Cassini mission, where solar intensity will be reduced by a factor of approximately 100 from the near-Earth value, the choice of a high concentration ratio would appear desirable. Though ratios as high as 30 to 50 can, in theory, be achieved and would provide incident cell illumination intensities high enough to avoid any occurrence of LILT degradation in silicon, this would impose a stringent pointing accuracy requirement across the array (i.e., no element could be more than approximately two degrees off the normal solar incident angle). Unlike planar arrays, concentrator performance can quickly drop to zero if the pointing requirement is not met. Furthermore, cell illumination intensities at 0.6 AU during the Venus flyby, or even 1 AU around the Earth, would lead to excessive (catastrophic) heating without the use of relatively massive heat sinks for the cells. This could possibly be avoided by the use of a variable concentration ratio, but no design has been developed and such a system would likely be very complex. Low concentration ratios (2 to 10) are possible, although this may increase susceptibility to LILT degradation at Saturn.

A final point is that cell efficiency can be improved slightly by a modest increase in illumination intensity and a decrease in cell operating temperature. However, increased intensity is generally accompanied by increased operating temperatures, requiring improved radiator designs to avoid severe efficiency losses (particularly at 1 AU). As a result, practical concentrator systems provide only a small (10 to 20%) improvement in power density (W/m^2) when compared to planar arrays. Consequently, the area of a Cassini array, either concentrator or planar, would be extremely large. In the case of a concentrator array with its relatively thick cross section, there would also be a significant increase in stowage volume, typically by a factor of 100 or more compared to a planar array.[3]

Ground testing of a large, medium-to-high concentration ratio array is expected to be quite difficult, due to the influence of gravity on the system and the need for accurate alignment to verify the proper light gathering behavior of the array. Costs are obviously difficult to determine without a well-defined system, but estimates indicate that the attendant reduction in cell quantity associated with a 20 to 50x concentrator system would result in a somewhat lower cost for the concentrator array compared to an equivalent planar array. The extent of this cost saving might not be significant, since overall design, fabrication, and test challenges would be quite formidable. The ability to perform ground tests, the need for additional thermal control, and the large size of the array are factors that would produce a high mass impact for the concentrator array. It is estimated that the mass of a concentrator (20x to 50x) system for Cassini would be a factor of three greater than an equivalent power planar system. In light of the currently available launch vehicle lifting capabilities discussed previously, this mass increase causes concentrator arrays to be eliminated from consideration for the Cassini spacecraft.

[3] This is based on existing high concentration ratio designs investigated during the past five years by DoD and NASA (Piszczor, 1990 and Stern, 1988).

4.3.3 Solar-Electric Propulsion (SEP) Suboption

As discussed in Appendix D, an all-solar Cassini spacecraft would require large arrays. Even if a spacecraft with such large arrays were practical in terms of maneuverability and reliability, its mass would be far in excess of current U.S. launch vehicle capabilities.

For SEP, however, the situation is somewhat different. By using the large arrays for propulsive augmentation and reducing their size (thereby reducing the mass), the trajectory injection energy requirement and launch vehicle lift constraints could possibly be met. Also, array size and mass could probably be reduced somewhat to accommodate launch vehicle lift constraints. In order to achieve the necessary injection energy, an SEP spacecraft would require about 20 to 30 kW in the vicinity of the Earth. This power requirement would necessitate that the four arrays only be on the order of 10 to 20 m in length, substantially reducing the associated mass. However, once at Saturn, the arrays would only be capable of supplying a portion of the requisite EOM watts of power. So, an SEP Cassini would still require one or more RTGs and would, therefore, constitute little more than a higher risk, higher cost alternative for achieving a non-Earth gravity assist. This characterization stems from the fact that an SEP mission to Saturn would still have to employ multiple gravity assists and would, therefore, still require approximately the same flight time. In addition, most of the chemical propulsion system and bipropellants would have to be carried along in order to conduct the short, high-thrust maneuvers necessary for Saturn orbital injection and conduct of the Saturnian tour, thereby eliminating much of the mass advantage traditionally associated with SEP. At the same time, however, the Project would add significant risk and cost on top of that already associated with the all-solar/chemical propulsion option. This increased risk and cost would assume two forms: technical and programmatic.

Technical problems arise from having to develop and operate a never-before-constructed ion engine capable of thrusting and throttling for many thousands of hours (far longer than any ion engine has ever been operated, even on the ground) in tandem with never-before-flown solar arrays. To bring such an engine to the point of flight development would require at least four-and-a-half years of development and life testing; an amount of time which, in and of itself, would consume any potential gains in transit time. NASA, in conjunction with the Air Force, plans to fly an SEP experiment to demonstrate orbit raising from LEO to GEO. This represents an early step in the flight development process. Even more time and resources, however, would be required to design and develop the surrounding propulsion system, supporting structure, power processing units, and power controller. All of these new, never-before-constructed systems would then need to undergo extensive vibration, thermal vacuum, flight, and spacecraft integration testing. In short, adding SEP to an all-solar Cassini would substantially increase cost and technical uncertainty.

The process of attempting to add SEP to a nominal all-solar Cassini would also lead to a substantial increase in programmatic risk and cost. The time needed to design, develop, and test all of the above SEP components would necessitate delaying the mission for a number of years.

4.3.4 Solar-Thermal Propulsion (STP) Suboption

As mentioned in Appendix A, the development of new, non-chemical propulsion technologies might enable indirect ballistic or non-Earth, gravity-assist trajectories to Saturn. STP's concentrator arrays might also provide a means for augmenting power, given sufficient funds and time to develop, test, and construct such a system (including development of an inflatable solar collector).

Because it is effective only at relatively low solar ranges, STP would be useful for propulsion only as a sort of high-performance, long-thrusting upper stage. (In Sercel, 1985, the solar-thermal propulsion module was used to augment a Centaur upper stage.) Cassini would definitely require a chemical propulsion system for Saturn orbit injection and maneuvers during the satellite tour, all of which occur at extreme distances from the Sun. An STP module would be used only during initial injection and then be discarded. The remainder of the mission would require an all-chemical system very similar in size and capacity to the current baseline design.

Assuming the inflatable concentrators could be developed and made to last many years, they could be used in power generation if they were not discarded during the early portions of the mission. However, like SEP, analysis indicates that the area of the concentrator would be insufficient to provide adequate electrical power. For this reason, Cassini would require additional means to generate electric power. However, if RTGs were used, STP would provide an environmentally distinct alternative to the baseline plan only in that it would reduce the Pu-238 requirement and have the potential to enable a non-Earth gravity-assist mission. As indicated in Subsections 3.2 and 3.3, however, alternate trajectories exist for Cassini that can be flown with a conventional, currently existing upper stage. While it is expected that STP would enable more opportunities for non-Earth gravity-assist missions and also shorten the flight time, these advantages would have to be weighed against the serious drawbacks of the higher development costs and programmatic risk associated with this new propulsion technology.

In summary, it would appear that STP offers more of a potential performance enhancement, as opposed to offering a significantly enabling new technology. A flight test of solar-thermal propulsion using inflatable concentrators is currently planned circa 2008 (Lowe, 1992). Substantial development efforts and funds would have to be employed to accelerate this schedule to a point where it could be confidently employed for Cassini and still return the science data within the time period of interest.

4.4 SENSITIVITY ANALYSTS: ADDITIONAL MEASURES FOR ENABLING CASSINI POWER ALTERNATIVES AND HOW THEY IMPACT MISSION SCIENCE OBJECTIVES

As discussed earlier in this section, an all-solar spacecraft seeking to satisfy, but still falling short of, the minimum Cassini science objectives entails a mass and associated injection energy requirement that exceeds the capabilities of all existing U.S. launch vehicles. In this subsection, we explore a second design point to determine the extent to which Cassini science objectives would have to be sacrificed before an all-solar spacecraft would be compatible with such launch vehicle mass constraints.

4.4.1 Mass Reduction Through Body-Fixed Instruments and Reduced Power for Science

The second design point, the reduced science all-solar Cassini, is described in Appendix F. It is based on the current baseline Cassini, which is the redesigned Cassini. The science platforms are body-fixed, eliminating the deployable booms and platform actuators. Also, the power available to science instruments during observational modes is further reduced by 50%, resulting in smaller solar arrays and additional mass savings. This power reduction would, however, necessitate the elimination of instruments from the payload or a reduction of data gathering opportunities for each instrument, or some combination of the two. The spacecraft launch masses for the current baseline Cassini and the reduced science all-solar Cassini are shown in Table 4-6. As shown, the mass of the reduced science design is still over 50 kg above the allowable launch mass. A positive launch margin can only be obtained by eliminating part or all of the launch vehicle margins and reserves. However, a reduction in reserves increases programmatic risk, as past experience has repeatedly demonstrated that such margins are required for successful project development.

As with the original all-solar design point, it must be realized that the level of detail in the study of the reduced science all-solar design does not include all of the impacts of modifying the Cassini mission to use solar power. Some examples of effects not considered, both of which would further increase array size and mass, include the need to replace RTG waste heat used to warm the propellant tanks, and the modification of the thermal and attitude control designs. In addition, most of the impacts to the first design point described in Subsections 4.2.3 and 4.2.4 under “Implications of Spacecraft Design” apply to this second design point.

Taking a closer look at one of the impacts of the reduced science design, that of increased spacecraft turn times combined with the elimination of the articulated science platforms, uncovers a number of problems. These include: (1) increased times for imaging mosaics, (2) inadequate turn rates for fields and particles instruments, (3) reduced image resolution due to inadequate target motion compensation, and (4) loss of instrument observation time during turns for communication with Earth. A possible solution to the turn rate problem is to increase reaction wheel capacity, at the expense of larger, more massive reaction wheels (larger than Upper

Table 4-6. Launch Masses for the Current RTG Baseline and the Reduced Science All-Solar Cassini (kg)

	Current RTG Baseline***	Reduced Science All-Solar (GaAs)
Orbiter Allocation	2150	2150
Probe Allocation*	352	352
Monopropellant	62†	207
All-solar Mass Δ**	0	876††††
Launch Adapter	190	250
Bipropellant***	1756	2458
Total Spacecraft Mass:	4510	6293
Titan IV (SRMU)/Centaur performance (including reserves††)	6234	6234
Margin (performance minus S/C mass)	1724	-59
Launch Vehicle Margins and Reserves†††	875	875

* Including Probe support equipment
** Non-articulated arrays
*** Bipropellant mass estimates are for modified mission described in Table 4-4 (ΔV=1576 m/s) .
**** JPL, 1992
† Lisman, 1992
†† Allowable S/C mass
††† Amount withheld from allowable S/C mass, due to launch vehicle
†††† From Table F-1

Atmospheric Research Satellite, or UARS, reaction wheels). Another possibility is to use spacecraft thrusters to perform spacecraft turns, requiring increased monopropellant. The options would need to be studied to determine feasibility.

Other possibilities exist as well, but all would result in increased spacecraft mass. Given the fact that a positive injection margin is possible for the reduced science all-solar design only through elimination of some of the launch vehicle margin, any additional increase in mass further removes this option from the realm of possibility.

Between the two design points discussed above, the original all-solar and the reduced science all-solar, lies a broad spectrum of mission opportunities and spacecraft configurations that yield varying levels of science return. Many intermediate scenarios can be postulated which would increase science return at the expense of spacecraft mass. However, since the second design endpoint, with a mass reduction to the point where the science no longer supports the basic objectives of the Cassini mission, is still, at best, marginal from a mass perspective, any alternatives which provide more science at the expense of increased mass are even more implausible.

4.4.2 Mass Reduction Through Reversion to a Saturn Flyby Mission

Another method for achieving the required mass reduction might involve foregoing orbital insertion at Saturn, i.e., making the mission a Saturn flyby mission. Restricting the spacecraft to this degree of capability would result in a substantial mass reduction through reduced propellant requirements. Elimination of orbital insertion at Saturn and subsequent tour-related maneuvers would reduce the mass of required bipropellant, monopropellant, and tankage. However, changing the mission from an orbital tour of the Saturnian system to a flyby entails a radical departure from Cassini's science objectives. The flyby would preclude fulfillment of at least one of the two requirements fundamental to achieving the Cassini mission objectives: a four-year tour of the Saturnian system involving close flybys and a variety of orbital geometries (Volume 1, Section 3-1). In so doing, it would impede satisfaction of all of Cassini's mission objectives requiring observation across time and/or in three dimensions. Under these circumstances, the Cassini mission would be reduced to a Voyager-type mission: a mission that has already been done.

Turning the mission into a flyby would make it impossible to release the Huygens Probe. A Probe-less mission would fail to accomplish a number of science objectives, and renege on the international agreements that have been reached concerning the Probe. The heat shield for the Probe can be designed for an entry velocity of 6 km/s, compatible with the current baseline mission. The entry velocity would increase for a flyby mission, to about 15 km/s. The design of the Probe's heat shield has a V^2 dependence; it would not be possible to accommodate the increased velocity.

APPENDIX A

TRAJECTORY AND LAUNCH VEHICLE POTENTIAL ALTERNATIVES

This appendix provides general background material about trajectory and launch vehicle potential alternatives. Subsection A.1 describes three types of interplanetary trajectory potential alternatives and their associated on-board propulsion subsystems: (1) direct Earth-to-Saturn trajectories, (2) those employing planetary gravity assists, and (3) low-thrust trajectories. The focus of Subsection A.2 is on various integrated launch systems.

A. 1 TRAJECTORY AND PROPULSION BACKGROUND INFORMATION

Two values important to interplanetary trajectory design, C_3 and $\overline{V}_{\infty L}$, are illustrated in Figure A-1, a sample trajectory to Saturn. The quantity C_3 (equal to $\overline{V}_{\infty L}^2$) is called launch or injection energy. C_3 is a measure of the amount of energy that the launch vehicle must impart to the spacecraft to inject it onto an interplanetary trajectory. $\overline{V}_{\infty L}$ is the excess velocity the spacecraft would have in a hyperbolic orbit of departure, if it were to travel an infinite distance away from the Earth.

The spacecraft's velocity with respect to the Sun, $\overline{V}_{s/c}$, is found by vector addition: the Earth's velocity with respect to the Sun, $\overline{V}_E$, is added to the spacecraft's asymptotic, hyperbolic departure velocity with respect to the Earth, $\overline{V}_{\infty L}$. For transfers to the outer solar system, the net effect of the Earth departure hyperbola is to increase the spacecraft's velocity with respect to the Sun above the value corresponding to the Earth's velocity. This requires that $\overline{V}_{\infty L}$ be pointed in the same general direction as the Earth's velocity vector. For transfers to the inner solar system, however, the spacecraft's velocity must be decreased, requiring $\overline{V}_{\infty L}$ to be opposed to the Earth's velocity.

In a similar fashion, the spacecraft's arrival velocity at the target body, $\overline{V}_{\infty A}$ is its velocity with respect to the Sun, $\overline{V}_{s/c}$, minus the target body's velocity with respect to the Sun, $\overline{V}_S$. A trajectory correction maneuver is required for braking at this time; the spacecraft's arrival velocity must be decreased so that it is captured into orbit instead of flying by. Rendezvous and orbiting missions require additional ΔV at the target body, while flyby missions do not. For a mission such as Cassini, the velocity change required to establish the desired orbit at the target body increases with increasing $\overline{V}_{\infty A}$.

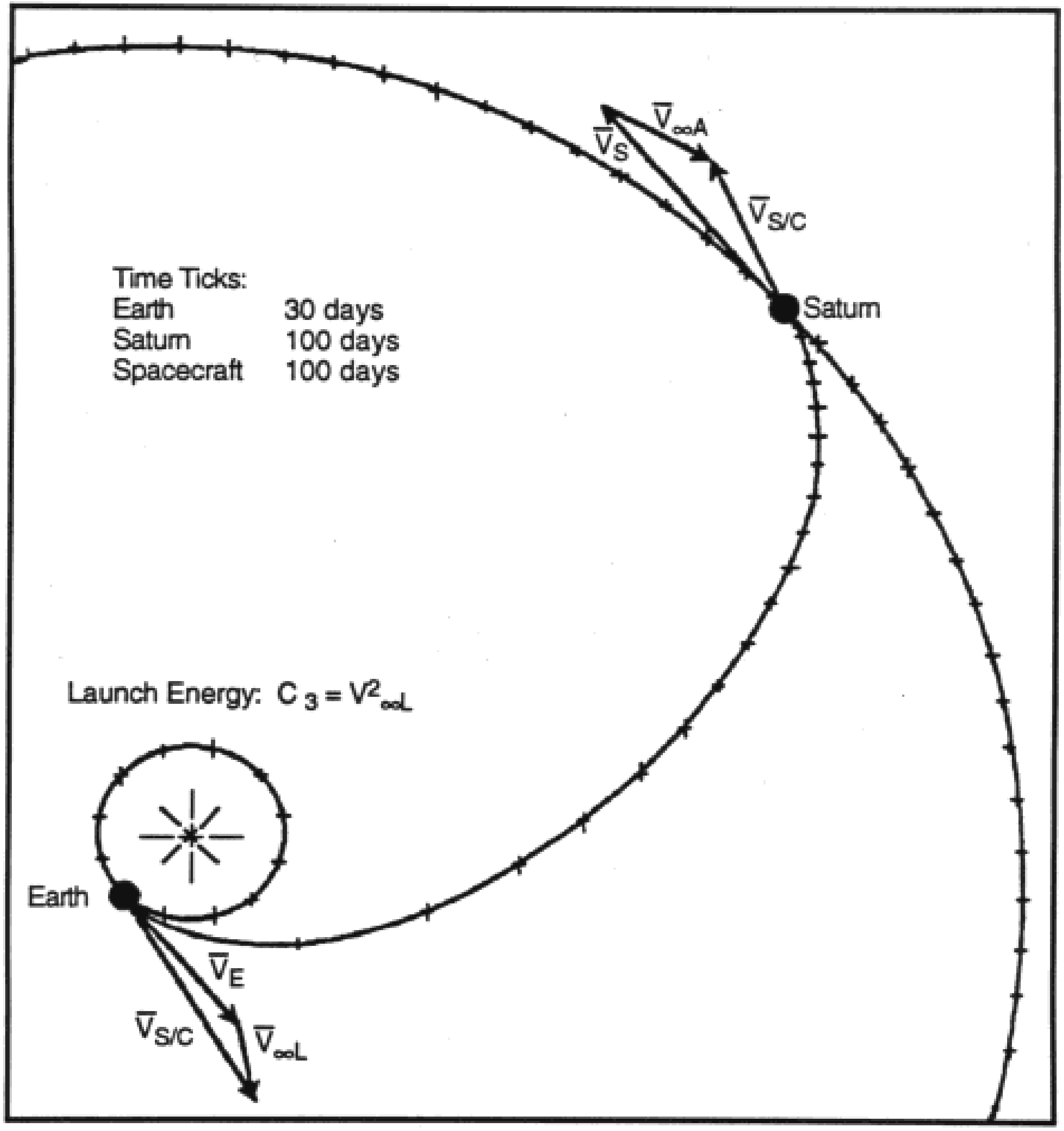


Figure A-1. Launch Energy and Arrival Velocity Definitions

Interplanetary trajectories can be classified as being either ballistic or non-ballistic. In a purely ballistic trajectory, the spacecraft's flight path is shaped solely by the gravitational influence of the Sun and planets. However, when a flight path includes ΔVs produced by the spacecraft's on-board propulsion subsystem, it is called a non-ballistic trajectory. These deterministic ΔVs have been provided by the use of a high-thrust chemical rocket engine with a specific impulse (I_{sp}) in the range of 200 to 450 s, supplying near-instantaneous changes in velocity along the trajectory. The resulting flight path consists of two or more ballistic trajectory arcs connected by spacecraft maneuvers.

All planetary missions flown to date or currently under development have used spacecraft equipped with chemical rocket engines. Velocity changes also can be accomplished by using other, non-chemical forms of propulsion that supply very low thrust levels. These low-thrust propulsion systems, which have never been used as a primary means of propulsion for a planetary mission, provide continuous thrust for much longer periods of time (i.e., over substantial fractions of the transfer orbit), resulting in trajectory arcs that are significantly different in character from typical ballistic ones. These non-ballistic trajectories, however, may contain extended coasting phases where the trajectory arcs are ballistic.

The unit of measurement used in discussing interplanetary travel is the astronomical unit, or AU; 1 AU is defined as the mean distance from the Sun to the Earth.

A.1.1 Direct Trajectories

It is highly desirable for planetary missions to use direct interplanetary transfers for the trajectory from Earth to the target body. Direct trajectories are the simplest interplanetary transfers and are characterized by very high launch energy and short flight time, which allows earlier science data return, lower operations costs and complexity, and greater spacecraft reliability. (The longer the mission duration, the greater the chance of a malfunction.) They impose few special requirements on spacecraft design and are always the preferred transfer mode if sufficient launch vehicle performance is available. One consideration, however, is that their higher arrival velocity necessitates more fuel and, therefore, larger tanks for a braking maneuver at the target body. Direct trajectories typically utilize chemical propulsion, do not employ planetary gravity-assist swingbys, and can be ballistic or non-ballistic.

Direct transfer opportunities are dependent upon favorable planetary alignment and repeat only when the same relative planetary alignment occurs. The time interval between such opportunities is called the synodic period. For the outer planets such as Jupiter and Saturn relative to Earth, it is slightly greater than one year. The synodic periods for Venus and Mars are 1.6 and 2.1 years, respectively. Some of these opportunities are a little better than others in performance due to slight orbital eccentricities and the relative inclinations of the two bodies involved. These small variations are usually of secondary consideration in the selection of a launch opportunity.

A.1.2 Gravity-Assist Trajectories

Like direct trajectories, this type of interplanetary transfer routinely depends on chemical propulsion, but gravity-assist trajectories utilize one or more planetary swingbys to modify their trajectory and significantly reduce either launch energy, arrival velocity, or flight time requirements. Several past missions have used gravity-assist swingbys. Pioneer 11, launched in 1973, used a Jupiter gravity assist to increase spacecraft velocity sufficiently to allow an arrival at Saturn. Mariner 10, also launched in 1973, used a Venus gravity assist to decrease spacecraft velocity in order to arrive at Mercury. The gravity assist from the Mercury encounter was then used to adjust the period of the orbit so that the spacecraft resumed to Mercury for two additional swingbys. Voyagers 1 and 2 (1977), like Pioneer 11, each used a gravity-assist swingby of Jupiter to reach Saturn. Voyager 2 also used one at Saturn to go on to Uranus, and a gravity assist of Uranus to target the spacecraft on to Neptune. For these early missions, the planetary gravity-assist technique was used to lower the launch energy requirement significantly below that of a direct, non-gravity-assist transfer, and to achieve multiple planetary swingbys in a single mission.

Recent missions using gravity-assist transfers include the International Cometary Explorer (ICE) mission to the comet Giacobini-Zinner and the Soviet Vega mission to the comet Halley. The ICE trajectory required several close lunar swingbys for gravity assist, while the VEGA mission used Venus swingbys to establish the transfer trajectories to Halley for the two VEGA spacecraft. The Galileo spacecraft successfully flew by Venus in 1990 and the Earth in 1990 and 1992 for planetary gravity assists that will enable the spacecraft to encounter Jupiter in December 1995.

A.1.3 Low-Thrust Trajectories

The trajectory modes considered thus far involve propulsion phases that are extremely short compared with the total transfer time of these missions. As described above, these transfer modes involve the use of a chemical rocket engine with a specific impulse in the range of 200 to 450 s, providing near-instantaneous changes in velocity along the trajectory. A low-thrust propulsion system, on the other hand, provides thrust over an appreciable fraction of the transfer trajectory. It is characterized by thrust acceleration levels of less than 10^{-4} Earth g's and by values of specific impulse that are factors of 5 to 50 times higher than a chemical system. Because the thrust period is extended over a much greater arc length during the transfer phase, the gravity losses are much greater than those associated with chemical propulsion, i.e., a greater total ΔV must be provided by the low-thrust propulsion system. These losses are more than compensated for by a much higher specific impulse, however, and a fair increase in performance is generally realized by these systems.

However, as discussed in the following subsections, there are significant technology availability and applicability issues associated with the propulsive technologies supporting these low-thrust trajectories. Subsection A.1.3.1 examines each of these propulsive technologies, discussing their readiness and attributes.

Subsection A.1.3.2 describes the type of low-thrust trajectories supported by these propulsive technologies and what supplemental propulsion and/or power measures might be necessary to satisfy mission objectives at Saturn's vast distance from the Sun. Subsection A.1.3.3 summarizes the technology issues of A.1.3.1, as well as the Saturn solar-distance implications brought out in A.1.3.2 and, where appropriate, ties them to the solar power issues examined in Section 4.

A.1.3.1 Low-Thrust Propulsion Potential Alternatives. Low-thrust propulsion systems have been extensively studied over the last 30 years for use on planetary missions. These systems include Solar-Electric Propulsion, Solar-Thermal Propulsion, Nuclear-Electric Propulsion (NEP), and Solar Sail. None of these low-thrust systems has, as yet, been used as a primary means of propulsion for any planetary mission; however, there has been an extensive technology development effort devoted to the various parts of these systems both at JPL and at other NASA centers. This technology effort has been focused particularly on SEP.

A. Solar-Electric Propulsion

This low-thrust propulsion potential alternative is the most mature and best understood of the low-thrust systems that have been investigated. A variety of electric propulsion thrusters have been flown in space by the U.S. and other countries over the past 30 years, but electric propulsion has never been used as primary propulsion in planetary missions. The propulsion potential alternative examined in this study uses large solar arrays to provide electrical power to a number of modular electric bombardment thrusters, using xenon as a propellant. A significant limitation for this type of propulsion system is that it can be operated only when sufficiently close to the Sun (less than 3 to 3.5 AU for the design used in this study).

One candidate for powering an SEP stage would be the Advanced Photovoltaic Solar Array, recently developed at JPL and TRW. Also under development at JPL, Hughes Aircraft, and the NASA Lewis Research Center, are xenon thrusters capable of being integrated into a modular propulsion system powered by solar arrays. The thrusters produce thrust by ionizing the xenon propellant and electrostatically accelerating it out of the spacecraft. Key areas still to be addressed in order to develop SEP to the point of flight readiness include thruster life testing, the development of the power processing and control system, the design and manufacture of the propellant tanks and supporting structure, and methods for dealing with large quantities of high-pressure or cryogenic xenon propellant. Several of these issues were discussed previously in Subsection 4.3.3.

B. Solar-Thermal Propulsion

This propulsion concept uses large solar concentrators to heat hydrogen or some other working fluid, which is then discharged through a nozzle to produce thrust. Dual-axis articulation of the concentrators is generally required in order to allow the engine nozzle to be pointed in any direction, while the solar radiation remains focused on the working fluid (the solar radiation is directed through a window or an open aperture in the engine). This system delivers higher thrust than

SEP and comparable specific impulse, but, like SEP, can only be used fairly near the Sun.[1] Given this restriction, STP could not be used as the sole propulsion system for Cassini. For Saturn orbit insertion or any large maneuvers in the outer solar system, a separate chemical propulsion system would have to be used. Also, the solar concentrators are designed to focus the solar radiation for propulsion, not energy generation. It is likely that some other source of energy would still be required in order to power the spacecraft.

Solar-thermal propulsion has been studied by the U.S. Air Force and JPL, and ground tests of an experimental solar-thermal cluster designed by Rockwell International's Rocketdyne Division have been conducted. Ground tests of heated hydrogen thrusters are in progress at Edwards Air Force Base. A moderately substantial program of technology development would still appear to be necessary, however, in order to employ this propulsion system. Performance of a solar-thermal propulsion system is strongly contingent on the feasibility of inflatable solar concentrators, whose deployment and use have yet to be tested in space. In addition, as of this writing, no solar-thermal thrusters have been flight-tested.

C. Nuclear-Electric Propulsion

Nuclear-electric propulsion has been examined in the past for missions similar to Cassini. In this propulsion concept, a small nuclear reactor of 100 to 300 kW_e, is used together with high-power ion thrusters similar to those used in an SEP system. This propulsion potential alternative would not be ready, at the earliest, before the beginning of the next decade due to the remaining development work needed to qualify a suitable nuclear reactor.

D. Solar Sail

Another low-thrust propulsion potential alternative evaluated for Cassini is the use of a solar sail. Thrust would be produced by momentum transfer from sunlight falling upon-a large, flat, very lightweight membrane. This technology concept was examined in detail during the late 1970s for a mission set that included a post-perihelion rendezvous with the periodic comet Halley (Sauer, 1976 and 1977), and had a membrane with an area of around 0.6 to 0.7 km^2. Although several conceptual designs exist, no solar sail has ever been used as primary propulsion, either for near-Earth or planetary applications. Solar sail requires no propellant, leaving that much more of the mass allocation available for payload. However, because the momentum impeded by the incident sunlight attenuates in inverse proportion to the square of the heliocentric range, the effectiveness of this technique is reduced as solar range increases—at some point it becomes increasingly difficult to maneuver the spacecraft. Also, the solar sail does not provide electrical energy; an electrical power source of some kind would still be required.

[1] In Sercel, 1985, use of STP was restricted to a heliocentric distance less than 1.2 AU.

A.1.3.2 Low-Thrust Trajectory Potential Alternatives. Low-thrust trajectory potential alternatives, corresponding to each of the propulsive technologies discussed above, have been extensively investigated for application to Cassini and similar missions. Due to the technology availability problems described in Subsection A.1.3.1, solar array technology applicability issues discussed in Section 4, and Saturn solar-distance constraints to be discussed below, these trajectories are not included in Section 3, Cassini Mission Alternatives. They are, however, briefly summarized here for documentation purposes.

A. Solar-Electric Propulsion Trajectories

Extensive studies of planetary missions using Solar Electric Propulsion have been done in the past (Jones, 1984 and Sauer, 1987). Based on this work, three basic SEP trajectory modes were considered for Cassini: SEP direct, SEP indirect, and SEEGA (Solar-Electric Earth Gravity Assist). The SEP direct trajectory could conceivably accomplish the transfer from Earth to Saturn in less than a full revolution around the Sun, while the SEP indirect trajectory makes more than one revolution around the Sun before final SEP thrust termination. The SEEGA, which uses a gravity-assist swingby of Earth, is an extension of the ΔV-EGA trajectory. Solar-electric, gravity-assist trajectories using planets other than Earth have not been investigated, but are likely possible. In tact, as with the other propulsion potential alternatives, they would probably exhibit better performance than the corresponding cases performed with chemical rocket engines.[2]

1. SEP Direct and Indirect Trajectories

Potential SEP direct and indirect Cassini trajectories exist, but because they are characterized by poor performance, do not appear to be feasible alternatives. The benefit of a SEEGA trajectory mode, by comparison, is that the Earth gravity-assist swingby reduces the propulsive capability required in an SEP system.

2. SEEGA Trajectories

SEP Earth gravity-assist trajectories are characterized by continuous SEP thrusting from launch through Earth swingby and continuing until slightly beyond 3 AU from the Sun. This trajectory mode requires a very low launch energy, only slightly more than the minimum required to escape Earth. Following final SEP thrust termination, the SEP propulsion module would be jettisoned and the remainder of the mission would be performed using the spacecraft chemical propulsion system. This system could be sized smaller than the one for an all-chemical propulsion potential alternative because it would not have to handle the relatively large deep space maneuver required for a ΔV-EGA transfer trajectory.

The most desirable SEEGA type trajectory for Cassini would utilize a Jupiter gravity assist in addition to an Earth gravity assist. For launch years when a Jupiter gravity assist is not possible, it might still be possible to perform a SEP

[2] The same considerations discussed in Section 4 relating to the size and use of arrays for the post-encounter phases of the mission would apply.

Cassini mission by utilizing a 2-1/4 year Earth gravity assist or a dual-Earth gravity assist using SEP. This latter mode was examined about 10 years ago using then-current projections of SEP technology and looked promising for a Saturn orbiter mission. This dual-Earth, gravity-assist trajectory uses a low launch energy, 1-l/4 year Earth gravity assist coupled with a longer 2- or 3-year Earth gravity assist, enabling the spacecraft to reach any of the more distant outer planets with good injection perfommance.[3] The time required from launch to the second Earth swingby is greater, however, than that needed for a more conventional three-year-plus ΔV-EGA trajectory. Also, as discussed in Subsection 4.3.3, SEP would entail most of the same problems as the all-solar spacecraft designs. To the extent that these problems would necessitate the use of smaller arrays, RTG augmentation would likely be necessary. This would leave SEP, in general, as a higher cost, higher risk means of achieving a non-Earth gravity assist. For the SEEGA trajectory, which use an Earth swingby, not even this would be the case.

3. Saturn Solar-Distance Considerations

Because power output from solar arrays decreases rapidly with heliocentric distance, there is insufficient power to operate SEP thrusters beyond about 3 to 3.5 AU from the Sun. Consequently, a Cassini spacecraft designed for solar-electric propulsion would also require a chemical propulsion system capable of large maneuvers for use in the encounter phases of the mission. The range of Saturn distances from the Sun during the nominal tour, June 2004 to June 2008, for example, is 9.0 to 9.3 AU, substantially past the SEP limit.[4]

B. Solar-Thermal Propulsion Trajectories

Trajectories using solar-thermal propulsion for a Cassini-like mission were investigated (Sercel, 1985). For this mission, the flight time was found to be comparable to SEP, although because of STP's higher thrust levels, the thrusting interval was much shorter; approximately one month for STP versus more than one year for SEP. As mentioned earlier, the restricted solar range over which this propulsion system is effective necessitates the use of an additional, all-chemical propulsion system to perform any large maneuvers, including rendezvous, which occur in the outer solar system. It is likely that planetary gravity assists could be used in conjunction with STP in order to improve performance. If the spacecraft were powered by RTGs, however, Earth gravity assists might have to be restricted in order to constitute an environmentally distinct alternative to the baseline.

C. Nuclear-Electric Propulsion Trajectories

A typical NEP trajectory to an outer planet would have a single thrust/coast/thrust trajectory arc and would not require a planetary gravity assist. Neglecting the fundamental issue of technology availability, the Cassini mission could probably be accomplished using an NEP system, and there would be no need for a separate chemical propulsion system. In fact, past mission studies (JPL 1986) have shown that an NEP system has the potential capability of delivering far more

[3] See Subsection 4.3.3 for further details on the feasibility of a Solar Electric Propulsion suboption.
[4] Ibid.

payload than the chemical engine potential alternative for Cassini. The use of an NEP system would also result in a large excess of electrical power available for science at the target. However, as discussed earlier, this propulsion potential alternative will not be available until a suitable nuclear reactor has been qualified. Such reactor readiness is not anticipated until the beginning of the next decade at the earliest.

D. Solar Sail Trajectories

Solar sail trajectories were extensively studied for both comet and outer planet missions about 10 years ago. These sail trajectories, like the NEP trajectories, did not involve the use of planetary gravity assists. Use of a solar sail, however, would require a chemical retro-propulsion system for the Cassini mission if the mission were to be accomplished in a reasonable length of time, because the transfer time required to actually rendezvous with Saturn using a sail would be two to three times as long as the time to just fly by the planet. Typical solar sail trajectories for a Saturn mission have been documented in an unpublished report (Wright, 1980).

A.1.3.3 Practicality of Low-Thrust Systems

In order for any of these systems to be considered for Cassini, they must exhibit a level of technical availability/applicability consistent with meeting science goals in accordance with the Program schedule. Only SEP and STP, assuming significant additional development funding, appear to offer marginal prospects for achieving such readiness. However, as shown in Section 4, the mass of a solar-powered Cassini spacecraft would be prohibitively large. In addition, Saturn's vast distance from the Sun necessitates that all of the solar-dependent 1ow-thrust systems be augmented with chemical propulsion systems for near-Saturn maneuvers, diminishing or eliminating any payload mass benefits accruing from the high specific impulse of low-thrust systems. For these reasons, low-thrust trajectories are not considered further as potential mission alternatives.

A.2 POTENTIAL LAUNCH SYSTEM ALTERNATIVES

The following subsections examine the various boosters, upper stages, and launch vehicle systems under consideration for future planetary missions, classifying them as existing and proposed U.S. and foreign launch vehicle systems. Discussion focuses on “large” and “heavy” class launch systems, as the present performance requirements for large planetary orbiter missions have surpassed the performance ceilings of smaller classes of launch vehicles. Evaluation is based on Subsection 2.2.1's definition of technically feasible alternatives. As a result of this study's evaluation process, only the following integrated launch systems are considered viable potential alternatives for the Cassini Project:

- Titan IV (SRMU) with Titan Centaur
- Titan IV (SRM) with Titan Centaur
- Space Transportation System (STS), also known as the Space Shuttle, with either an IUS or with an IUS and PAM-S.
- Two launches of the STS with an as yet undeveloped powerful upper stage, followed by upper stage/spacecraft assembly in low Earth orbit.
- Ariane 5 with Centaur IIA upper stage.
- Energia with the Energia Upper Stage and/or the Retro and Correction Stage.
- Energia-M with Centaur IIA upper stage or Block 'DM' + Star 63F.
- Proton-M with Block 'D' + Star 63F
- Split mission using two Proton launches, sending two smaller orbiters (one with the Huygens probe) to Saturn.

Table A-1 is a summary of the overall characteristics of large and heavy lift boosters for planetary missions.

A.2.1 System Considerations

The launch system for a planetary mission comprises a booster and a compatible upper stage. Typically, the booster operates from the ground to insert the upper stage/payload combination-into a desired parking orbit, and the upper stage(s) injects the payload system from the parking orbit into the desired interplanetary trajectory. But in certain instances, when the booster is incapable of placing a fully loaded upper stage/payload combination into a parking orbit, a part of the propellant for restartable liquid upper stages is used to accomplish this. For example, the Titan IV (SRMU)/Centaur launch vehicle, to be used for planetary orbital missions, requires a portion (~20%) of the Centaur propellant to place the Centaur/spacecraft combination into a low-Earth (e.g., 150 x 175 km) parking orbit. The rest of the Centaur propellant is used during a second burn to provide energy for interplanetary injection.

Another possible launch approach for planetary missions is to have a dual launch and assembly on-orbit. This can sometimes be accomplished by launching an upper stage and placing it in a parking orbit, then launching the

Table A-1. Large and Heavy Booster Characteristics

Booster	Typical LEO Capability, per Launch (1,000s of lb)	Stated Availability (Year)	Payload Fairing Size (meters)	Upper Stage Compatibility * (High Energy)	Possible Missions
Titan IV (SRM)	40	Current	Diameter: 5.1 Lengths: 17, 20, 23 and 26	IUS Titan Centaur	Planetary, LEO, SSO, GSO
Titan IV (SRMU)	50	Late 1995	Diameter: 5.1 Lengths: 17, 20, 23 and 26	IUS Titan Centaur	Planetary, LEO, SSO, Polar, GSO
Single STS (Shuttle)	40-50	Current	Diameter: 4.6 Length: 18.3	IUS TOS	Planetary, LEO, GSO
Dual STS with assembly on-orbit	80-85	Current	Diameter: 4.6 Length: 18.3	Multiple upper stages	Planetary
Ariane 5	46	Late 1995	Diameter: 4.6 Length: 12.0	Ariane 4 H10 Cryogenic Stage. Centaur IIA	Planetary, GSO, LEO
Energia	200-400	Current≠	Diameter: 5.5 Length: 41	EUS RCS	Planetary, LEO, GSO
Energia-M	70	Uncertain	Diameter: 5.1 Length: 21.5	RCS	Planetary, LEO, GSO
Proton	45-47	Current	Diameter: 3.7 Lengths: 4, 7.6, 8.5, and 11.35	Proton Upper Stage Block 'D' + Star 63F	Planetary, LEO, GSO

LEO: Low Earth Orbit
SSO: Sun Synchronous Orbit
GSO: Geo-Stationary Orbit

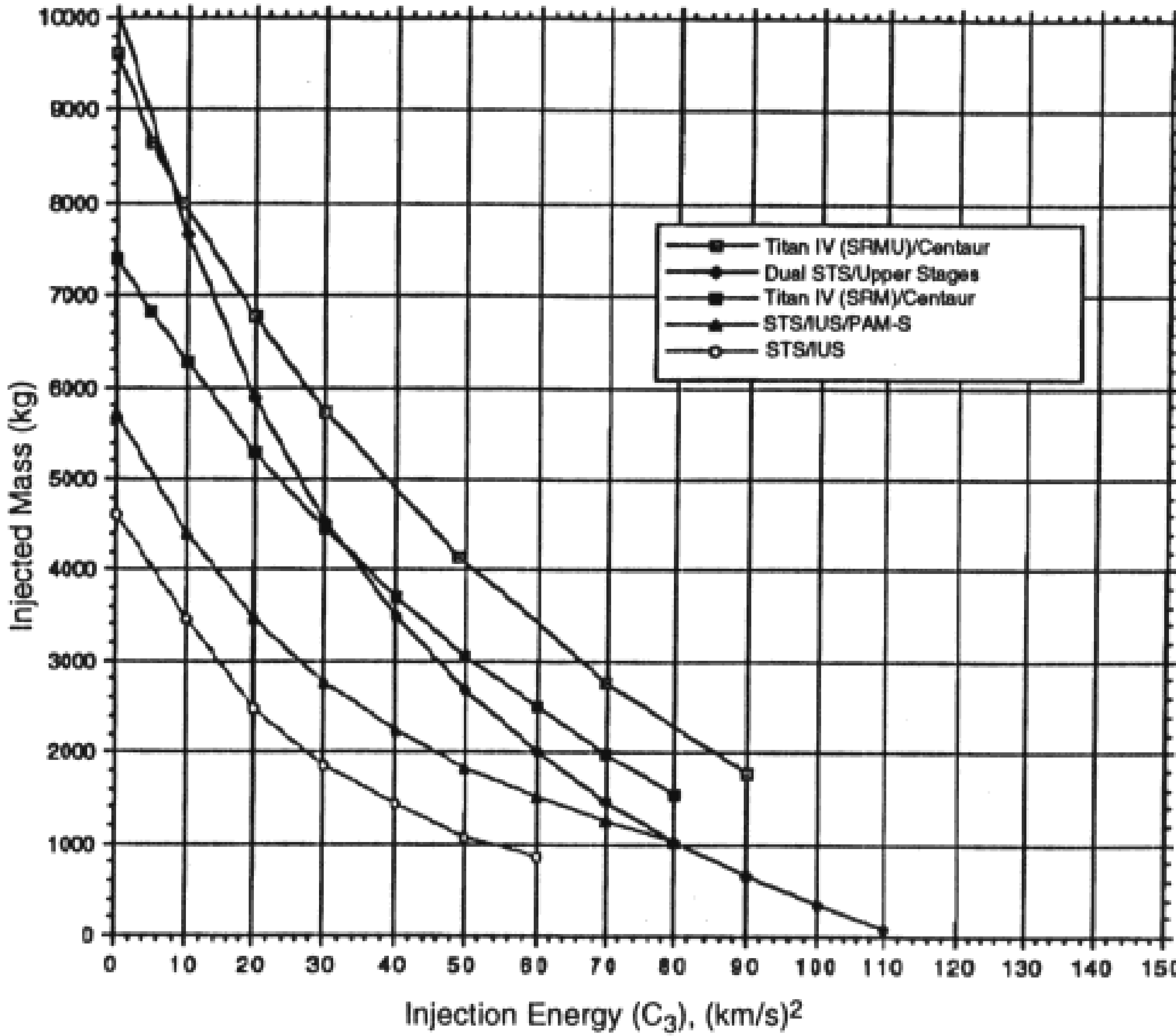
*From a performance standpoint.
≠Even though there were two test flights of the Energia before 1989, there have not been any flights since. This fact along with the problems discussed in Subsection A.2.3.2.2 call into question the actual "availability" of the booster and associated upper stages.

spacecraft and rest of the upper stage, then assembling the upper stage and spacecraft on-orbit. The Cassini Project also investigated the possibility of a dual launch where two smaller Cassini orbiters (one carrying the Huygens probe) are separately launched.

It is generally not possible to arbitrarily mix and match boosters and upper stages to deliver the payload to the desired trajectory. Upper stages are usually designed for use with certain boosters. The booster and the upper stages have to be compatible from performance and integration standpoints. It is, however, a fairly standard practice to adapt an existing upper stage for a mission by making mission-peculiar modifications as necessary to meet various requirements. For instance, the Centaur restartable upper stage on the Titan IV booster is designed for a geostationary orbit mission with three main engine burns. The same Centaur will be modified to launch planetary missions with two main engine burns.

Payload dynamic envelope size inside the payload fairing is another important consideration in the selection of a launch system. Usually, a booster vehicle is designed to accommodate payload fairings of various sizes to suit upper stage and spacecraft volume requirements. For example, the Titan IV (SRMU) could use either a 17.1-, 20.1-, 23.2-, or 26.2-meter-long (56-, 66-, 76-, or 86-foot-long) payload fairing. The volume available for the spacecraft would depend on the size of the upper stage used for the mission. The future heavy lift boosters also would be designed to use several sizes of payload fairings to accommodate various upper stage and spacecraft volume requirements.

Other criteria not listed above (e.g., cost, reliability, injection accuracy) may be important in the selection of a launch vehicle for individual missions. Figures A-2 and A-3 provide performance curves for selected integrated launch systems. The only reserve included is a three-standard-deviation (3σ) flight performance reserve (FPR). Hence, the data illustrate idealized performance. In the application of these curves, the user must allow for additional significant launch-vehicle-related and mission-peculiar reserves. The required reserves include a launch vehicle contingency reserve (LVC), a launch vehicle Project Manager's reserve, launch window and launch day wind reserves, and a mission-peculiar hardware adjustment reserve. These estimates should be subtracted from the idealized maximum performance to obtain a more realistic injected mass capability for planning purposes. For instance, at the present time, the reserves add up to about 14% of the Titan IV (SRMU)/Centaur maximum payload capability. When a launch system is in the conceptual or early design phase, mission planners typically use an allocation of 25% of its capability for these reserves.



-- CAUTION --

Injected mass capabilities include only three sigma FPR* and do not include LV* contingency, LV manager's reserve, launch window reserve, mission peculiar hardware reserve, and structural beef-up for heavy payloads.

-- GROUNDRULES --

TITAN IV (SRM)/CENTAUR AND TITAN IV (SRMU)/CENTAUR

- **Planetary Centaur configuration used**
- **ETR 93 degree azimuth launch**
- **86 x 95 nmi parking orbit**
- **66 ft payload fairing length**
- **Data from NASA LeRC**

STS/IUS AND STS/IUS/PAM-S

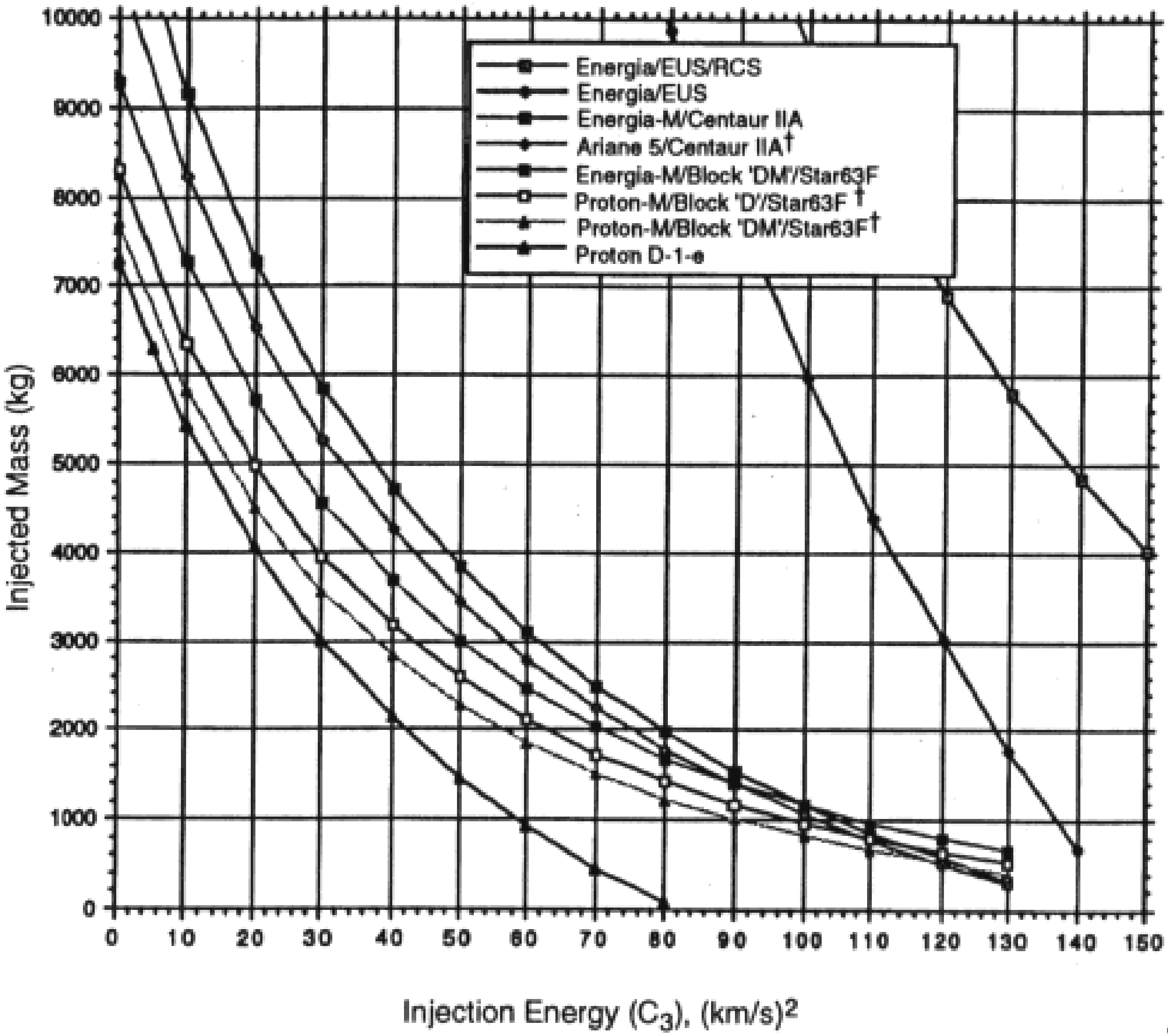
- **ETR due east launch**
- **Magellan, Galileo, & Ulysses IUS model used**
- **Ulysses PAM-S model used**

DUAL STS/MULTIPLE UPPER STAGES

- **First STS launches partial upper stage stack**
- **Second STS launches remainder of upper stage and spacecraft**
- **Second launch performs rendezvous, on orbit assembly and deployment of upper stages and spacecraft**

* FPR = Flight Performance Reserve
* LV = Launch Vehicle

Figure A-2. Performance of U.S. Launch Systems



-- CAUTION --

Injected mass capabilities include only three sigma FPR* and do not include LV* contingency, LV manager's reserve, launch window reserve, mission peculiar hardware reserve, and structural beef-up for heavy payloads.

-- GROUNDRULES --

ENERGIA/EUS AND ENERGIA/EUS/RCS
- Energia launch from Baikonur to 200 km circular, 51.6 degree parking orbit
- Energia uses four strap-ons for EUS and six or eight for EUS/RCS upper stages
- First EUS burn used for parking orbit insertion
- 25% performance reduction for uncertainty in EUS parameters

ENERGIA-M/CENTAUR IIA AND ENERGIA-M/BLOCK 'DM'/STAR63F
- Energia-M launch from Baikonur to 200 km circular, 51.6 degree parking orbit with Block 'DM' or Centaur IIA first burn
- Energia-M smaller version of Energia
- Block 'DM' is Proton current 4th stage
- Centaur IIA is Atlas IIA upper stage

ARIANE 5/CENTAUR IIA
- Ariane 5 launched from Kourou in French Guiana
- Ariane 5 second stage replaced by Centaur IIA
- Ariane 5 injection to 75 by 200 km, 5 degree inclination parking orbit
- Centaur IIA off-loaded by about 3500 kg

PROTON-M/BLOCK 'D' or 'DM'/STAR63F AND PROTON D-1-e
- Proton D-1-e is current configuration
- Proton-M will be modernized version of three stage Proton
- All Proton launches from Baikonur
- For heavy payloads a first burn of Block 'DM' or 'D' needed to insert into 200 km, 51.6 degree inclination parking orbit

* FPR = Flight Performance Reserve
* LV = Launch Vehicle
† Does not account for extra mass penalty from required larger payload fairing.

Figure A-3. Performance of Foreign Launch Systems

A.2.2 Potential U.S. Launch System Alternatives

A.2.2.1 Titan IV (SRMU) and (SRM)

A.2.2.1.1 Booster and Upper Stage

The Titan IV (SRMU), which is the current baseline launch vehicle for the Cassini mission, is the most capable American expendable booster under development, with the capacity to place about 50,000 lb (22,700 kg) into low Earth orbit. The existing Titan IV (SRM) booster, successfully flown in eight out of nine missions, has an LEO capability of about 40,000 lb (18,140 kg). It uses the seven-segment solid rocket motor as strap-one, whereas the Titan IV (SRMU) will use the more-capable, three-segment solid rocket motor upgrade. Figure A-2 shows the Titan IV/Centaur launch vehicle performance with SRM and SRMU for planetary missions. Figure A-4 shows the outboard profile of the Titan IV (SRMU).

The Centaur is a versatile, high-energy, cryogenic upper stage with multiple restart capability. It has flown in various configurations since November 1963. The Centaur baselined for Cassini is designed to fly atop the USAF Titan IV boosters, and is a derivative of the wide-body Centaur G-prime that initially was to be used with the Shuttle (i.e., prior to the 1986 Challenger accident). Figure A-5 shows a Titan Centaur schematic drawing, and Table A-2a shows its characteristics. While the Titan IV (SRMU) or (SRM) booster cannot place a fully fueled Centaur into orbit, and hence cannot utilize the Centaur's full potential for planetary missions, some useful multiple-gravity-assist Cassini trajectories are enabled by this combination.

A.2.2.1.2 Available Trajectories to Saturn

The Titan IV with Centaur has the performance capability to enable at least a minimally acceptable science return for the following trajectories and launch opportunities for the Cassini mission (see Figure 3-1):

- VVEJGA October 1997 (with SRM or SRMU)
- VEEGA December 1997 (with SRM or SRMU)
- VEEGA March 1999 (with SRM or SRMU)
- VEEGA August 2000 (with SRM or SRMU)
- VEEGA January 2001 (with SRM or SRMU)
- VVVGA March 2001 (with SRMU and a rhenium main engine on the spacecraft).
- VEEGA May 2002 (with SRM or SRMU)

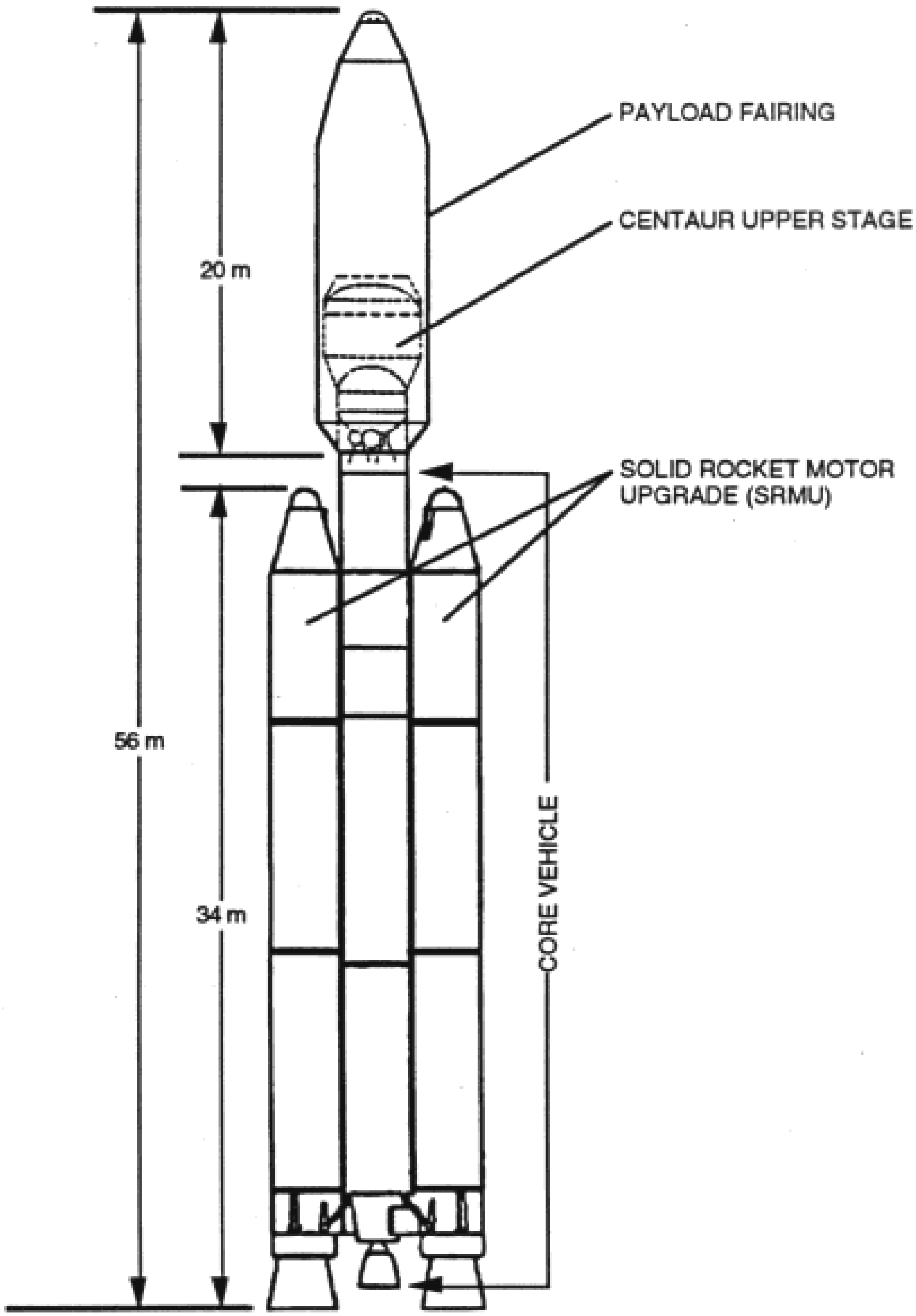


Figure A-4. Titan IV (SRMU)/Centaur Outboard Profile

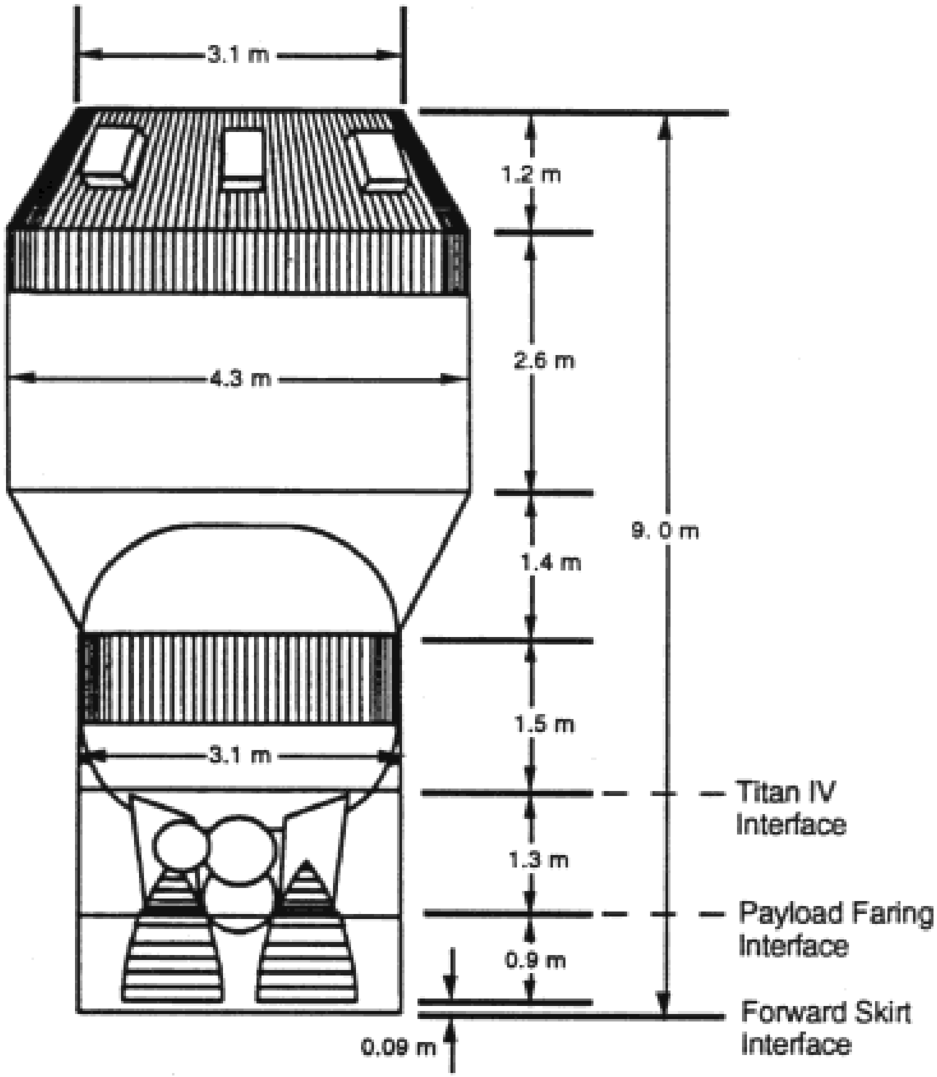


Figure A-5. Titan Centaur Cryogenic Upper Stage

Table A-2a. Upper Stage Characteristics

Upper-Stage	Description	Availability or Status	Booster Compatibility	Possible Mission	Stage Data
IUS (USAF, Boeing)	2 stage, solid, 3-axis stabilized, inertial guidance. Success rate 86%.	Current	Shuttle Titan IV	Planetary, GSO	Stage 1: M_I = 1110 M_p = 9760 I_{sp} = 292.9 Stage 2: M_I = 1150 M_p =2740 I_{sp} =300.7 Total: L=10 D=2.90
PAM-D (McDonnell Douglas)	Solid, spin-stabilized, nutation control stage (Ulysses mission-unique PAM-S).	Current	STS/IUS	Planetary, GTO	M_I = 2180 M_p = 2020 I_{sp} = 292.1 L=2.0 D=1.2
Titan Centaur (USAF, General Dynamics)	LO_2/LH_2, twin engine restartable, 3 axis stabilized, inertially guided.	Current	Titan IV	Planetary, GSO, HEO	M_I = 2770 M_p = 20,700 I_{sp} = 444.2 L=9.0 D=4.3
Proton Upper Stage 'DM' (Russian)	LO_2/kerosene single engine restartable 3-axis stabilized inertially guided.	Current	Proton	Planetary GSO	M_I = 2650 M_p = 14,870 I_{sp} = 361 L=5.5 D=3.7
Retro & Correction Stage (Russian)	LO_2/kerosene, restartable, inertially guided, 3-axis stabilized.	Unknown	Energia	Planetary, GSO	M_I = 2800 M_p = 14,870 I_{sp} = 361 L=5.5 D=3.7

GSO: GEO-stationary orbit; 35,700 km circular, i = 0°
HEO: High Earth Orbit; e.g., 100,000 km circular SIRTF orbit
GTO: Geo-Transfer Orbit
LEO: Low Earth Orbit
M_I: Inert mass of the stage (kg)
M_p: Propellant mass of the stage (kg)
I_{sp}: Specific impulse (seconds)
L: Stage length (meters)
D: Stage diameter (meters)

A.2.2.1.3 Technical Feasibility

Both the Titan IV (SRMU)/Centaur and the Titan IV (SRM)/Centaur would be available systems offering reasonable prospects of success for the Cassini mission. Also, both launch systems would be compatible with Cassini system elements and would not exceed physical constraints on flight systems.

A.2.2.1.4 SRM Comparison to SRMU Baseline

A. Non-EGA and EGA Trajectories

The Titan IV (SRM) does not enable any non-EGA trajectories for the Cassini mission.

B. Science Return

The Cassini Project Science Group has estimated that the Titan IV (SRM)/Centaur would return less science than the baseline mission, but still accomplish the mission's science objectives.

C. Concerns

Though a Titan IV (SRMU) has not been launched yet, they are being manufactured and have successfully completed their qualification program. First launch of a Titan IV (SRMU) is expected in late 1995. Additionally, the Titan IV (SRMU) satisfies all Cassini feasibility requirements for technology readiness. The Titan IV (SRM) has less performance than the SRMU, so it would not enable a non-EGA trajectory alternative and would always provide less science than the baseline. A rhenium engine would be necessary for the spacecraft to enable the March 2001 VVVGA trajectory. A rhenium engine is a spacecraft engine with a rocket chamber fabricated from rhenium and has an internal oxidation-resistant iridium coating. A version of this engine that uses mono-methyl hydrazine fuel has been in development for NASA missions. Another version of the engine, using nitrogen tetroxide, is being developed for commercial spacecraft use. Neither engine has been flight-tested, and NASA currently has no funds for the qualification or testing of the engine. Additionally, though the SRMU/Centaur (assuming a rhenium engine is on the spacecraft) could enable a March 2001 VVVGA trajectory, the SRMU is not powerful enough for a non-EGA backup opportunity in March 2002. Therefore, for the Titan IV SRMU/Centaur, only EGA opportunities after March 2001 could qualify as backup opportunities for a March 2001 VVVGA primary mission.

A.2.2.2 Single Space Transportation System Launch

A.2.2.2.1 Booster and Upper Stage

The STS, or Space Shuttle, is a crewed launch vehicle that is winged and is attached to a large external fuel tank with two large solid propellant boosters strapped to its side (see Figure A-6). The external fuel tank contains liquid oxygen and liquid hydrogen that are used to fuel the three Shuttle main engines. The Shuttle has successfully launched over sixty times with only one failure.

As a result of the 1986 Challenger accident, it was decided not to carry the high-energy, Centaur G-prime upper stage on board for reasons of crew safety. Since then, the Shuttle has flown only the less energetic, solid-propellant IUS for the Magellan, Galileo, and Ulysses (with a PAM-S upper stage) interplanetary mission launches. The IUS is a solid-propellant, two-stage vehicle. Figure A-7 depicts the outboard profiles and dimensions for the IUS. Both Galileo and Ulysses used gravity-assist trajectories due to the performance limitations of the STS/IUS launch system. The performance for this system, shown in Figure A-2, is the lowest of the launch vehicle systems described herein.

Figure A-7 also depicts a PAM-D. The PAM-S is a unique version of the PAM-D upper stage developed for the high-energy Ulysses mission. It is a one-stage solid propulsion vehicle, which in combination with the two-stage IUS provided Ulysses with its required interplanetary injection energy of about 129 km^2/s^2. The spin-stabilized stage, which uses the Morton Thiokol Star 48B solid rocket motor, was manufactured by McDonnell Douglas Astronautics Company for the Ulysses mission. Figure A-2 also includes the planetary performance capability for the STS/IUS/PAM-S launch system.

A.2.2.2.2 Trajectories to Saturn and Technical Feasibility

As can be inferred from Figure 3-1 there are no single launch, STS launch vehicle configurations that would enable the Cassini mission to Saturn. Therefore these potential alternatives will not be considered further.

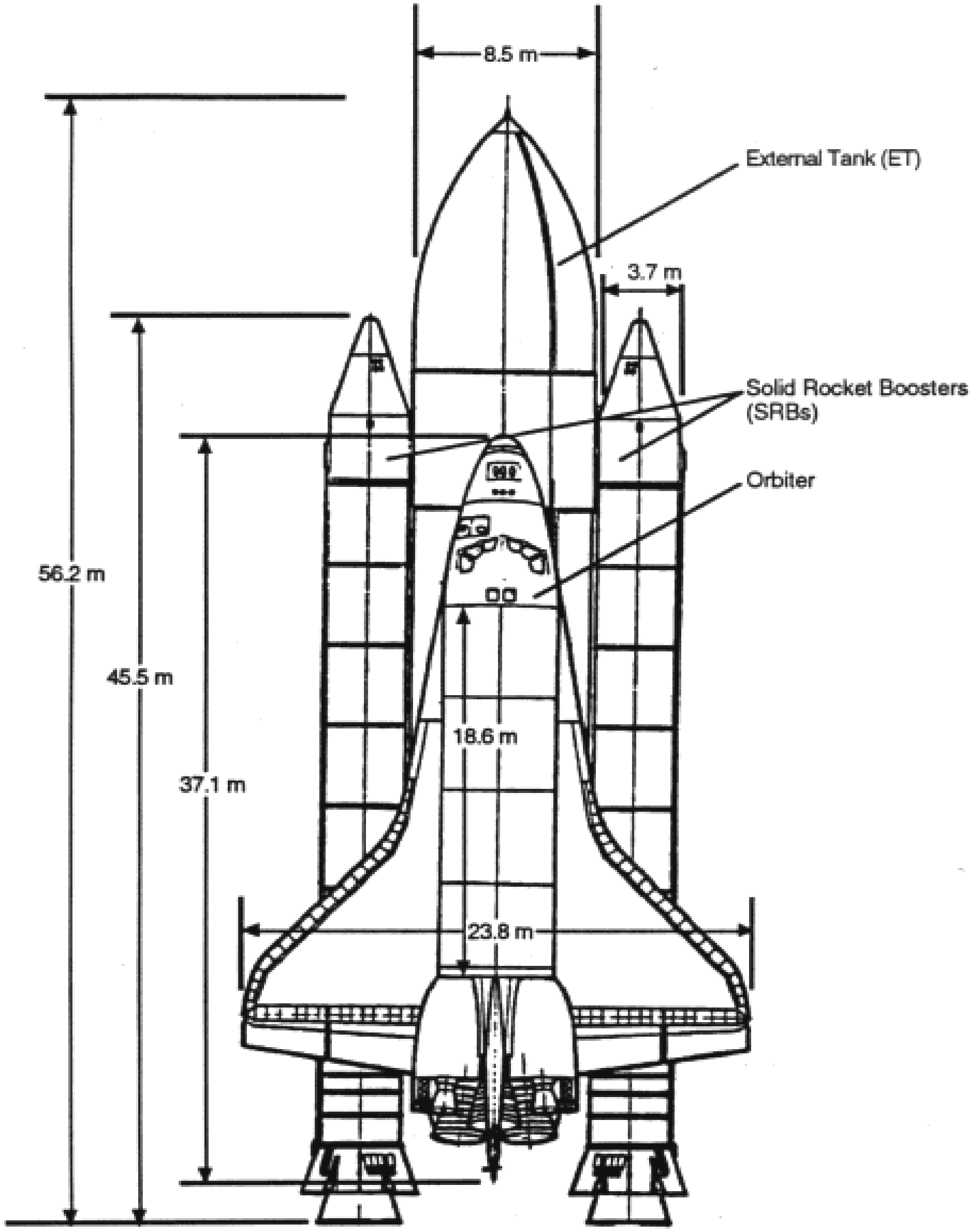


Figure A-6. Space Transportation System Outboard Profile

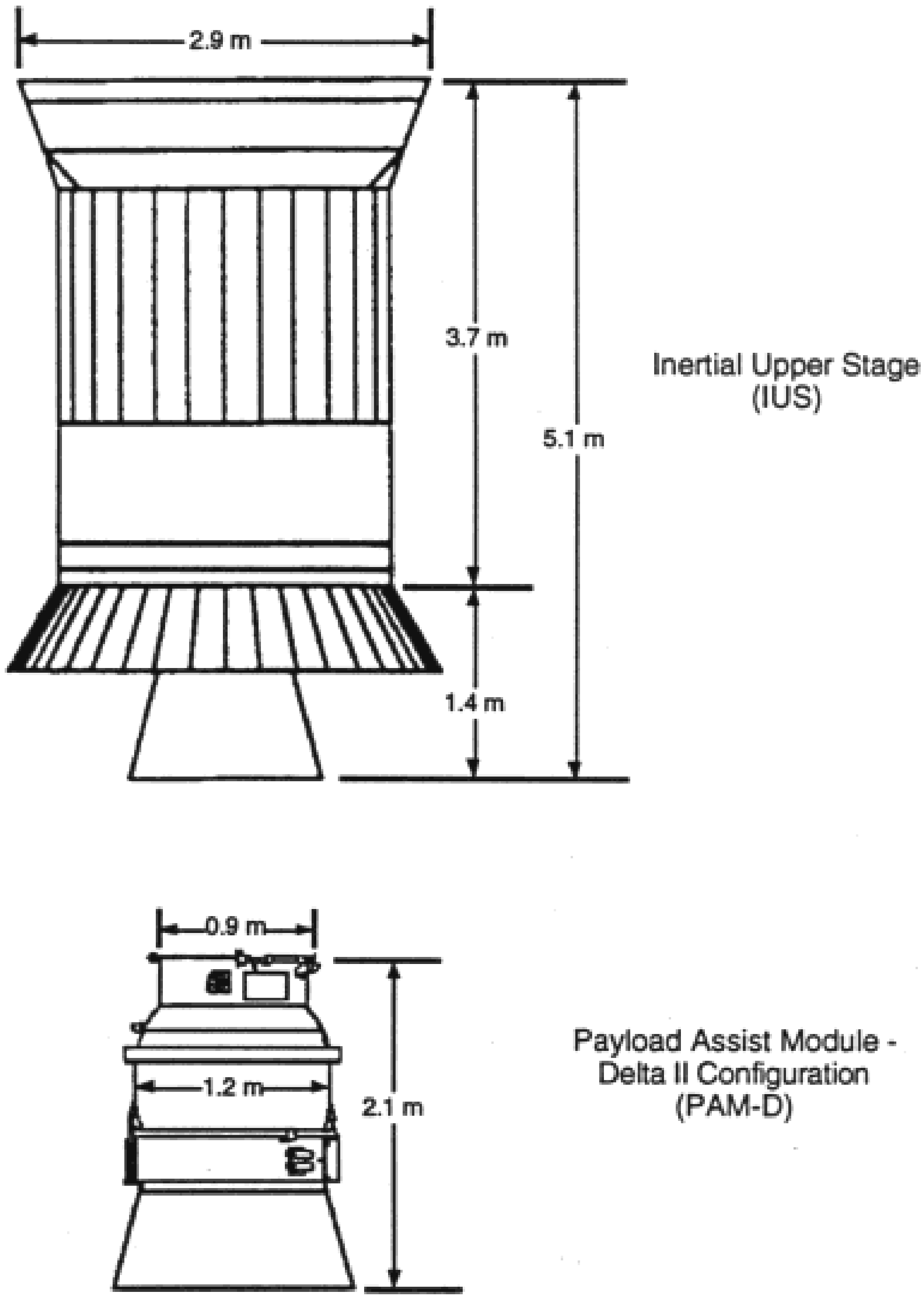


Figure A-7. IUS and PAM-D Upper Stages

A.2.2.3 Dual STS Launch with Powerful Upper Stage and Assembly On-Orbit

A.2.2.3.1 Booster and Upper Stage(s)

The STS is described in Subsection A.2.2.2.1 (also see Figure A-6). Since crew safety guidelines prohibit the use-of the powerful Centaur cryogenic upper stage on the Space Shuttle, an alternative powerful upper stage would be required if the dual STS were used for launching the Cassini spacecraft. This potential alternative would require assembling large upper stage elements while in low Earth orbit. Operationally, the elements of the stages and payload would be launched on two STS launches and then assembled in orbit by the STS crew. After assembly the stages would be ignited in a sequence that would inject the Cassini spacecraft into the proper interplanetary trajectory.

A.2.2.3.2 Available Trajectories to Saturn

This dual STS potential alternative would require first launching the partial propulsion stack into a storage orbit that is sufficiently high (due to orbital decay) so that by the time the second STS arrives (between 21 and 51 days later) in a lower orbit, the propulsion stack and Shuttle with payload (Cassini and partial upper stack) can rendezvous. The Shuttle crew would then assemble the complete stack which would include the Cassini spacecraft (see Figure A-8).

The dual STS potential alternative would enable the Cassini spacecraft to return at least the minimally acceptable science for the following launch opportunities (Figure 3-1):

- VVEJGA October 1997 (see Subsection A.2.2.3.3)
- VEEGA December 1997 (see Subsection A.2.2.3.3)
- VEEGA March 1999
- VEEGA August 2000
- VEEGA January 2001
- VEEGA May 2002

A.2.2.3.3 Technical Feasibility

Even though this launch system was determined to be 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 opportunities. Though the dual STS potential alternative appears technically feasible it is somewhat inferior to the baseline (see Subsection A.2.2.3.4).

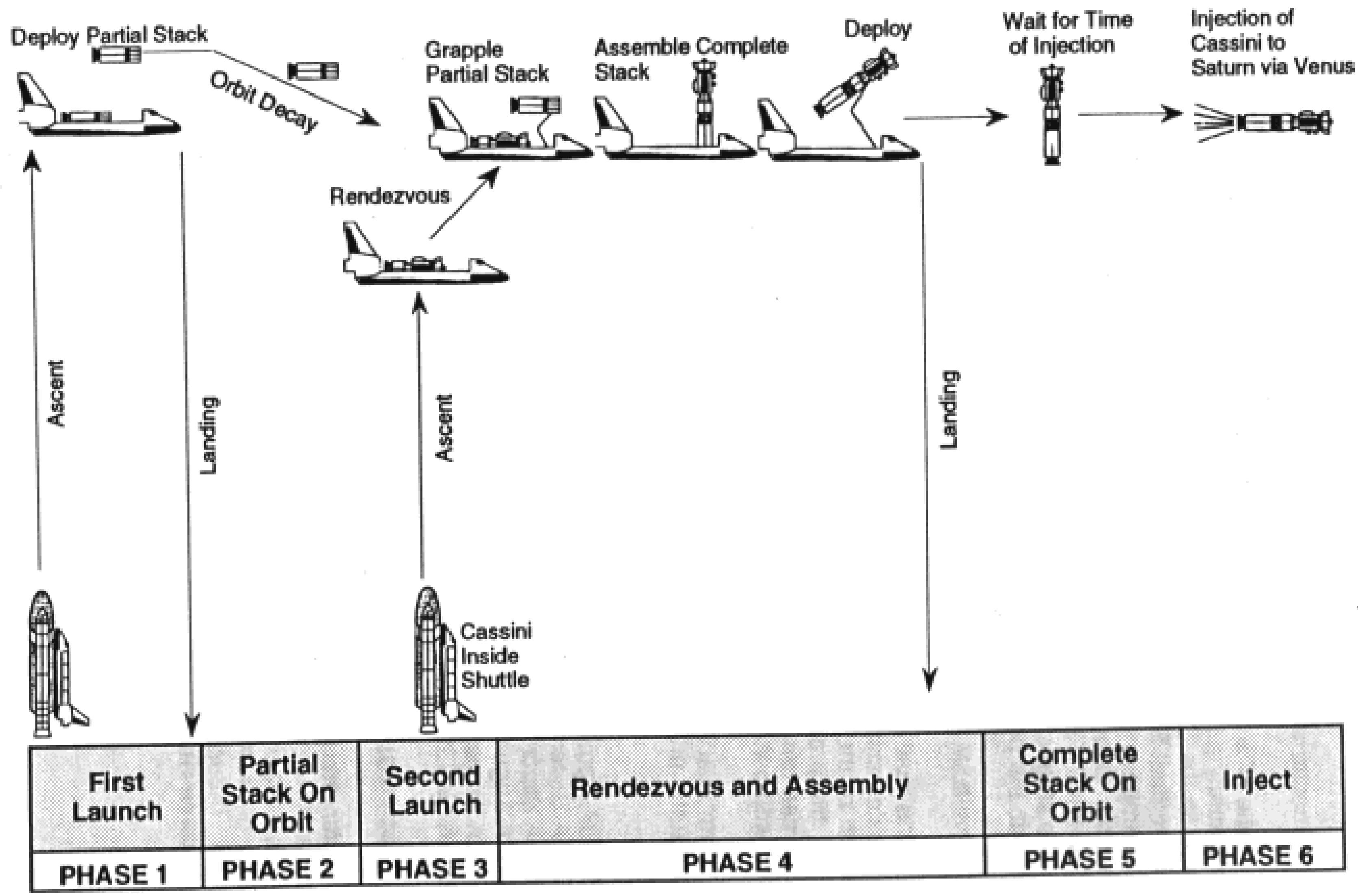


Figure A-8. Double STS Launch and Spacecraft/Upper Stage Assembly On-Orbit

A.2.2.3.4 Comparison to Titan IV (SRMU)/Centaur Baseline

A. Non-EGA and EGA Trajectories

The injection performance from the dual STS potential alternative would not provide any additional trajectories that could be environmentally more attractive than the baseline launch system.

B. Science Return

Because of temperature control requirements for the Huygens probe during the extended Earth orbital phase, the Probe coast time (between Probe release and Titan atmospheric entry) must be reduced from 22 to 15 days. This is because the Probe would heat up while on-orbit (i.e., with the Shuttle), potentially damaging probe batteries, unless the number of RHUs in the Probe is reduced. The reduction of RHUs means the probe can only safely withstand a 15-day coast to Titan. The reduced coast time results in more precise targeting for the Probe delivery into Titan's atmosphere, but a larger required orbiter deflection maneuver results in the loss of a few subsequent Titan encounters. The science return would be less than that of the Titan IV (SRMU)/Centaur case, but better than the Titan IV (SRM)/Centaur case.

C. Concerns

Though the STS reliability appears greater than the Titan IV, the Shuttle potential alternative would require two successful launches as opposed to one for the Titan IV. There is slightly more technical complexity associated with the dual STS launch system since it entails launching back-to-back Shuttle launches on time, (i.e., separated by at least 21 days, but no more than 51 days) and on-orbit assembly (taking up to 9 days) being completed before a 2-day injection period (for the primary launch opportunity). Designing and then integrating a new powerful upper stage would introduce additional technical complexity compared with the baseline.

A.2.3 Potential Foreign Launch System Alternatives

Currently, the U.S. does not have any programs funded to develop a launch vehicle with a lift capability greater than the Titan IV (SRMU). Though the following potential foreign launch system alternatives either are still in the development stage and clearly have uncertain development schedules, it is still reasonable to assess their technical capability of sending the Cassini spacecraft to Saturn. Internationally, Russian Proton and Energia launch vehicles have similar or potentially greater lift capability, respectively, than the Titan IV (SRM). However, the status of the Energia launch vehicle development program is uncertain. Information on the current status and development schedules for the Energia and the RCS and EUS upper stages were based on discussions with Russian officials during 1992 and 1993. Some of the schedule information was difficult to corroborate given conditions within the former Soviet Union.

A.2.3.1 Ariane 5 with Centaur IIA

A.2.3.1.1 Booster and Upper Stage

The European Ariane 5 launch system is being developed to replace the operational Ariane 4 series by 1999. The first flight of the more capable Ariane 5 is scheduled for late 1995. The first commercial launch is scheduled no earlier than October 1996.[5] Ariane 5 will be launched from the near-equatorial launch site Kourou in French Guiana.

The Ariane 5 is a two-stage core vehicle with two solid strap-one. The first stage of the core is a liquid-oxygen, liquid-hydrogen single engine stage. The second stage is a storable propellant single engine stage. The largest fairing size currently planned for a single launch mode is a 4.6 m diameter and 12 m length fairing. A schematic of Ariane 5 is shown in Figure A-9.

The lift capability of the system is comparable to the Titan IV (SRMU) at about 46,000 lb to low Earth orbit, inclined at 5.2 degrees. The Ariane 5 is designed to chiefly place payloads into geo-transfer orbits. Currently, there are no plans to develop any upper stages for this system. However, the Ariane 4 H10 cryogenic third stage or Centaur IIA could potentially be modified and used to inject planetary payloads. For example, the second stage of the Ariane 5 could potentially be replaced with a Centaur IIA. The performance of such a configuration can be seen in Figure A-3. Even in this case, the 10 m long Centaur IIA would be too long for the payload fairing which is only 12 m long, not leaving enough room for the Cassini spacecraft.

[5] *Space News*, “Europe's Ariane 5 to Face Series of Rocket Test Firings,” Vol.5 No.5 (April 11,1994).

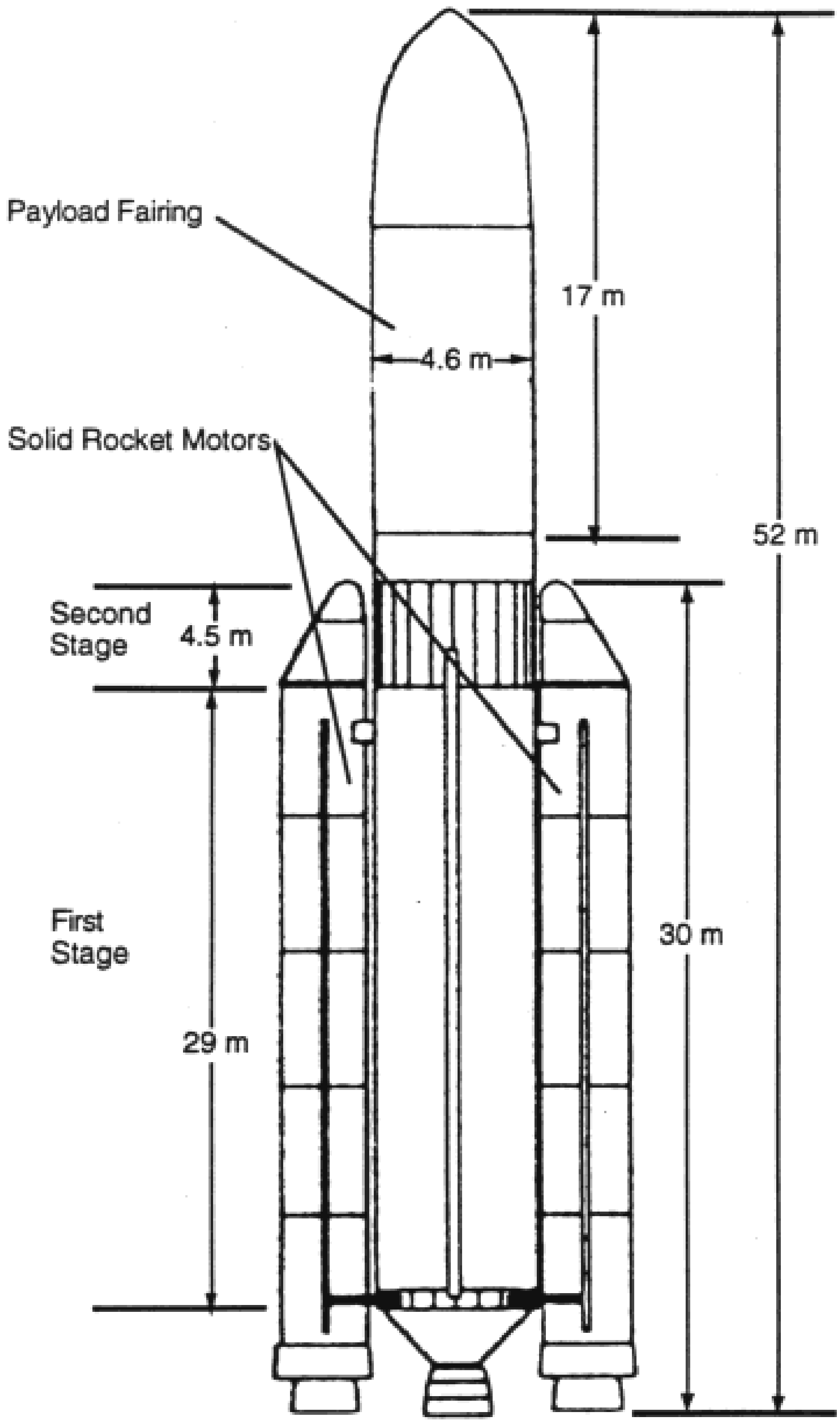


Figure A-9. Ariane 5 Outboard Profile

The Centaur IIA is the version of the Centaur cryogenic stage which currently is used by the Atlas II family of launch vehicles. Centaur was developed as the world's first high-energy, liquid-oxygen, liquid-hydrogen propellant stage and first flew in November 1963. Since then it has undergone several performance and reliability upgrades. The Centaur IIA utilizes two Pratt & Whitney RL10A-4 engines, which are available in either fixed or extendible nozzle configurations. The stage characteristics are described in Table A-2b.

A.2.3.1.2 Available Trajectories to Saturn

The Ariane 5 with Centaur IIA would enable the Cassini spacecraft to either return the full science or the minimally acceptable science for the following launch opportunities (see Figure 3-2):

- VVEJGA October 1997 (see Subsection A.2.3.1.3)
- VEEGA December 1997 (see Subsection A.2.3.1.3)
- VEEGA March 1999
- VEEGA August 2000
- VEEGA January 2001
- VVVGA March 2001 (minimum science if a rhenium main engine is used on the spacecraft)
- VVVGA March 2002 (minimum science if a rhenium main engine is used on the spacecraft and if that engine performed at its theoretical potential. Also would need larger bi-propellant tanks that may not be compatible with spacecraft design.)
- VEEGA May 2002

A.2.3.1.3 Technical Feasibility

For the Ariane 5 to be feasible, an upper stage similar to the Centaur IIA would have to be integrated with the Ariane 5. This would entail substantial analysis, integration, and qualification, including implementing major pad and operational modifications for the upper stage, necessary interfaces, and Cassini spacecraft. For example, a necessary modification would be making alterations to the pad to handle cryogenic upper stages (e.g., Centaur IIA), and for installing RTGs. The Ariane 5 PLF is too small for the currently planned Cassini spacecraft, so a larger PLF would have to be designed, built, and integrated with the future Ariane 5 booster. Regardless of the current development status and potential feasibility of this launch system, 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 opportunities.

Table A-2b. Upper Stage Characteristics

Upper Stage	Description	Availability or Status	Booster Compatibility	Possible Mission	Upper Stage Data
EUS (Russian)	LO_2/LH_2, multi-start single engine, inertially guided stage.	Unknown	Energia	Planetary, GSO, HEO	M_I = 10,500 M_p = 68,000 I_{sp} = 468 L = 16.5 D = 5.7
Centaur IIA (General Dynamics)	LH_2-LO_X restartable, 3-axis stabilized, inertially guided.	Current	Atlas IIA Atlas IIAS	Planetary, GTO	M_I = 1990 M_p = 16,600 I_{sp} = 449.5 L = 10.05 D = 3.05
Star 63F (Thiokol)	Solid propellant, spin stabilized. Would require alterations to make it 3-axis stabilized.	Current, but would need some alterations.		Planetary, GSO	M_I = 374 M_p = 4301 I_{sp} = 297.7 L = 2.71 D = 1.59
Star 48V (Thiokol)	Solid propellants, 3-axis stabilized.	Test phase	Conestoga	Planetary, LEO	M_I = 176 M_p = 2024 I_{sp} = 292.07 L = 2.03 D = 1.24
Dual STS upper stage	No set design.	Currently no official design.	Dual STS Launches	Planetary	

GSO: GEO-Stationary Orbit; 35,700 km circular, i = 0°
HEO: High Earth Orbit; e.g., 100,000 km circular
GTO: Geo-Transfer Orbit
LEO: Low Earth Orbit
M_I: Inert mass of the stage (kg)

Mp: Propellant mass of the stage (kg)
I_{sp}: Specific impulse (seconds)
L: Stage length (meters)
D: Stage diameter (meters)

A.2.3.1.4 Comparison to Titan IV (SRMU)/Centaur Baseline

A. Non-EGA and EGA Trajectories

The performance of the Ariane 5 with Centaur IIA is essentially the same as the Titan IV (SRMU)/Centaur. This launch system only possibly enables one additional trajectory over the Titan IV (SRMU)/Centaur. This trajectory, the March 2002 VVVGA could be a backup to a March 2001 VVVGA primary. However, before the 2002 VVVGA could be feasible using an Ariane 5, a rhenium engine would have to be flight qualified, integrated with the Cassini spacecraft design, and then perform better then current tests demonstrated.[6] Preliminary tests suggest performance is sufficiently below what would be necessary to enable a March 2002 VVVGA.

B. Science Return

The launch capability of the Ariane 5 with Centaur IIA upper stage (ignoring the performance penalty from the increased mass from a required larger PLF) is comparable to the Titan IV (SRMU) capability and thus could return as much science as the baseline launch system.

C. Concerns

There would be added technical complexity due to the need for integrating a new upper stage configuration for a new launch vehicle. This would entail not only integration complexities with the spacecraft, but with replacing the Ariane 5 second stage with a Centaur IIA. Even doing this, it does not appear that the Cassini spacecraft would fit in the Ariane 5 PLF. It would also be difficult to integrate the upper stage and prepare the Cassini spacecraft for launch at an unfamiliar overseas launch facility.

A.2.3.2 Energia with EUS or RCS

A.2.3.2.1 Booster and Upper Stages

The Russian Energia heavy lift vehicle is the most powerful launch system used in the last decade. The baseline Energia consists of four strap-on boosters, each equipped with four liquid oxygen/kerosene RD-170 engines, attached to a cryogenic core stage fitted with four LO_2/LH_2 rocket engines (Isakowitz, 1991; Gubanov, 1990). Whereas each of the four strap-one is the same as the first stage of the Russian Zenit launch system, operational since 1985, the core stage is new. A 6.7 by 42.1 m (22 by 138 ft) cargo container is side-mounted on the core and houses the upper stage and payload. This configuration can place approximately 205,000 lb (93,070 kg) of sub-orbital payload into a 0 by 200 km (125 mi), 52° inclined orbit.[7] Payloads can then be boosted to other orbits either by using their own propulsion systems or by an upper stage. Table A-1 shows the approximate near-orbit

[6] Note: that is also assuming all the concerns in Subsection A.2.3.1.4C are sufficiently mitigated.

[7] 0 by 200 km orbit denotes that the payload would be sub-orbital (i.e., booster alone is not powerful enough to place payload into orbit).

performance capabilities of the Energia booster (see Figure A-10), Figure A-11a shows the outboard profiles of these variants, and Figure A-11b shows the flight sequence.

Although Energia has not been launched other than in a four strap-on configuration, potential utility of the heavy-lift booster Energia with the six strap-one configuration instead of the baseline configuration of four strap-one) is shown in the performance curves in Figure A-3. Depending on the booster and the upper stage configuration, the Energia could potentially enable JGA and direct trajectories for Cassini. Such trajectories would have advantages over either the primary or backup trajectory of the baseline. Figure A-10 shows the outboard profile of the Energia vehicle with the as yet unavailable EUS and RCS upper stages.

Energia does not place a heavy payload into low Earth orbit, as the core remains sub-orbital. Additional stage(s) are needed to transfer the payloads into LEO, higher Earth orbits, or interplanetary trajectories. Russia was reportedly developing two such stages for the Energia system, the Retro and Correction Stage, also known as the Block for Transfer and Correction (BTK), and the Energia Upper Stage. Presumably, they could be used separately or together, depending on the mission, and would be designed to fit inside the side-mounted cargo container.

According to Russian statements, one of the two upper stages under development is the Retro and Correction Stage.[8] It is an LO_2/kerosene, re-startable, inertially guided stage. Indications are that it would be a modified version of the existing Proton fourth stage, to be used with the Energia booster to place payloads into low and high Earth orbit. The RCS in combination with the other upper stage reportedly being developed for the Energia booster, the EUS, can be used for planetary missions. Figure A-12 shows the schematics of the RCS upper stage. Estimates of the performance characteristics for the EUS and the RCS are shown in Tables A-2a and A-2b, and an estimated performance curve is shown in Figure A-3.

The EUS would be a multi-start, inertially guided, cryogenic upper stage (Figure A-12), advertised by the Russians as one of the upper stages that Energia's heavy lift booster could use to place large payloads into geo-stationary orbit and interplanetary trajectories (Isakowitz, 1991; Dorr, 1991). It is not known if this stage is in production or just being proposed as a concept for future high-energy missions. Based on data from Russian sources, the planetary performance capabilities for the EUS and EUS+RCS upper stage combinations with the Energia booster were evaluated. These performance capabilities were used to estimate the planetary performance curves shown in Figure 3-2.

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[8] Rockwell International, "Space Transportation Propulsion USSR Launcher Technology - 1990," June 1990, document # N91-28220.

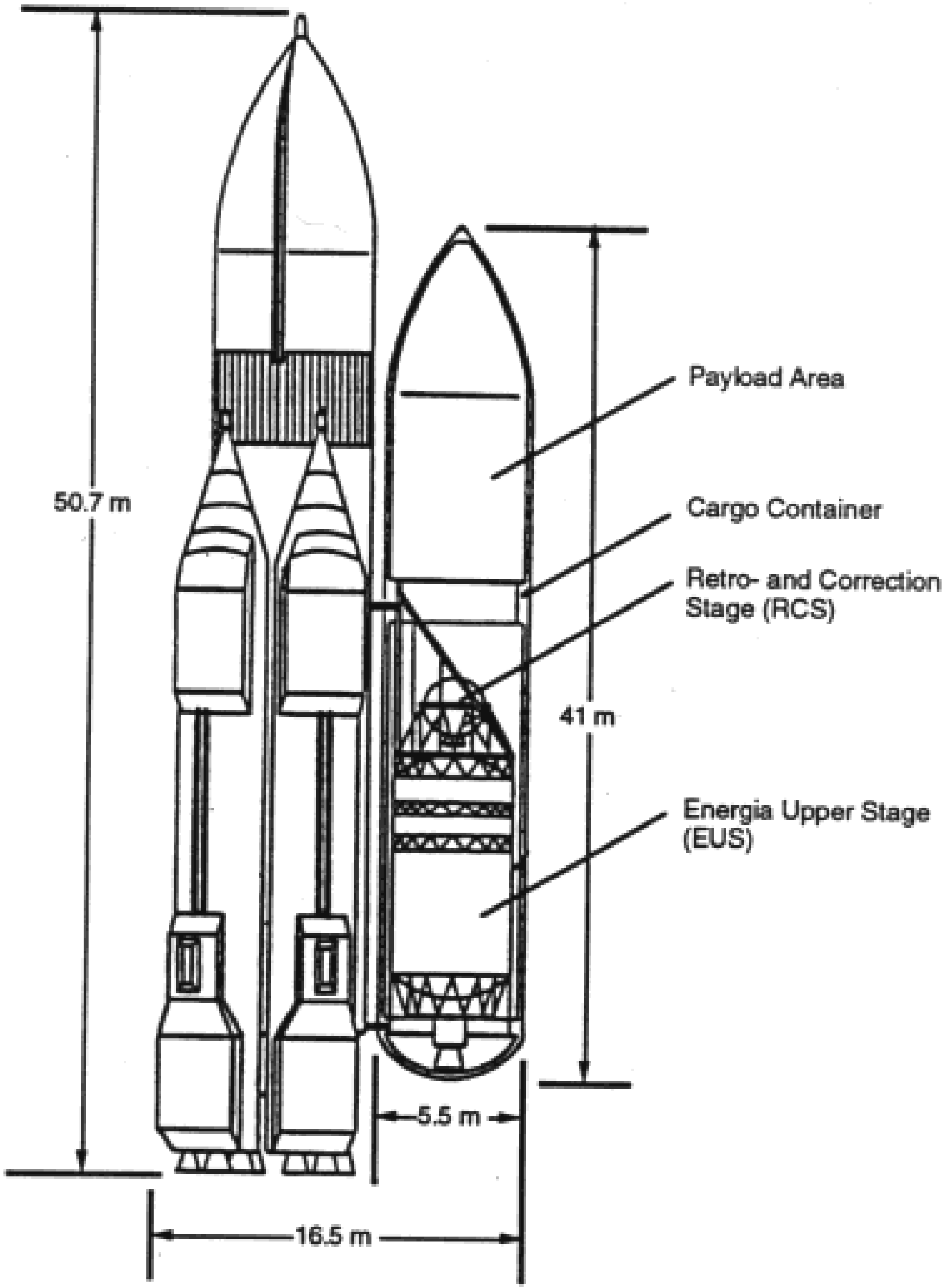

Figure A-10. Energia with EUS and RCS Upper Stages

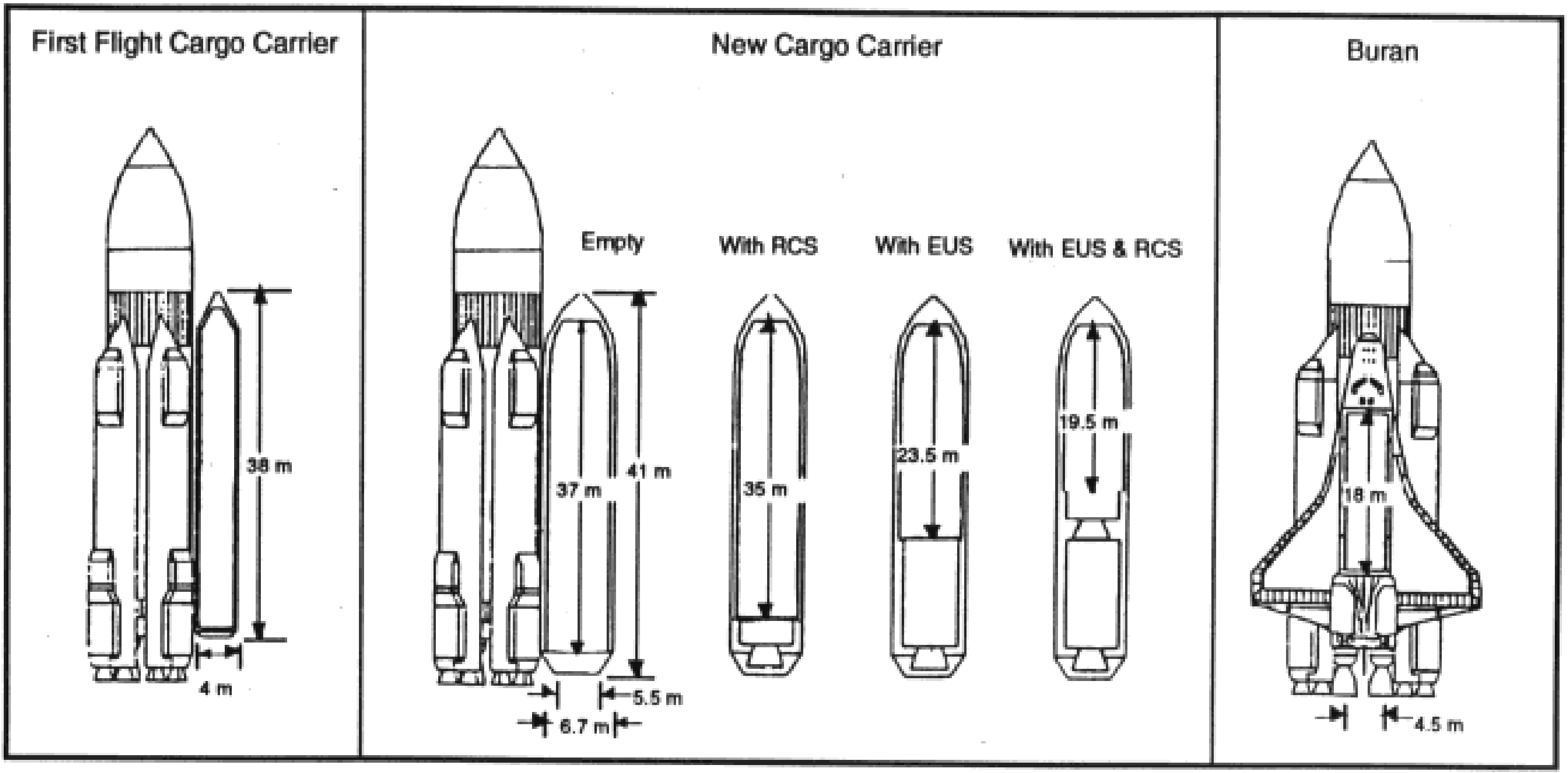

Figure A-11a. Energia Variants

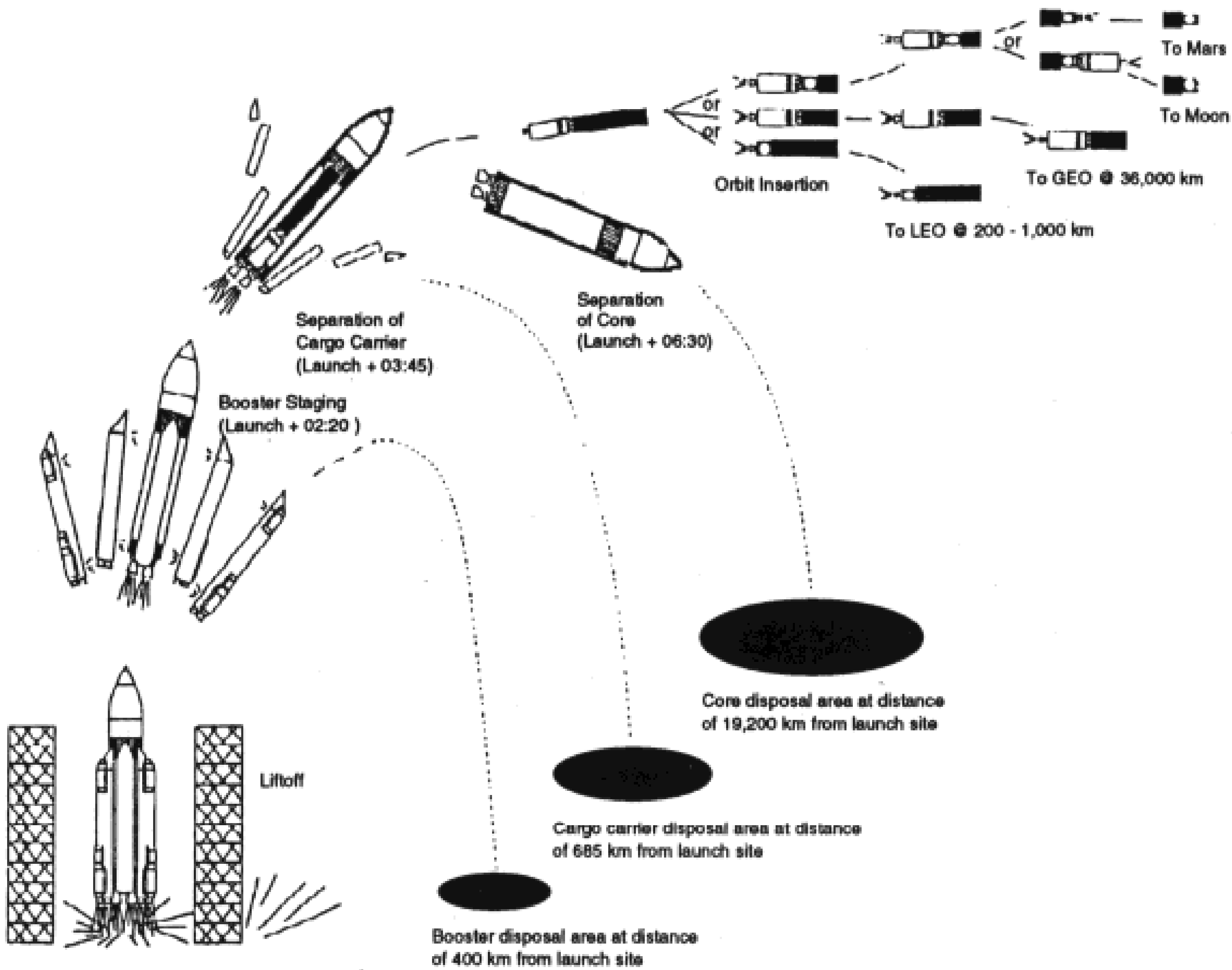

Figure A-11b. Energia Typical Flight Sequence

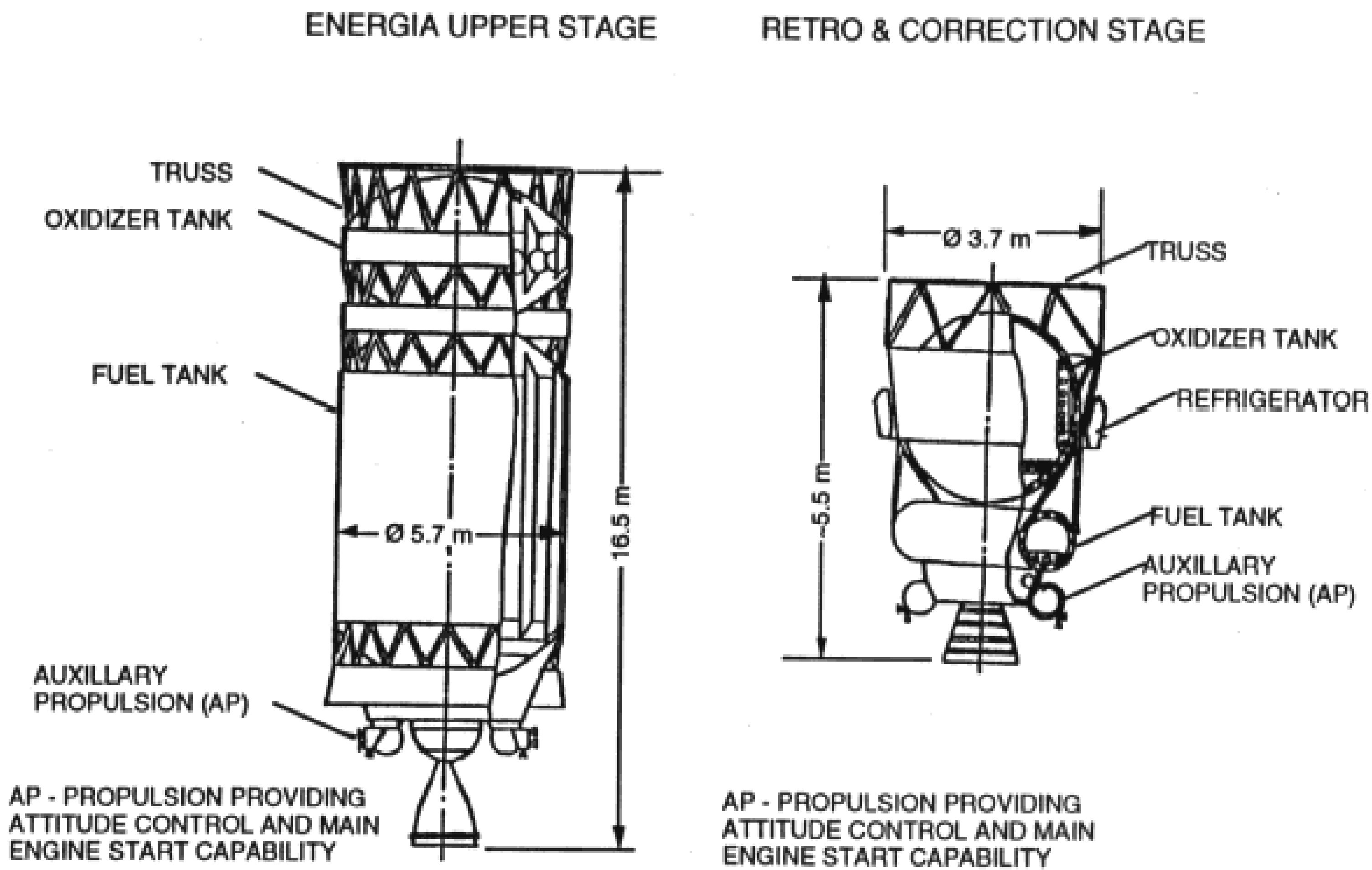

Figure A-12. The Energia Upper Stage (EUS) and the Retro and Correction Stage (RCS)

A.2.3.2.2 Technical Feasibility

Tables A-2a and A-2b summarize the respective characteristics of the various domestic and foreign upper stages discussed above. At present, there is insufficient data on technical performance to accurately evaluate the feasibility of Energia for use by Cassini. The following information would have to be determined in order to permit an informed decision on the use of Energia:

- Future status of the Energia program.
- Future plans and schedules of Energia flights for the foreseeable future.
- Dates when the Energia upper stages (RCS and EUS) will be operational, and flight schedules for these upper stages.
- Status of Energia booster derivatives, other than the baseline (four strap-on) configuration.
- Energia, RCS, and EUS costs.
- Technical details of the Energia, RCS, and EUS ascent and burn profiles.
- Assumptions and ground rules for ascent profile trajectory optimization.
- Launch time line and sequence of events.
- Accurate details of all Energia stages, plus the RCS and EUS.
- Launch operation and payload integration details.
- Payload environment details.
- Injection accuracy.
- Potential accident scenarios, environments, and probabilities.

It must be recognized that the Energia has a very brief flight history, which calls into question the long-term reliability and availability of this system. Although the flights to date were reported as being successful (in terms of booster operations) only two Energia launches have occurred. There have been no Energia flights for more than five years. (A "successful" maiden test flight of the baseline Energia occurred in May 1987 with a dummy payload, where the upper stage of the payload failed due to unknown reasons. A subsequent second flight occurred in November 1988, carrying the unmanned shuttle Buran.)

Given the above discussion regarding Energia, it may be considered a technically "potentially feasible" launch system, but it is not yet amenable to detailed comparison with the baseline due to incomplete and insufficient information.

A.2.3.3 Energia-M with Centaur IIA or Block 'DM' + Star 63F

A.2.3.3.1 Booster and Upper Stage(s)

The Energia-M is planned to be a smaller version of the already discussed Energia booster. It would use two strap-on blocks instead of four that were used with the larger Energia. Also the liquid-hydrogen, liquid-oxygen, single-engine core stage would be a scaled down version of the Energia core with four engines. It would use a 5.1 m diameter, 21.5 m length PLF in line with the core stage. A schematic of an out board profile for Energia-M is shown in Figure A-13.

The Energia-M could potentially place about 34 tons of payload in near low Earth orbit (sub-orbital), which compares to about a 100 tons capability of the larger baseline Energia. Both boosters would require an upper stage or payload propulsion system to achieve orbit. This is presumably to allow core re-entry and ocean impact, similar to the Space Shuttle external tank.

NPO Energia recently decided to maintain a core technical team for the Energia system to support the International Space Station program. A recent treaty between Russia and Kazakhstan stipulates that the Baikonur Cosmodrome be available to launch systems, including Energia, for the next twenty years. A full scale model of Energia-M underwent launch pad compatibility at Baikonur.[8] Actual development of the booster has not begun, therefore future availability is highly uncertain. The Energia-M booster is planned to be compatible with a modified version of the current Proton fourth stage and a newly designed cryogenic stage, as upper stages for geo-stationary orbit and other missions. The Centaur IIA could potentially be used with the Energia-M for the Cassini mission. However, major pad and operational modifications would be necessary to implement and launch such a configuration. The Centaur stage would be placed atop the booster and inside the 21 m long PLF. This would allow plenty of room in the PLF for the Cassini spacecraft. The Centaur IIA is discussed more fully in Subsection A.2.3.1.1. The estimated performance of the Energia-M with Centaur IIA is shown in Figure A-3. Another upper stage configuration which could be potentially easier to implement is the Block 'DM' + Star 63F. However, as can be seen in Figure A-3, the Block 'DM' + Star 63F is estimated to have worse performance than the Centaur IIA configuration.

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[8] Lenorovitz, J., “NPO Energia Assures Users of Heavy Booster’s Viability,” *Aviation Week & Space Technology*, January 24, 1994.

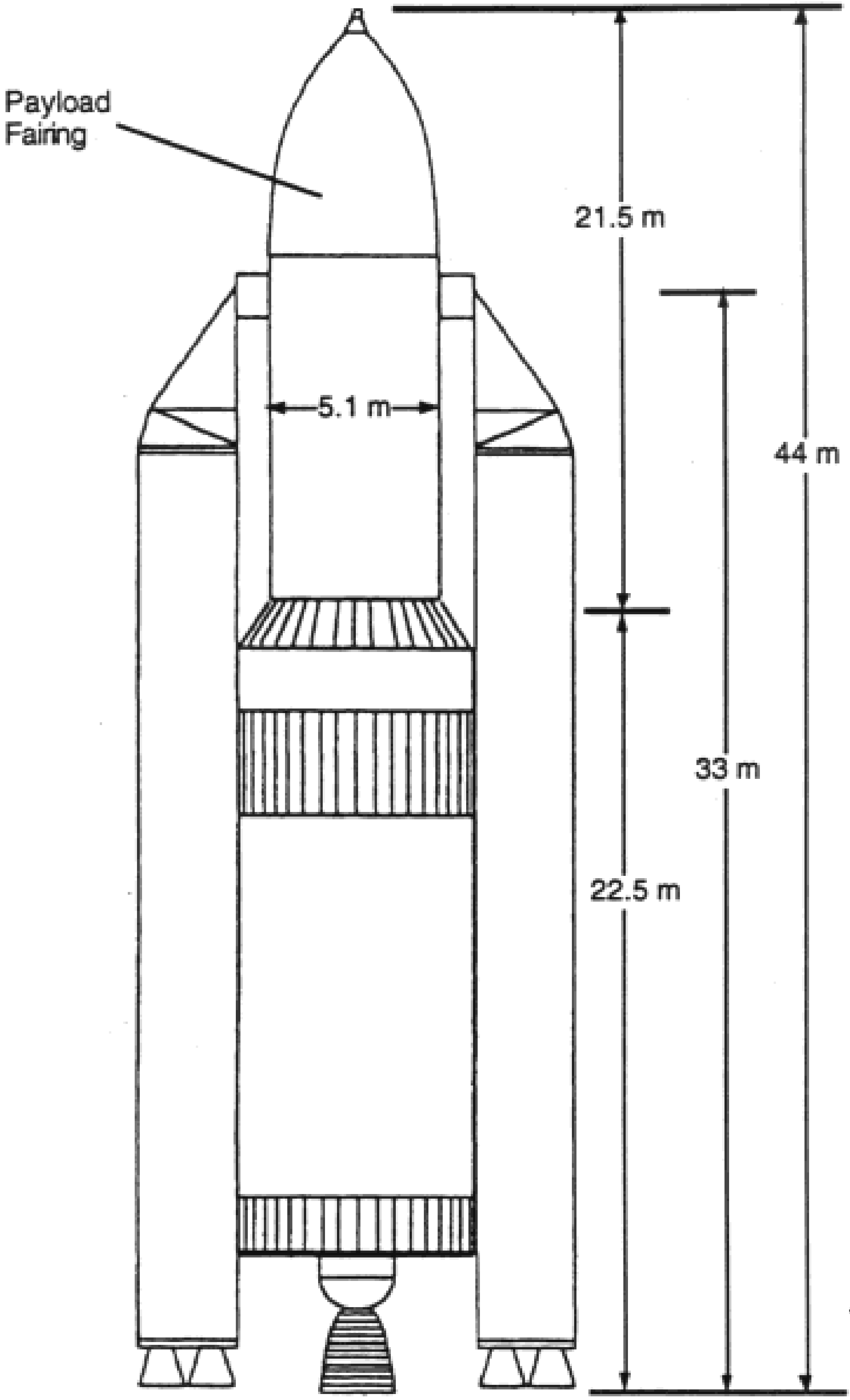

Figure A-13. Energia-M Outboard Profile

A.2.3.3.2 Available Trajectories to Saturn

The Energia-M with Centaur IIA potential alternative would enable the Cassini spacecraft to either return the full science or the minimally acceptable science for the following launch opportunities (see Figure 3-2):

- VVEJGA October 1997 (see Subsection A.2.3.3.3)
- VEEGA December 1997 (see Subsection A.2.3.3.3)
- VEEGA March 1999
- VEEGA August 2000
- VEEGA January 2001
- VVVGA March 2001
- VVVGA March 2002
- VEEGA May 2002

The VVVGA opportunities in March 2001 and 2002 would not be feasible for an Energia-M with Block 'DM' + Star 63F (see Figure 3-2).

A.2.3.3.3 Technical Feasibility

For this potential alternative to be feasible an upper stage similar to the Centaur IIA would have to be integrated with the Energia-M. This would entail implementing major pad and operational modifications, for both the upper stage and Cassini spacecraft. Additionally, regardless of the current development status and potential feasibility of this launch system, 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 opportunities.

A.2.3.3.4 Comparison to Titan IV (SRMU)/Centaur Baseline

A. Non-EGA and EGA Trajectories

The performance of the Energia-M with Centaur IIA is similar to the Titan IV (SRMU)/Centaur, and does not enable any new primary mission alternatives. This launch system only potentially enables one additional non-EGA opportunity, the March 2002 VVVGA, which would be a backup opportunity to a March 2001 VVVGA. However, this would require alterations to the baseline Cassini design since such a mission would require larger bi-propellant tanks. There would also be additional mission risk associated with a 2002 VVVGA since the cruise duration (i.e., before SOI) would be almost 12 years.

B. Science Return

The launch capability of the Energia-M with a Centaur IIA upper stage exceeds the Titan IV (SRMU) capability. Therefore, the Energia-M could return as much science as the baseline for each possible launch opportunity

C. Concerns

Only a model of the Energia-M has been made and development schedule and future availability are highly uncertain. Even if the booster were to become available, there would be added technical complexity due to the need for integrating a new upper stage configuration for a new launch vehicle. It is also reasonable to expect greater difficulty associated with spacecraft and launch system preparation if the launch were to take place at an unfamiliar launch facility.

A.2.3.4 Single Proton with Block 'D' + Star 63F

A.2.3.4.1 Booster and Upper Stages

The Russian Proton launch vehicle is an all-liquid, three- or four-stage vehicle, and has flown missions since 1965. The four-stage configuration has been used to inject large payloads into geostationary and interplanetary orbits. With a lift capability of about 45,000 lb (20,430 kg), the first three stages are used to place the fourth stage and payload into a low Earth parking orbit. The fourth stage then places the payload on an escape trajectory for an interplanetary mission (Isakowitz, 1991; Lockheed, 1993; Space Commerce, 1989).

The Proton has been offered as a commercial launch vehicle in the West. It has been a Russian workhorse launch vehicle for over 200 missions and appears to be reliable and efficient. The performance capability is shown in Figure A-3 for interplanetary missions. It is in the bottom of the Titan IV (SRM)/Centaur class of capability. A schematic representation of the current Proton D-1-e is shown in Figure A-14.

The Proton-M, would be a modernized version of the Proton D-1-e, is being developed and could be available in 1994. This version would incorporate a number of changes in the first three stages to improve the accuracy of the stage impact points and raise the performance capability to about 48,000-51,000 lb (22,000-23,000 kg) to low Earth orbit. A successful launch of this variant with some of the modifications has already occurred.

There are two versions of the Proton upper stage, the Block 'DM' and the Block 'D'. Both are single-engine, LO_X /kerosene propelled, three-axis stabilized, inertially guided, restartable stages (Isakowitz, 1991; Dorr, 1991). In operation since 1970, both versions have been used for numerous Russian geostationary, high inclination, and planetary missions. The difference between these two versions is that the 'DM' module is self-guided and controlled, whereas the 'D' module depends on spacecraft systems for guidance and control. Consequently, the 'D' module is lighter and hence has greater performance than the 'DM' module. Traditionally, the Russians have used the 'D' module for their planetary missions (e.g., Phobos

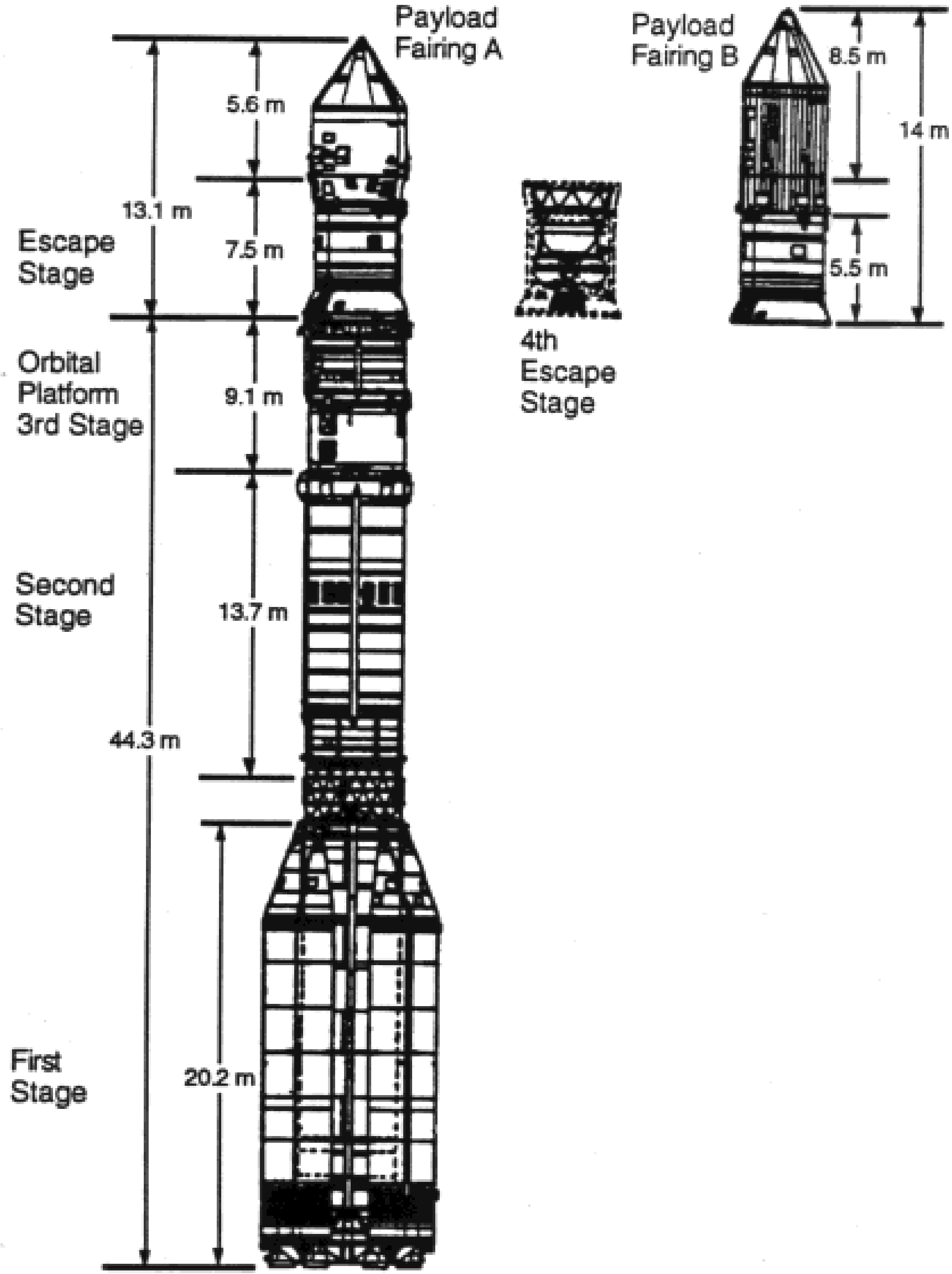

Figure A-14. Proton D-1-e

missions to Mars). The 'DM' module can inject 2200 kg into a geostationary orbit which is equivalent to an injection energy of 48 km^2/s^2. The performance difference between the two modules for a planetary mission is due to the Block 'D' dry mass being 775 kg less than that of the 'DM'. If the Cassini mission desired to take advantage of this performance, the spacecraft would have to be modified to control and guide the 'D' module.

For the Proton-M to even be a potentially feasible launch vehicle for the Cassini mission, not only would it need to use the 'D' module, but it would require an additional upper stage (e.g., a Thiokol Star 63F). The Star 63F potentially could be used as a fifth stage on the four-stage Proton-M. The Star 63F is a flight proven, Thiokol solid propellant motor with a long nozzle. The motor characteristics are shown in Table A-2b. The Star 63F is baselined as 2-axis stabilized so some modifications to make this upper stage 3-axis stabilized would be required. Essentially, the Proton-M with Block 'D' and Star 63F could provide performance comparable to the Titan IV (SRM)/Centaur launch system.

A.2.3.4.2 Available Trajectories to Saturn

The Proton-M potential alternative would enable the Cassini spacecraft to return only the minimally acceptable science for the following launch opportunities (Figure 3-2):

- VVEJGA October 1997 (see Subsection A.2.3.4.3)
- VEEGA December 1997 (see Subsection A.2.3.4.3)
- VEEGA March 1999
- VEEGA August 2000
- VEEGA January 2001

A.2.3.4.3 Technical Feasibility

When evaluating this potential alternative it was assumed that the Proton-M would perform at its maximum potential, which is an optimistic assumption. The Proton-M, to be feasible, would require development of a new upper stage configuration. Additionally a larger payload fairing would have to be developed for the Cassini spacecraft and this upper stage configuration to fit on top of the booster. Since this mission is marginally viable, even assuming maximum performance and the smaller payload fairing, it is likely that the increased mass of a larger payload fairing would make this potential alternative technically infeasible. Moreover, regardless of the current technical feasibility of this Proton launch system, there would be insufficient time to develop and implement the integration design for the spacecraft, launch vehicle and upper stages without incurring unacceptable development, integration and schedule risk for the 1997 launch opportunities.

A.2.3.4.4 Comparison to Titan IV (SRMU)/Centaur Baseline

A.. Non-EGA and EGA Trajectories

To the extent that the Proton-M with Block 'D' + Star 63F exhibits performance equal to or less than the Titan IV (SRM)/Centaur, it is incapable of enabling any alternate trajectories different from those identified for the Titan IV (SRMU)/Centaur. Hence, it would not be integral to any environmentally significant mission alternatives above and beyond those identified for the baseline launch system.

B. Science Return

If it is assumed that the necessary changes to the PLF would not make this potential alternative infeasible, then it is estimated that this potential alternative would meet the minimum science return requirement discussed in Section 2.

C. Concerns

There would be added technical complexity due to the need for integrating a new upper stage configuration and then integrating all these changes with the Cassini spacecraft. For example, compared to the baseline there would be additional technical complexity and integration work since the Block 'D' would require control and guidance from the Cassini spacecraft. The Star 63F would also require a new 3-axis thrust vector control system since the currently available Star 63F is only 2-axis stabilized. The Proton also does not have a flight termination system (FTS). Therefore, if a FTS was deemed a requirement (e.g., for safety), further technological development and integration would be required. It would also be more difficult to conduct these operations at an unfamiliar launch facility.

A.2.3.5 Split Mission: Dual Launches of Proton-M with Block 'D' + Star 63F

A.2.3.5.1 Booster and Upper Stage

This potential alternative would entail launching two spacecraft to Saturn, with both delivering orbiters (but one of the orbiters would still carry the Huygens probe to Titan). For a discussion of the Proton and its upper stages see Subsection A.2.3.4.1.

A.2.3.5.2 Available Trajectories to Saturn

This potential alternative would enable the Cassini spacecraft to return at least as much science as what is expected from the Proton-M with Block 'D' + Star 63F carrying the baseline Cassini spacecraft for an equivalent launch opportunity (see Proton opportunities in Figure 3-2 and Subsection A.2.3.4.4B):

- VVEJGA October 1997 (see Subsection A.2.3.5.3)
- VEEGA December 1997 (see Subsection A.2.3.5.3)
- VEEGA March 1999
- VEEGA August 2000
- VEEGA January 2001

A.2.3.5.3 Technical Feasibility

The Proton-M, to be feasible, would require development of a new upper stage configuration and larger payload fairing (see Subsection A.2.3.4.3). Additionally, this potential alternative would also require the use of 5 RTGs (i.e., 3 RTGs on one orbiter and 2 RTGs on the other). Additionally, regardless of the technical feasibility of this potential alternative launch system, there would be insufficient time to develop and implement the integration design for two spacecraft, two launch vehicles, and four upper stages without incurring unacceptable development, integration, and schedule risk for the 1997 launch opportunities.

A.2.3.5.4 Comparison to Titan IV (SRMU)/Centaur Baseline

A. Non-EGA and EGA Trajectories

To the extent that the Proton-M exhibits performance equal to or less than the Titan IV (SRM), it is incapable of enabling any alternate trajectories different from those identified for the Titan IV (SRMU). Hence, it would not be integral to any environmentally significant mission alternatives above and beyond those identified for the baseline.

B. Science Return

Though this potential mission alternative cannot be directly compared with the Titan IV (SRMU)/Centaur due to radical changes in tour design and ground operations, it would be expected to return approximately the same amount of science given the same launch opportunity.

C. Concerns

All the concerns in Subsection A.2.3.4.4C, would apply to two spacecraft and two launches.

A.2.4 Additional Proposed Launch Vehicle Systems

For future missions, planners may have additional launch systems to consider, some of which may offer significantly better capabilities. All would attempt to make maximal use of existing technology to reduce launch costs while improving reliability, operational flexibility, and compatibility of the launch system with both cargo and manned payloads. Furthermore, all are based on the idea of a core launch vehicle which, through modular performance improvements, can meet all the nation's medium and heavier lift launch requirements (20,000 to 50,000 lb [9080 to 22,700 kg] to LEO) in the near term. Since, however, such technologies/systems are not currently funded and do not have firm development and fabrication schedules, they are not considered feasible alternatives.

A.2.4.1 Proposed Boosters. In recent years, various advisory groups have recommended the development of a heavy lift booster that uses state-of-the-art technology, reduces costs, and simplifies payload integration and the launch process. None of these concepts has been funded for system development; the present status is confined to technology studies. The proposed availability of these new booster concepts is beyond the year 2000. No reliable performance data are currently available for these advanced concept systems.

A.2.4.2 Proposed Upper Stages. The current Atlas Centaurs and Titan Centaurs are equipped with twin Pratt and Whitney RL10A-3-3A engines for the Centaur stage. The future Centaur upper stages for the Atlas IIA and IIAS launch vehicles will utilize the new RL10A-4 engine that is under development for these programs. This engine has a thrust rating of 22,300 lb (99,190 N) and an Isp of 453.4 seconds, resulting in performance improvements for the geostationary transfer orbit. Recently, it was proposed that this improved Pratt and Whitney engine be used for the Titan Centaur stage in a single engine configuration. Studies by General Dynamics have indicated that incorporation of a single engine mode for the Centaur upper stage will improve reliability, increase payload capability, improve operability and reduce recurring cost.

APPENDIX B

ALTERNATIVE FUEL CONSIDERATIONS

NOTE:

The alternative RTG fuel analysis text and table included in this appendix were provided by the U.S. Department of Energy (DOE), in correspondence to Mr. Howard Wright, NASA HQ/Code SL, March 2, 1993.

Since the 1950s, virtually all naturally occurring and man-made radioisotopes have been evaluated to determine which are suitable to use as heat sources for thermoelectric power applications. Suitability generally requires long half-life, low radiation levels, high power density, and a stable fuel form at high temperatures. A number of radioisotopes possess some of these characteristics. Several prototypic power systems employing radioisotopes other than plutonium-238 (Pu-238) were taken to the development stage, however, Pu-238 is the only radioisotope which has been actually utilized in space power systems. Candidate radioisotopes that have been investigated include cerium-144, polonium-210, strontium-90 (Sr-90), curium-242, curium-244 (Cm-244), thulium-170, thulium-171, and cesium-137. None of these radioisotopes are suitable for current space power application because they either possess too short a half-life for long duration missions, or have sizable neutron and/or gamma emissions that result in excessive weight for the shielding required to protect a spacecraft's scientific instruments.

The baseline Cassini mission ends 10.7 years after launch in October 1997. Alternative missions could extend to 15 years and beyond. The half-life of a desirable radioisotope fuel for these RTGs should be in the range of 15 to 100 years. Of the over 1400 radioisotopes in the most current chart of radionuclides, only sixteen meet this criterion. After further sorting to select those exhibiting the other necessary characteristics described above, only three candidate fuels remain for long-duration, deep space missions: Pu-238, Sr-90, and Cm-244.

There is currently no available source of Cm-244, and its production requires reactor irradiation and chemical processing operations very similar to those for Pu-238. Additionally, there are disadvantages which make it a less attractive fuel than Pu-238, e.g., it has a shorter half-life (18 years, as opposed to 87 years for Pu-238), and its relatively high spontaneous fission, with associated neutron and gamma emissions, significantly increases shielding requirements and reduces the power-to-weight ratio of the power system, as compared to Pu-238. Therefore, it is concluded that there is no advantage to using Cm-244 instead of Pu-238, which already has a proven design and safety record.

Strontium-90 decays by beta decay to Yttrium-90, which in turn decays to a stable isotope by beta decay. Associated with the stopping of betas in the fuel form is intense Bremmstrahlung production. Once born, Bremmstrahlung are indistinguishable from gammas. The maximum Bremmstrahlung energy from the Yttrium-90 is about 2.2 MeV. Minimal encapsulation thicknesses would result in an extremely high estimated unshielded gamma dose rate that may adversely affect mission experiments, and would certainly require heavy shielding to protect personnel during RTG production and handling.

Table B-1 provides a generic study which compares the fuel characteristics of the Plutonium-238 fueled GPHS-RTG to be used in the Cassini mission with a Curium-244 fueled system and an RTG employing Strontium-90. The basis for comparison (as indicated) is thermal output at the end of a 15-year mission. Fuel loading specified in the table is that at launch time or beginning-of-life (BOL).

The technology for production of Pu-238 fuel and its use in radioisotope power systems has been clearly demonstrated over the past 30 years. Pu-238 fueled power systems have been through rigorous testing and safety evaluations; approximately $40 million to $50 million has been expended in qualifying these power systems for space applications. Any new radioisotope fuel would need to be qualified by the same rigorous methodology, which could be expected to be even more costly today.

Table B-1. Isotope Fuel Characteristics

CHARACTERISTIC	ISOTOPE Plutonium-238	Strontium-90	Curium-244
Half-life (yrs)	88	28	18.1
Emission type	Alpha	Beta	Alpha
Fuel form	Dioxide	Oxide	Oxide
Melting point (°C)	2400	2457	1950
Specific power (W/g)	0.40	0.42	2.42
Power density (W/cc)	4.0	1.94	26.1
Activity (Curies/W)	30.73	147.61	29.12
FUEL LOADING/300 W_e RTG* (Note: 3 RTGs required for the Cassini S/C)			
Thermal output (W_t)	4500	5788	7092
Fuel weight (kg)	11.250	13.781	2.931
Fuel volume (cc)	1125	2984	272
Activity (Curies)	1.38×10^5	8.54×10^5	2.06×10^5
RADIATION LEVELS (BOL)			
Gamma dose rate**	10^{-2}	~10,000	~0.9
Neutron dose rate**	4.8×10^{-2}	0	~16.5
Gamma shield thickness*** (cm of uranium)	0	10	5.6
Fast neutron flux @ 1 m (n/cm^2sec)	330	0	116,000

* Basis: same thermal output as Pu-238 15 years after fueling (3993 W_t)

** Biological dose rate at one meter from the RTG (rem/hr)

*** Shielding to reduce gamma dose rates to 10^{-2} rem/hr

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** Biological dose rate at one meter from the RTG (rem/hr)
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APPENDIX C

COMPARISON OF POWER TECHNOLOGIES

This section summarizes the comparison made between a number of electrical power generating systems which could potentially be technically available in the time frame of the Cassini mission. The only feasibility criteria addressed in this appendix are the technology readiness issues of meeting the 1997 launch date, and meeting design lifetime requirements of 12 years[1] with both sufficient power output and a viable mass for the selected launch vehicle. Results of the comparison indicate that a non-concentrating or low concentration ratio photovoltaic solar array and a radioisotope thermoelectric generator are the two leading candidate electrical power generating systems.

C. 1 INTRODUCTION TO ELECTRICAL POWER GENERATION

Essentially, all electrical power generating systems may be divided into two parts—an energy source and an energy conversion subsystem. The energy source may consist of concentrated or non-concentrated beams of photons from the Sun, chemicals for a battery or fuel cell or heat from radioisotopes, reactors, or combustion of fuels. The energy conversion subsystem takes energy from the energy source and converts it into electricity (i.e., by using photovoltaic cells, fuel cells, thermoelectric couples, or power cycles). The energy sources and energy conversion subsystems considered in this comparison are shown in Table C-1. Only certain combinations of energy sources and energy conversion subsystems are inherently compatible, and these compatible systems are marked with an “X” in the table.

Each energy source was subjected to an initial evaluation which reduced the number of potential energy sources down to a non-concentrating or low concentration ratio solar collector and a General Purpose Heat Source (GPHS). Each remaining energy conversion technology and energy source under consideration was then evaluated against the comparison criteria as a complete power system. One of the most promising of these alternative power systems (non-concentrating solar photovoltaic) is evaluated as a subsystem of a complete spacecraft design and described in detail in Appendix D. Also, a low-concentration solar photovoltaic power subsystem is evaluated in Appendix E.

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[1] The 12 years is established by Project policy and is intended to cover at least the second Titan encounter. The spacecraft may continue to collect and transmit data beyond the end of the 12-year mission.

Table C-1. Power Systems Considered

Conversion Technologies	ENERGY SOURCES: Radioisotope	Solar Non-Conc./ Conc.	Reactor	Fuels and Chemical	Power Beaming Microwave/Laser
STATIC:					
Photovoltaic		X/X			/X
Rectenna					X/
Thermoelectric	X	/X	X	X	
Thermionic	X	/X	X	X	
Alkali Metal Thermoelectric Converter (AMTEC)	X	/X	X	X	
Thermo-photovoltaic	X	/X	X	X	
Fuel Cells				X	
Primary Battery				X	
DYNAMIC:				X	
Rankine	X	/X	X	X	
Brayton	X	/X	X	X	
Stirling	X	/X	X	X	
Magneto-Hydrodynamic				X	
Internal Combustion				X	

"X" denotes compatible source/conversion technology pairs.

C.2 IDENTIFYING FEASIBLE POWER TECHNOLOGIES

C.2.1 Feasibility Criteria

The feasibility criteria used in the following discussion were selected for this summary appendix to evaluate the various energy sources and power systems in a qualitative and quantitative manner. These particular criteria were selected from the more comprehensive list of feasibility issues actually used, because they were the criteria that helped identify the power systems with the highest potential for feasibility. No significance or weighing is intended by the order of, or the presence or absence of, these or any other relevant comparison criteria. Many of these feasibility criteria are closely related and therefore are combined into the following groups:

C.2.1.1 Technical Readiness. The current Cassini baseline mission provided most of the values included in the technical readiness feasibility criteria.

- Power level: Approximately 675 W EOM (ignoring thermal heating requirements supplied by the RHUs and the RTG waste heat to the propulsion module).
- Launch date: October 1997 (all power systems must be space-qualified prior to launch).
- Lifetime: 12 years (power system must operate reliably for the specified lifetime, preferably with minimum complexity and single-point failure modes).

C.2.1.2 Launch Vehicle Constraints on the Energy Source/Power System. Assuming the baseline Titan IV (SRMU)/Centaur launch vehicle, the maximum mass for the spacecraft power subsystem is severely restricted.

The total mass of the spacecraft can be influenced by the power system in the following areas:

- Energy source/converter/storage mass.
- Additional fuel and tankage required for attitude control/propulsion.
- Thermal/nuclear radiation shielding.
- Structural requirements for stability during launch, deployment, and maneuvers.

C.2.1.3 Design Environments for the Energy Source/Power System. The range of distances of Cassini from the Sun, and the corresponding range of intensities of solar radiation, are shown below. The Cassini spacecraft must be able to operate at these distances and under these conditions.

Minimum/Maximum Spacecraft-Sun Range (AU) During Entire Mission 0.63/9.3	Minimum/Maximum Spacecraft-Sun Range (AU) During Primary Encounter 9.0/9.3
Minimum/Maximum Insolation (W/m^2) During Entire Mission 15.6/3410	Minimum/Maximum Insolation (W/m^2) During Encounter 15.6/16.7

Cassini begins orbiting Saturn when Saturn is making its closest approach to the Sun, at 9.0 AU. While Cassini will still be orbiting Saturn as it moves out to its maximum distance from the Sun (i.e., 10.1 AU), the end of the planned mission occurs when Saturn is at 9.3 AU. For the entire mission, Cassini's closest approach to the Sun occurs while it is making a Venus swingby at 0.63 AU.

C.3 CANDIDATE TECHNOLOGIES

The range of energy sources and energy conversion technologies considered is shown in Table C-1. Each column represents a current or potential near-term energy source and each row represents a current or potential near-term conversion technology.

An “X” has been placed in each element of the matrix to indicate which energy sources and conversion technologies might be compatible for the Cassini mission. One sample combination is a solar non-concentrating energy source and a photovoltaic energy conversion system that converts photon energy to electrical energy. This particular element in the matrix could include a number of designs from fold-out or roll-out deployable arrays to fixed arrays.

Only those energy sources/conversion technologies considered current or potentially mature enough to launch in 1997 were included in Table C-1. Specifically not included because of technical immaturity were:

Energy Sources	Energy Conversion Technologies
Thermonuclear fusion	Beta photovoltaics (nuclear battery)
Cold fusion	Thermally pumped laser
Matter/antimatter	Nitinol (bimetallic) engine
Solar wind utilization	Pyroelectric energy conversion
Stellar hydrogen collection	Nernst (thermomagnetic) generator
Free radical/metastable storage	

C.4 FEASIBILITY ASSESSMENT OF ENERGY SOURCES

Each energy source identified in Table C-1 is evaluated separately, using the feasibility criteria described in Subsection C.2.1. This evaluation results in the elimination of all the energy sources except the non-concentrating and low concentration ratio solar, and the GPHS. The following discussion describes what feasibility criteria were significant in the analysis.

C.4.1 Radioisotope

A radioisotope is an unstable radioactive element that generates heat and emission of various types of radiation as a result of nuclear decay. Pu-238 was selected for all previous U.S. space radioisotope energy sources, from the first system in 1961 (U.S. DOE, 1987) to the most recent in 1990. Pu-238 has a long half-life, a low gamma radiation field, and a moderate neutron radiation field, making it the preferred option for the Cassini mission. For an assessment of alternative isotopes, see Appendix B.

Pu-238 has been used in various forms and in varying quantities on a wide assortment of spacecraft, including Earth-orbiting weather satellites, lunar power systems left by the Apollo astronauts, Viking Mars landers, and missions to the Sun, Jupiter, Saturn, Uranus, Neptune, and beyond (Table C-2). Pu-238 radioisotope energy sources used for all of these missions met or exceeded their design requirements (Bennett, 1983; JPL, 1987), although the design requirements and the designs for encapsulating these radioisotopes have evolved significantly with time.

The designs of these radioisotope containment schemes have evolved through extensive analysis and testing to the present GPHS. A GPHS is a rectangular brick with four, sintered, PuO_2 fuel pellets encapsulated in an iridium alloy. There are two fuel capsules per impact shell, and two impact shells per aeroshell. The impact shells and aeroshells are made of graphite (Shock, 1980). The GPHS will be evaluated against the feasibility criteria as the only radioisotope heat source considered for the Cassini mission.

Table C-2. Radioisotope Developments

Power System Inventory Name	Mission	Pu-238 Fuel Form	Pu-238 per RTG (Curies/RTG)
SNAP-3B	Transit 4 A and B (1)	Metal	1500-1600
SNAP-9A	Transit 5BN-1,2, and 3 (1)	Metal	17,000
SNAP-19	Nimbus B-1 and 3 (2) Pioneer 10 and 11 (4) Viking 1 and 2 (2)	Plutonia Molybdenum Cermet	20,000
SNAP-27	Apollo 12, 13, 14, 15, 16, and 17 (1)	Oxide Microspheres	44,500
Transit-RTG	Triad-01-1x (1)	Plutonia Molybdenum Cermet	24,000
MHW-RTG	LES 8 and 9 (2 each) Voyager 1 and 2 (3 each)	Pressed Oxide	80,000
GPHS-RTG	Galileo (2) Ulysses (1)	Pressed Oxide	132,200

C.4.1.1 Technical Readiness. Since each GPHS provides 250 watts of thermal power, there should be no problem closely matching the thermal input requirements of almost any energy converter to satisfy the Cassini electrical power demand. The more restrictive design limit is the 1373 K maximum temperature limit of the GPHS, which was set to control grain growth in the iridium alloy fuel clads. This temperature limit constrains the power output of some of the energy converters that require higher temperatures to achieve reasonable efficiencies and also limits the lifetime of the power source, since a compromised fuel clad would reduce the safety margin if Earth reentry were to occur. If operated at or below these temperatures, there is no apparent reason why the GPHS should not satisfy the full spacecraft design lifetime; it has been space-qualified and launched on both the Galileo and Ulysses missions, and is currently operating as designed.

C.4.1.2 Launch Vehicle Constraints on the Power Source. The primary constraint the launch vehicle places on the power system is mass. GPHS is a relatively low-mass energy source, providing 173 W_t/kg. Since the Pu-238 inside the GPHS has a relatively low nuclear radiation field, the mass of any shielding required will be small. Mass of the GPHS does not present a problem for the current launch vehicle.

C.4.1.3 Design Environments for the Energy Source. The GPHS has no known design environment problems (e.g., insufficient insolation, see subsection C.2.1.3).

C.4.1.4 Radioisotope Comparison Summary. The GPHS ranks very high in all of the feasibility criteria when proper design and safety precautions are followed and will be carried into the next level of the evaluation process. It will be evaluated with various energy conversion technologies to determine the best power system for the Cassini mission.

C.4.2 Solar (Non-Concentrating/Concentrating)

From a human perspective, there are few things that are more reliable and more frequently experienced as a source of heat or energy than the Sun. The solar spectrum is similar to that of a black body at 5800 K, which produces an energy flux of 1353 W/m_2 in space at the average distance between the Earth and the Sun (1 AU). The energy delivered to a surface normal to the Sun is inversely proportional to the square of the distance from the Sun to the surface. The Sun represents a fixed point in space relative to the rest of the solar system. Thus, for Cassini to collect the maximum normal solar energy (flux) during rendezvous with Saturn and its satellites requires either tracking of the Sun or sufficient collection area to satisfy power requirements in all spacecraft orientations. Another issue introduced when using solar energy is solar eclipse—when an object comes between the Sun and the spacecraft and blocks out incoming insolation. Some type of energy storage is required during an eclipse to maintain a continuous power output.

The Sun's energy is typically collected on flat-plate, non-concentrating solar collectors (often called panels), or it can be concentrated using a reflective surface or lens (referred to as a concentrator). Both non-concentrating and concentrating solar collectors will be evaluated separately below.

C.4.2.1 Solar Non-Concentrating.

A. Technical Readiness

1. Power level

The area of a solar array is determined by the spacecraft power demand, solar cell conversion efficiency, mission duration, degradation rate, energy storage, and available insolation (which is affected by the distance from the Sun to the spacecraft and by Sun occultations). The duration of Sun occultations and the power demand profile will determine the amount of electrical storage required, which represents an excess charging power demand that must be satisfied with additional collector area. Since the solar array area is a function of the conversion efficiency, it is not appropriate to calculate a collector area until all candidate conversion technologies are evaluated.

The maximum range for Cassini from the Sun, the minimum insolation, and the power demand for the spacecraft are listed with the feasibility criteria (Subsection C.2.1). The relatively small variation of power demand on the spacecraft throughout the mission, combined with the large variation in solar flux, means the minimum acceptable collector area is determined by the power demand at the maximum solar range.

2. Launch date

Relatively small solar arrays have been used quite successfully on numerous Earth-orbiting missions and missions to the inner planets. However, for Cassini, the uncertainty lies in developing a large, space-qualified, deployable array with reasonable mass and reliability.

3. Lifetime

The operating lifetime of solar arrays in space is substantial (as demonstrated by Earth-orbiting and inner planetary missions), and the degradation mechanisms and rates are well understood for certain converter materials. Solar tracking collectors, if used, would add complexity to the spacecraft software, because they must determine where to point solar panels to collect the maximum amount of energy. Spacecraft hardware is complicated by additional structural, articulating, and driving mechanisms. Any additional complexities introduced on a spacecraft tend to reduce lifetime and/or reliability.

B. Launch Vehicle Constraints on the Power Source

The mass of an array is significantly influenced by the photovoltaic converter selected, which will be evaluated later with the other power systems. At this time it is assumed there is no obvious incompatibility between the baseline launch vehicle and a non-concentrating solar collector. The size of the collectors being estimated (from 10's to 100’s of square meters) indicates that some type of fold-out or roll-out deployable array will be required to satisfy the dimensional stow volume constraints of the launch vehicle. A number of deployable solar arrays have been demonstrated in space, but none have been fully covered with cells and of the size required for the Cassini mission.

C. Design Environments for the Power Source

Solar range, intensity, and occultations should not present any survivability concerns to panel structures with the current minimum perihelion. This environment is very benign when compared to Mariner 10, which launched in November 1973 and used solar panels at Mercury (0.387 AU), where the insolation is 9,034 W/m^2. Mariner 10's solar arrays were tilted to reduce the incident solar energy. The lowest insolation values experienced during other phases of the mission, including aphelion and during occultations, represent additional challenges for solar power generation and energy storage on the spacecraft. Combining the requirements of high and low insolation and temperature environments with a low mass requirement could also be a major design challenge.

D. Solar Non-Concentrating Comparison Summary

None of the feasibility criteria presented would eliminate a nonconcentrating solar collector from consideration at this time, which means it will be carried into the next level of the evaluation process. The collector will be evaluated with photovoltaic energy conversion technologies to determine if a feasible power system exists for the Cassini mission.

C.4.2.2 Solar Concentrating Collectors. A concentrating solar collector can be any type of reflector or lens which takes the direct beam component of insolation and concentrates the energy onto a receiver or converter surface. The very low insolation at large solar ranges experienced by the Cassini mission suggests there may be energy conversion advantages to using concentrators as an alternative to flat plate collectors. This is particularly true for certain types of conversion systems that suffer from reduced efficiency at low-intensity illumination at low temperatures (e.g., some silicon photovoltaic cells). The higher energy flux provided by concentrating collectors makes possible higher temperatures, which increases the number of energy conversion systems that may be considered and increases the efficiency of those converters that are Carnot devices.

A. Technical Readiness

The amount of power a concentrating collector can deliver to an energy conversion system is a function of the insolation, size, and efficiency of the collector for a given receiver temperature and concentration ratio. Plans exist to conduct a brief STS spaceflight experiment (early in FY96) using an inflatable paraboloid antenna: this will demonstrate the inflatable paraboloid antenna's deployment and surface accuracy. This will not demonstrate a functioning solar concentrator over the required operating lifetime. Unlike the non-concentrating solar collectors, a large, space-qualified, deployable solar concentrator has never been developed. It must be assumed that if a concentrating collector were developed of sufficient size and performance to compensate for the low-input insolation, it could deliver adequate power to a compatible energy conversion system and generate the necessary electrical energy for the mission.

Concentrating collector size, efficiency, receiver temperature, and concentration ratio are all interrelated. A high-concentration-ratio reflective collector requires a well-focused, accurately pointed, properly contoured, and highly reflective surface to maximize the concentrator's efficiency, particularly at high operating temperatures. As the size of the concentrator increases it becomes more difficult to maintain the proper contour of the reflective surface which, if not maintained, results in lost energy from the beam missing the receiver. Increasing the receiver temperature of the collector will tend to increase the efficiency of many energy conversion devices, but will reduce the efficiency of the collector. This is because the higher receiver temperature will re-radiate or lose more energy as a function of receiver temperature to the fourth power. Increasing the receiver aperture size, which effectively reduces the concentration ratio, will also increase the amount of incoming insolation; it will also increase the amount of re-radiation from the receiver, which could reduce collector efficiency and receiver output temperature.

Concentrating collectors have a major disadvantage compared to non-concentrating collectors because of their inability to use the diffuse or scattered solar energy. The Sun's energy may become scattered as a result of imperfections in or on the reflective or refractive surface. A typical low-mass, space-compatible reflective surface might be aluminized kapton, which would lose 20% of performance due to reflectivity and wrinkling, even under the best conditions (Ralph, 1990). These losses would increase significantly if an inflatable concentrator were used. Scattering may also result from dust particles accumulating on surfaces or from particles in space located between the Sun and the receiver's aperture. These factors will increase the required collector area for a given energy conversion system.

The ability of a concentrating collector to satisfy both launch date and lifetime requirements is highly questionable because concentrators have not been demonstrated in space, making the system very uncertain for a long-term, deep space mission scheduled for launch in 1997 (Stella, 1990). The first-generation space station selected non-concentrating over concentrating collectors for power generation. Development is ongoing for the station to use erectable concentrators, instead of the self-deploying solar concentrators that would be required for the Cassini mission.

Electrical power has been successfully generated using various types of concentrating collectors with many different types of energy conversion devices for terrestrial applications. Unfortunately, there are a number of significant differences in a solar concentrating power system located on Earth compared to what is required of one located in deep space. Most terrestrial power systems are not low-mass, volumetrically constrained, self-deploying, continuously operated, highly reliable, area-limited to maximize science instrument viewing angles, and required to be maintenance-free and have a 12-year lifetime. The Galileo mission antenna is an example of how the deployment of a relatively small paraboloid, which is not hampered by the added requirement of being made of a mirror-quality material capable of reflecting the solar spectrum, can present significant problems.

The amount of time required to develop and space-qualify a large concentrating collector capable of a 12-year, continuous operation lifetime is extremely difficult to predict. For instance, it would be almost impossible to obtain sufficient failure mechanism data prior to the 1997 launch date to validate the 12-year lifetime, assuming that mass, performance, and reliability goals were all satisfied. Some preliminary concepts for a concentrating solar collector system have been developed, but the lack of detailed designs and demonstration testing on the ground (or in space) makes the full impact of these issues difficult to determine.

The large number of possible concentrating solar collector designs requires that they be categorized and compared as generic types to determine which would have the highest potential feasibility for Cassini. One way to categorize concentrating collectors is whether the receiver (or focal plane) of the concentrator is a single, large, centralized area or multiple, small areas distributed over many concentrators. The geometric shape of the focal plane, whether a point, line, or plane, is another typical category that relates to the geometric concentration ratio (CR) of the collector. Finally, collectors may be divided into whether the focal plane views the Sun directly or all energy entering the focal plane arrives from a reflective surface. Table C-3 shows examples of many collector designs and how they are organized. Any of them may be scaled to be either large or small, which means that either centralized or distributed designs are possible.

To concentrate the Sun's energy requires some three-dimensional characteristics of both centralized and distributed receiver designs, but there are significant differences between the two generic types. Centralized concentrators will, by necessity, have larger envelope dimensions in all three directions, which will probably reduce the science instruments' field-of-view. The distributed concentrators will be more two-dimensional, similar to the non-concentrating solar arrays. In deep space applications there are few, if any, benefits for a distributed receiver design in comparison to a non-concentrating array: The design, fabrication, and deployment complexity will be greater, as will total mass, since the increased reflector area will not compensate for the lower area mass of the reflective surface compared to the array area mass. The area cost of a reflective surface should be lower than that of solar cells, but will probably be offset by the added area, design, and fabrication complexity. If solar cells were selected that were highly susceptible to LILT effects, concentrators would have the advantage of increasing the intensity of the solar radiation falling on the cells and, hence, their associated temperature. As a result, the cells would no longer be operating in their LILT regime and would not exhibit reduced efficiency.

Table C-3. Types of Concentrating Solar Collectors (and References)

FOCUS GEOMETRY	DIRECT SOLAR ILLUMINATION	INDIRECT SOLAR ILLUMINATION
POINT (High CR)	Fresnel Lens (Piszczor, 1990)	Cassigranian, Parabolic Dish (Rockey, 1980a,b)
LINE (Medium CR)	Compound Parabolic Concentrator (CPC) (JPL, 1979)	Parabolic Trough (JPL, 1983; Rockey, 1980b; Rockey, 1981; Stern, 1988)
PLANE (Low CR)	Enhanced Array (Ralph, 1990; Hughes, 1980a,b; Kosta, 1978; Aerospatiale, 1986)	Inverted Pyramid (Harvey, 1986; French, 1983; Biss, 1983; Rockwell, 1983)

The illumination direction of the collector could be a significant influence on the reliability of the power system. Direct illumination designs with focal planes that are never covered may deliver at least partial power during the mission, even if a deployment failure occurs. Most indirect solar illumination designs are usually completely dependent on successful deployment of the concentrator. These concerns make a direct solar illumination collector a more reliable design and the preferred choice.

The complexity and reliability of different collector designs are related to the focal plane geometry and geometric concentration ratio. With increased geometric concentration comes greater design complexity, structural rigidity, surface alignment accuracy, and Sun-tracking accuracy, to make the Sun's rays hit the small focal plane target. These characteristics apply to an inflatable concentrator, which also introduces operating lifetime and material properties issues like transmissivity and surface reflectivity as a function of time in the space environment. The least complicated, most reliable, and easiest to develop collector would consist of a single flat plane reflecting onto another receiver plane that faces the Sun at all times. This design reduces the number of moving parts, and if deployment fails, the focal plane still receives at least the non-concentrated insolation. This design would have the added advantage of reducing the spacecraft's exposed area (thereby reducing solar-pressure-induced course deviations) by a relatively simple retraction where the reflectors are stowed behind the planar receiver area.

The central receiver, direct-illuminated, low-geometric concentration ratio, planar focus designs have the characteristics most desirable for a first-time, deep space solar concentrating collector design. More specifically, the enhanced array would have a simpler deployment mechanism than the inverted pyramid designs, resulting in the enhanced array being considered further in Appendix E. The split reflector design of the enhanced array allows minimum reduction in performance should any panels fail to deploy. The panels on opposite sides do not overlap and restrict each other, and if the second reflector does not deploy, the first will still partially enhance the array performance. It should be pointed out that the enhanced array design will cause a significant reduction in the field-of-view of the science instruments compared to the baseline and nominal all-solar designs. The basic simplicity and inherent reliability of the enhanced array design makes it the most logical concentrating collector choice for a near-term, deep space mission development and test effort.

A technical feasibility study for space concentrators was completed by Hughes Aircraft Company (Hughes, 1980b). This study surveyed the primary characteristics that a space solar concentrator power system would have to consider to establish technical feasibility. As such, it identified key characteristics relevant to a solar concentrator manufacturer. Its scope and funding was very limited, however, and it did not consider the effects of a large solar collector on a Cassini-type spacecraft.

B. Launch Vehicle Constraints on the Power Source

The mass is influenced significantly by the aperture area of the collector, which is directly related to the efficiency of the energy converter selected. The energy conversion technologies under consideration result in collector sizes in the 10's to 100's of square meters, indicating that some type of complex, three-dimensional deployment will be required to satisfy the dimensional stow volume constraints of the launch vehicle. Concentrating solar collectors have not been demonstrated in space, and the added complexity of a space-qualified, deployable solar concentrator would require some development. All of these unknowns indicate that a solar concentrator may not easily integrate into the selected launch vehicle and still satisfy the launch mass constraints.

C. Design Environments for the Power Source

The launch environment could be very detrimental to a concentrating collector, which must subsequently be deployed and must maintain an accurately contoured surface with a properly located receiver. Launch vibrations increase the probability of deployment failure due to entanglement or jamming and possible damage to the optical surface.

D. Solar Concentrating Collectors Summary

Concentrators in space have been conceptually evaluated but represent a very immature technology, and the increased risk due to uncertain performance justifies eliminating concentrators from further consideration, with the possible exception of the low-technology, low concentration ratio enhanced array concentrators. This leaves non-concentrating photovoltaic arrays and enhanced array concentrators as the only solar power options meriting further investigation.

C.4.3 Reactors (SP-100, SNAP 10A, Topaz, Star-C, Romashka)

A nuclear reactor depends on a sustained chain reaction of the fission of atoms and the resulting energy release to generate heat. Reactors are often categorized as fast, intermediate, or thermal, depending on the neutron energy spectrum in the reactor (Shirbacheh, 1987). Typically, a reactor consists of enriched uranium-235 oxide, carbide, or nitride fuel. The fuel is contained in fuel pins, and a coolant flows around the pins to transfer the fission heat from the reactor to the conversion system. Two other means of cooling the reactor and transporting heat to the conversion system are: (1) heat pipes, with their phase-change, heat-transport fluid, and (2) conducting heat directly through the vessel wall to the converter. Another type of space reactor is an in-core thermionic reactor, which integrates a thermionic converter into the reactor such that the fuel pin is the hot side of the converter and the coolant flow is the cold side.

C.4.3.1 Technical Readiness. The power level requirement for a Cassini reactor system should not be a problem. Reactors currently being designed and developed which satisfy the criticality limit of a space nuclear reactor tend to exceed the input power required of any reasonable conversion system that fulfills the Cassini output power requirements.

The launch date and lifetime requirements are a significant risk, because the reactor's technical maturity is low and a long lead time is needed to design and conduct validation tests of a new space reactor design. There are a number of technical risks associated with space reactors, even though several U.S. reactors have been developed and tested; one (SNAP 10A) was launched into space on April 3, 1965. The launched reactor power system was a 500-W thermoelectric power arrangement that operated successfully for 43 days before a spurious command signal (resulting from a high-voltage failure of the spacecraft decoder) caused the reflectors to release from the reactor, resulting in a shutdown (six, 1984). The Soviet Union has flown nuclear reactor power sources since about 1965, primarily with thermoelectric converters, on approximately 33 spacecraft, most of them on radar ocean reconnaissance satellites (RORSAT) for tracking U.S. Navy ships (Primak, 1989). Two of these reactors gained worldwide recognition when they re-entered the Earth's atmosphere; a third, Cosmos 1900, narrowly escaped reentry. Cosmos 954 re-entered over Canada in January 1978, and Cosmos 1402 re-entered in 1983 (Bailey, 1988; Kiernan, 1990). Two high-power reactors using thermionic converters, called Topaz, were launched in 1987 by the Soviet Union on Cosmos 1818 and 1867. Space reactors (either U.S. or Soviet designs) have demonstrated short lifetimes in space (43 days and one year, respectively) and questionable reliability (Bennett, 1989; Aftergood, 1991).

Control of a space reactor is another complexity. In general, a space reactor system, even with the experience expected to be gained from the Space Power 100-kW_e (SP-100) reactor ground tests (originally intended for 1995; now delayed), will be entirely too late and will not have sufficient technical maturity for a long-term mission scheduled for a 1997 launch date (Kiernan, 1990).

C.4.3.2 Launch Vehicle Constraints on the Energy Source. The mass of a reactor energy source is a major concern. The SNAP-10A reactor had a mass of 274 kg (604 lb) and required an additional 217 kg (478 lb) mass shield (U.S. DOE, 1983). More recent preliminary reactor designs have mass estimates of 200 to 300 kg (440 to 660 lb), not including shielding.

C.4.3.3 Design Environments for the Energy Source. Solar range and insolation are not concerns for a reactor, since output power is independent of solar insolation.

Micrometeoroids are a concern, because the primary cooling loop used in most reactors represents a single-point failure for the mission and must be well protected. This tends to increase the mass and/or reduce the lifetime and reliability.

C.4.3.4 Reactor Summary. The combined disadvantages of high mass and low technical maturity eliminate reactor energy sources from further consideration for Cassini.

C.4.4 Fuels and Chemicals

Fuels and chemicals (hydrogen and oxygen, primary batteries, etc.) are an 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 for use in a number of energy conversion systems. These types of systems are common on spacecraft. For example, fuel cells are used on the Space Shuttle, and batteries and propellants are found on most spacecraft. On spacecraft, propellants are used primarily for propulsion and not for electric power generation.

C.4.4.1 Technical Readiness. Existing power systems in space using various fuels and chemicals have the capacity to generate the maximum electrical power required. The technology exists to satisfy both launch date and, possibly, lifetime requirements. This simply means that the technology exists and other factors must be addressed to determine the design viability.

C.4.4.2 Launch Vehicle Constraints on the Energy Source. The most difficult design requirement for stored fuels and chemicals is combining power and life-time, which results in extremely massive power systems. A reliable means of putting a large, long-lived, complex spacecraft into a dormant (low power) mode does not exist at this time. It is necessary to use the existing power demand profile as a best estimate for total energy consumption. Combining the mission power profile and the operating lifetime of Cassini results in over 60 MW_e-h. Assuming a hydrogen and oxygen system (about the highest currently available energy fuel-per-unit mass) and 100% efficient conversion (an idealistic assumption), the resulting mass for the fuel and oxidizer, without tanks, transport, or storage equipment, is 16,000 kg (35,300 lb); this is 4 to 6 times the total allocated dry launch mass of the spacecraft. When more realistic conversion efficiencies are assumed, the mass increases by factors of 4 to 16. One of the highest energy-density primary batteries (lithium thionyl chloride) has an energy density that is 6.2 times lower than hydrogen and oxygen (Halpert, 1986); the former increases the power system mass to 100,000 kg (220,500 lb) for Cassini.

C.4.4.3 Design Environments for the Energy Source. Fuels and chemicals usually require some type of thermal control, which is affected by solar insolation. However, this should not be a driving consideration for this technology.

C.4.4.4 Fuels and Chemicals Summary. Mass makes any form of stored fuels and chemicals unrealistic for Cassini due to the limits of the available launch vehicle. Consequently, this energy source has been eliminated from further consideration.

C.4.5 Power Beaming from Earth (Microwave and Laser)

Power beaming from Earth consists of generating and transmitting microwaves or laser beams from Earth to a spacecraft, receiving or collecting the beam energy on board, then converting the energy to electricity. Key parameters for designing a power beaming system include technology and hardware readiness, transmission distance, user power required, and atmospheric attenuation and spreading. These design parameters lead to the selection of a particular frequency at which power is transmitted and the thermal management system which cools the transmitter and receiver. Once these parameters are set, the size of the transmitter and receiver antennas are determined based on the product of the areas of the transmitting ground antennas and the spacecraft receiving antenna (U.S. DOE, 1 990c).

C.4.5.1 Technical Readiness. The power requirement and the distance from Earth to the spacecraft are primary drivers for this type of system. Large transmission distances (greater than 5000 km or 3125 mi) dictate the need for very short wavelengths (laser frequencies) rather than long wavelength microwave transmissions for reasonably sized transmitters and receivers (Cull, 1989). The power transmission efficiency is influenced by beam spreading, jitter, and pointing accuracy. Another concern for beaming power is the need to have multiple transmitters on the Earth to continuously supply power to the spacecraft at all orientations and locations during the mission. The only exceptions are during occultations of the beam, when batteries could be used.

A laser power transmission system design was developed using current technology that requires five years to flight-qualify (Coomes, 1989). This 4% efficient system could only deliver 20 W_e at a maximum range of 6000 km. These values are orders of magnitude below the range and power requirements of Cassini.

It is not feasible 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 coherent, high-power laser light to such a small target (out to between 9.0 and 9.3 AU) represents a long-term space technology development effort. Any technology that is this immature cannot have sufficient available data to ensure the satisfaction of design lifetime requirements.

C.4.5.2 Launch Vehicle Constraints On/Design Environments for the Energy Source. The technology is not well enough understood to address these issues at this time.

C.4.5.3 Power Beaming Summary. The technical immaturity of any type of beamed power system eliminates it from further consideration for Cassini.

C.4.6 Energy Sources Summary

An evaluation of the energy sources in the matrix has resulted in the identification of the GPHS (radioisotopes), the non-concentrating solar collector, and the enhanced array concentrator as worthy of further consideration. This energy source identification eliminates a number of potential energy converters that are not compatible with radioisotopes, non-concentrating, or low-concentration solar energy.

C.5 POWER SYSTEM TECHNOLOGIES EVALUATION

Only those technologies compatible with radioisotopes, low-concentration solar, and solar non-concentrating power sources will be evaluated against the feasibility criteria. They are the only viable energy sources for Cassini, and the only ones not eliminated by the comparison study previously described. Table C-4 identifies the applicable conversion technologies; an “X” denotes compatible source/conversion technology pairs.

C.5.1 Static Conversion Technologies

C.5.1.1 Photovoltaics with Solar Non-Concentrating or Low Concentration. In a photovoltaic cell, photons are converted directly to electricity at the junction of N and P semiconductor materials. Photovoltaic arrays have been used on most spacecraft between the orbits of Mercury and Mars (Stella, 1990). Current technology cell materials include Silicon and Gallium Arsenide/ Germanium; advanced, high-efficiency, thin, radiation-resistant materials include Indium Phosphide (InP) and Copper Indium Diselenide (CIS). Current solar array designs are categorized as rigid, roll-out or foldout flexible substrate. The primary difference and uncertainty with photovoltaic arrays with a low concentration ratio (enhanced array) is with the collector that was discussed earlier, and not with the array.

Table C-4. Applicable Conversion Technologies Evaluated

Conversion Technology	ENERGY SOURCE		
	Radioisotopes	Non-Concentrating Solar	Low Concentration Solar
STATIC:			
Photovoltaic		X	X
Thermoelectric	X		
Thermionic	X		
Alkali Metal Thermoelectric Converter (AMTEC)	X		
Thermophotovoltaic	X		
DYNAMIC:			
Rankine	X		
Brayton	X		
Stirling	X		

A. Technology Readiness

A photovoltaic array, if large enough, could conceivably provide sufficient power for a Cassini-type mission if all other constraints were satisfied. Single-crystal Si cells (initially used over 30 years ago in space applications) or GaAs/Ge cells, which the U.S. and Japan manufacture, are currently available. The Japanese have launched at least one geosynchronous satellite powered entirely with GaAs/GaAs cells (Stella, 1990). GaAs/Ge has replaced GaAs/GaAs in the U.S. due to lower cost and reduced fragility. The U.S. recently launched MSTI, which was powered by a small GaAs/Ge panel. In comparison to Si cells, GaAs/Ge cells do not exhibit a significant degradation in efficiency at low insolation intensities and low temperatures; they also have higher radiation resistance and are higher in cost and mass. Si cells are a mature technology with efficiencies of at least 13.8% at standard conditions (AM0 28° C). GaAs/Ge, although more efficient (18% AM0 28° C) than silicon, have less flight experience. Other cell materials are too immature technologically to consider further.

The operating and design lifetime of silicon cells is understood well enough that reasonable predictions of degradation can be made and performance compensated for in sizing the array. The lifetime and degradation of GaAs/Ge is less certain because there is a limited operating data base. A deployable photovoltaic system, regardless of cell type used, is more complex than a fixed array, and deployment mechanisms represent a single-point failure that reduces reliability and lifetime in most designs.

The most promising near-term array configuration is the Advanced Photovoltaic Solar Array (APSA) recently developed for NASA/JPL by TRW (TRW, 1990; TRW, 1986) that uses 62 μm-thick silicon cells, deployable, foldout, flexible-array technology. APSA has been modified to use GaAs/Ge cells instead of silicon cells. After the silicon APSA prototype was completed in 1990, ground testing was conducted using a variety of cells (Flood, 1989; TRW, 1991). No funding has been allocated for flight-testing APSA at this time; however, other foldout arrays have flown (e.g., Olympus and CTS). A modified version of APSA using GaAs/Ge cells is presently baselined for EOS-AM, which has an approximate power level of 6 kW.

B. Launch Vehicle Constraints on the Power System

Preliminary mass estimates indicate that the large power and solar ranges required for Cassini drive the mass of a photovoltaic power system to a very high value.

C. Design Environments for the Power System

Cassini will experience solar insolations from the orbit of Venus to that of Saturn. In the past, NASA has exposed solar arrays to insolation values between the orbits of Mercury and Mars.

D. Photovoltaic Non-Concentrating and Low-Concentration Solar Power System Summary

The APSA with either Si or GaAs/Ge cells represents a potential power system alternative for Cassini, which will be evaluated further in Appendix D. A very conceptual enhanced array design is described in Appendix E.

C.5.1.2 Thermoelectrics with GPHS. In 1822-1823, Thomas Seebeck discovered that when two electric conductors of different material are joined in a closed circuit and the two junctions are kept at different temperatures, an electric voltage is produced (JPL, 1987). This thermoelectric effect, when combined with a high-temperature heat source like a radioisotope and a low-temperature heat sink, is used to produce electricity for various applications. The Navy has used this type of power system in the ocean, and numerous space missions have included this type of power system as well.

C.5.1.2.1 GPHS-RTG

A. Technology Readiness

Radioisotope heat sources (e.g., GPHS) coupled with thermoelectric converters (e.g., SiGe unicouples) have already demonstrated the Cassini beginning-of-life power and launch date requirements. GPHS-RTGs have been used on the Galileo and Ulysses spacecraft: power, launch date, and lifetime design requirements can be satisfied with existing hardware designs with no additional technology development necessary (Bennett, 1983 and Bennett, 1986). Although the end-of-life power and design lifetime have not been demonstrated in space with the current GPHS-RTG, the Multi-Hundred-Watt RTG on the Voyager 1 and 2 spacecraft has demonstrated the end-of-life power and design lifetime (greater than 15 years) using identical couples and materials.

B. Launch Vehicle Constraints on the Power System

The mass of the GPHS-RTG has been measured; a power system using three GPHS-RTGs is within the mass constraints of the launch vehicle.

C. Design Environments

The GPHS-RTG has demonstrated operational success at solar ranges from 0.7 AU at Venus with Galileo to 5.4 AU during closest approach to Jupiter with Ulysses. Other types of RTGs on Voyager and Pioneer have now exceeded 50 AU.

D. GPHS-RTG Summary

In summary, the successful experience with the GPHS-RTG in space applications makes it a candidate for deep space, long-duration missions such as Cassini.

C.5.1.2.2 Modular Radioisotope Thermoelectric Generator (MOD-RTG)

A MOD-RTG has never been built, but the current design consists of a number of separate power modules which may be stacked and supported in such a way that it is possible to obtain a wide variety of RTG electric-output power levels using the same hardware. Each power module is projected to provide about 16.8 W of electricity at 28 V, at the beginning of the mission (BOM), and consists of a single GPHS providing energy to 8 multicouples. Each multicouple contains 20 SiGe/GaP unicouples in a glass-bonded thermopile that provides 2.1 W of electricity at 3.5 V (BOM).

A. Technology Readiness

The modular nature of the MOD-RTG would make it possible to achieve the BOM power requirements for Cassini. Satisfying the EOM power requirements would be much more uncertain, since only a few multicouples have been performance and life tested (to only 15,000 hours at standard operating conditions). A few multicouples were "accelerated" life-tested for 4000 hours at only 25 and 50° C above normal operating conditions. These same multicouples were operated for 1000 hours at standard operating conditions. Test results indicate improved efficiency would be expected from the multicouple compared to the standard GPHS-RTG unicouples. However, the degradation rate for the multicouples appears to be higher than that for the unicouples, which would result in a lower EOM power output. The extremely limited amount of lifetime and performance data available at this time and the lead time required to go from laboratory test data to qualified flight hardware makes the MOD-RTG a technologically non-viable option for a Cassini 1997 launch date (Hartman 1992, 1993a, 1993b).

B. Launch Vehicle Constraints on the Power System

Assuming the performance and lifetime predictions are correct, the mass of the MOD-RTG should be lower than the mass of the GPHS-RTG, which would fall within the mass constraints of the launch vehicle.

C. Design Environments

There are no anticipated design environments that would be problematic for the MOD-RTG, but this must be verified with tests.

D. MOD-RTG Summary

The MOD-RTG must be eliminated from consideration for Cassini since the immature technology makes the performance and lifetime very uncertain for a long-duration mission. Immediate resolution of these problems would still leave the difficult task of final flight hardware design and fabrication to be completed in time for a 1997 launch.

C.5. 1.3 Thermionics with GPHS. A thermionic converter uses heat applied to a high-temperature cathode to drive electrons across a gap to the low-temperature heat sink anode, which then drives an external electrical load. The gap between the cathode and the anode is typically evacuated with a very small separation distance and has a low-pressure, ionized cesium vapor added. Thermionic converters tend to be high-current, low-voltage devices with efficiencies in the 5 to 20% range, depending on operating temperatures.

A. Technical Readiness

A thermionic converter is a very high-temperature device (greater than 1600 K). The maximum surface operating temperature of the GPHS is 1373 K. The low performance of a thermionic converter below 1600 K and the increased lifetime risk to the GPHS above 1373 K makes this converter and the GPHS energy source incompatible.

B. Thermionics with GPHS Summary

The basic incompatibility of the thermionics with GPHS eliminates this power system. There is insufficient time to develop either a higher temperature GPHS that could be compatible with a thermionic conversion subsystem, or a higher efficiency, long-lived, thermionic converter for use with the existing GPHS.

C.5.1.4 AMTEC with GPHS. An alkali metal thermoelectric converter (AMTEC) is an energy conversion device using the unique property of beta-alumina, which allows charged sodium ions to pass through it, but not neutral sodium atoms. High-temperature and high-pressure liquid sodium on one side of the beta- alumina allows electrons to flow from the liquid through an electrical load and then returns the electrons to the sodium ions on the opposite side of the beta-alumina plate. The low-pressure side is the condensing side of the loop where the sodium condensate is re-circulated and heated to continue the cycle.

A. Technology Readiness

This technology is in the laboratory development phase and requires performance, degradation, spacecraft integration, launch, lifetime, and O g issues to be resolved before it can be considered for a spacecraft application (JPL, 1990a). Projections indicate that this power system will not be ready for launch on a high-reliability, long-lived mission until the late 1990's to early 2000's (Sievers, 1992).

B. AMTEC with GPHS Summary

These development issues will take much longer than the five years currently available to have flight-qualified hardware ready for Cassini launch, thereby eliminating it from further consideration.

C.5.1.5 Thermophotovoltaics with GPHS. A thermophotovoltaic (TPV) converter is similar to the solar photovoltaic converters described earlier. The primary difference is that instead of using photon energy from the Sun, the TPV uses another heat source for the photon supply, and has photovoltaic converters designed to maximize the efficiency of the system.

A. Technology Readiness

Unfortunately, based on work to date, a TPV is a very high temperature conversion system (greater than 1500 K) and is incompatible with the GPHS allowable temperatures for a conversion system having reasonable performance (Home, 1982; NASA, 1981), or requires a special band pass filter to be developed. Obtaining reasonable performance is further complicated by the fact that the TPV converters need to be relatively cool (~300 K), while the emitter or GPHS heat source needs to be high in temperature. The TPV converters can only use a limited fraction of the emitted spectrum of energy to generate electricity. An effective means of minimizing the unused energy emitted from a lower temperature GPHS heat source (e.g., band pass filter) must be developed before a TPV power system could be made that would be higher in efficiency and use less Pu-238 than the current GPHS-RTG.

A theoretical design model has been developed that predicts efficiencies as high as 14.4%, but these numbers have not been demonstrated or validated by testing (Day, 1990).

B. Thermophotovoltaics with GPHS Summary

The basic temperature incompatibility of the existing GPHS, combined with the immature TPV technology relative to performance and lifetime, eliminates this power system; there is insufficient time to develop a higher temperature GPHS that could be compatible with a TPV conversion subsystem, or a high-efficiency TPV conversion system compatible with a 1373 K temperature GPHS.

C. Static Conversion Technologies Summary

An evaluation of the static conversion technologies in the modified matrix (Table C-4) has resulted in elimination of all static conversion technologies except Si or GaAs/Ge photovoltaics with APSA or the enhanced array concentrator, and the GPHS-RTG.

C.5.2 Dynamic Conversion Technologies

Certain characteristics common to all dynamic energy conversion systems have major impacts on planetary missions. All rotating or reciprocating machines introduce dynamic forces into the spacecraft regardless of the effort to eliminate this problem. These vibrating and gyroscopic motions also impact attitude control, increasing the spacecraft mass and reducing the lifetime of other components. Dynamic system specific power (kW_e/kg) and efficiency, particularly for rotating machines, increases with the size or power level of the equipment. This characteristic is not desirable for a low-power, long-lived system. To minimize the system mass, a single, large, high-efficiency converter requiring a low-mass energy source is needed. This conflicts with reliability needs, which require many small converters. The lifetime and reliability issues are a major concern because a single micro-meteoroid penetration of a machine causes its working fluid to leak and, if lost, discontinue operation. Finally, no dynamic power conversion system has been built and tested for use in space.

C.5.2.1 Rankine with GPHS. The basic Rankine Cycle engine consists of a pump to increase the pressure of the liquid, a boiler to add heat to the pressurized liquid, changing it to a vapor, a turbine to expand the vapor through and extract the work from the fluid, and finally, a condenser to reject the waste heat and return the vapor to a liquid. The Rankine Cycle engine is unique from the other dynamic systems because the working fluid experiences phase changes from liquid to vapor and vapor to liquid throughout the cycle. The selection of the working fluid is the most crucial design issue because it affects operating lifetimes, temperatures, pressures, and component/cycle performance.

A. Technology Readiness

There is little doubt that a terrestrial engine can be built to generate power at the level required for Cassini. However, to build a Rankine Cycle to operate in 0 g is a major development risk, particularly for the condenser. Some form of artificial gravity from a jet condenser or a rotating radiator is required to separate the liquid phase from the vapor phase. The working fluid is not only vulnerable to leakage from the engine but contamination and degradation within the engine. These technologies have been analyzed and tested on the ground but have no long-term, 0 g experience in space (Bland, 1983).

B. Rankine with GPHS Summary

The lack of operating experience in space and the five-year time frame until the Cassini launch date eliminates the Rankine Cycle from further consideration.

C.5.2.2 Brayton with GPHS. The Brayton Cycle is similar to the Rankine Cycle with the exception that the pump is replaced with a compressor, because the working fluid remains a gas at all times.

A. Technology Readiness

The Brayton Cycle, like the Rankine Cycle, has extensive terrestrial experience, much of which is relevant to a space power conversion system. One specific Dynamic Isotope Power System (DIPS) that combines a Brayton Cycle with a radioisotope heat source has been studied extensively (U.S. DOE, 1990a; U.S. DOE, 1 989b). The early designs of these systems appear reasonable but have not been fabricated, ground-tested, or space-qualified.

B. Brayton with GPHS Summary

The amount of time available to fabricate, ground test, and space-qualify a Brayton engine is insufficient to satisfy the Cassini launch date requirement.

C.5.2.3 Stirling with GPHS. The Stirling Cycle uses a gas working fluid, typically hydrogen or helium. Ideally, the working fluid is isothermally compressed while heat is rejected to an external sink. Heat is then added to the working fluid from the regenerator during a constant volume process. Isothermal expansion of the working fluid takes place while work is extracted and heat is added from an external source. Finally, heat is resumed to the regenerator during a constant volume process. If all processes are conducted reversibly, then the efficiency of the Stirling Cycle approaches that of the Carnot Cycle, the most efficient cycle theoretically possible.

A. Technology Readiness

In practice there are two types of Stirling engines: the kinematic type, which relies on cranks and rods to move the power and displacer pistons; and the free-piston type where the gas springs and mass of the pistons are used to sequence the power and displacer pistons. The kinematic engine is not a long-lifetime system. Consequently, only the free-piston designs can be considered for space applications.

There are two types of free-piston Stirling engines in development. The first is a 25-kW_e design. For a development engine, it has demonstrated good performance. However, it suffers from the same problems as its smaller sized predecessors—a limited unattended operating lifetime (Bear, 1983).

These engines must therefore be eliminated from additional consideration; there is insufficient time to develop these engines for Cassini, and there could be a potential lifetime problem. This leaves only the diaphragm-type, small Stirling engine for further consideration.

Small Stirling engines for artificial hearts (White, 1982; White, 1983) and the Harwell diaphragm engines (Cooke-Yarborough, 1980; Cooke-Yarborough, 1981) have demonstrated lifetimes up to 10 and 7 years at power levels of 6 and 60 W, respectively, for terrestrial applications. The 60-W Harwell engine used in a lighthouse application was extremely massive, and the design is not conducive to space applications. The artificial-heart Stirling is a low-mass, compact design that conceivably could be modified to drive a linear alternator instead of a pump. Neither the engine nor the linear alternator has been designed or tested for a space application, hence, it is unlikely that either could be space-qualified for launch on Cassini.

B. Stirling with GPHS Summary

The overall conclusion is that the Stirling engine is not currently a technologically viable alternative for Cassini and, consequently, it is eliminated from further consideration.

C.5.2.4 Summary of Dynamic Conversion Technologies with GPHS. All of the dynamic conversion technologies in the modified matrix (Table C-4) have been eliminated as options worthy of further consideration due to their technological immaturity.

The conclusions are that the conventional GPHS-RTG, APSA (i.e., non-concentrating solar with photovoltaics), and low-concentration arrays using Si or GaAs/Ge cells are the only technologies that can potentially satisfy Cassini power system requirements. The GPHS-RTG powered spacecraft design is discussed in Volume 1 of these supporting studies. The solar photovoltaic spacecraft are evaluated in Section 4 of this volume and in Appendixes D, E, and F.

APPENDIX D

ALL-SOLAR POWERED CASSINI SPACECRAFT

This appendix describes the original all-solar design, which is based on the original baseline Cassini spacecraft with scan platforms. This appendix describes the detailed design study conducted to determine the impact on the spacecraft and mission design if photovoltaic arrays are used for electrical power generation instead of plutonium-fueled RTGs. The ground rules for this study were to minimize (1) impact to the spacecraft design and (2) the expected science impacts resulting from any spacecraft changes.

This investigation focuses on the impacts to the Cassini Orbiter due to Saturn operations. It does not examine impacts to the Probe design, the effects of near-Sun operations (i.e., Venus flybys), the spacecraft effects associated with a more complicated mission design, or other factors that will increase the mass, cost, and complexity of the all-solar Cassini design. Information for the baseline RTG design was based upon,the latest mass and power estimates available when the study was done (JPL, 1991b).

The study is broken into two major sections. The first section examines an all-solar design using silicon solar cells, and the second examines a design using GaAs cells. Each section details the assumptions and requirements used for that option, the spacecraft design description, impacts to spacecraft design and performance, and cost impacts. At the end of the appendix is a brief comparison of the silicon and GaAs designs.

D.1 PHOTOVOLTAIC TECHNOLOGIES

Two solar conversion technologies were determined to be technically mature enough to be considered for Cassini, silicon and gallium arsenide solar cells.

Silicon solar cells have been used for space power since the late 1950s. Along with advances in silicon semiconductor device technology (particularly materials and fabrication methods), solar cell improvements continued into the 1980s. In the 1960s it was realized that other materials might provide higher solar cell performance than Si, and initial development was started on GaAs devices. It was not until the late 1970s, however, that the efficiencies for GaAs-based cells began to equal and then exceed those of Si. Similar to silicon cell development, improvements in GaAs devices were closely related to advances in GaAs optoelectronic devices.

Both Si and GaAs solar cells are based on the use of single-crystalline materials. The GaAs crystal is composed of two elements, gallium and arsenic. Both cell types are formed by doping with minute quantities of impure elements to form the positive/negative junction that is needed. In practice, modern space-qualified GaAs cells have an additional layer of AlGaAs (an AlGaAs/GaAs device), but this would not be noticeable to the cell user; all of the cell types mentioned here will look very similar. In production quantities, the thin AlGaAs/GaAs cell offers higher efficiency than Si (18% vs. 13.8%), better efficiency at elevated temperatures, and improved radiation resistance. On the other hand, GaAs-based cells are more brittle than Si, slightly more than twice as dense, and have twice the mass of typical Si circuits. In addition, silicon efficiency improves at lower temperatures, narrowing and even reversing their differences. Recent efforts have been directed at making the AlGaAs/GaAs/Ge solar cells more robust and less dense.

For scientific and commercial satellites, Si has been the historically preferred technology based on experience and cost. GaAs cells are several times more expensive than silicon, and many applications do not require the added performance afforded by GaAs cells. However, as GaAs cell costs are being reduced, GaAs is becoming more popular within the commercial industry, and more and more commercial and scientific satellites are proposing GaAs for power conversion.

D.1.1 APSA Solar Array

After thorough consideration of current solar array technology, the Advanced Photovoltaic Solar Array (APSA) was determined to be the best solar array for an all-solar Cassini. The APSA is a new type of lightweight deployable solar array that may be suitable for long-duration interplanetary missions. The design is an improved, flexible blanket array design that uses thin solar cells (Si or GaAs) to minimize mass. Based on existing technology, a prototype APSA array has been completed at TRW, Redondo Beach, California. Prototype wings have been tested to verify the performance of various mechanisms and to demonstrate its mode of deployment. Environmental tests (vibration, acoustics, and temperature cycling) and strength/stiffness tests have also been completed.

However, an APSA array of the size required for Cassini has never before been fabricated, or tested under conditions comparable to those associated with the Cassini mission. Major modifications to the prototype design would be anticipated due to the significant size and area differences. For use on the Cassini mission, development would have had to have started no later than the end of 1992 in order to support the present launch schedules. A modified version of APSA using GaAs/Ge cells, which has a power level of -6 kW, is presently baselined for EOS-AM.

D.2 ALL-SOLAR CASSINI USING SILICON CELLS

The first part of the study utilizes silicon cell technology for power conversion. The silicon powered Cassini discussion is broken into six major subsections: the assumptions used during the study, the predicted performance of an APSA array at Saturn, a description of the design changes required to modify the spacecraft for solar power, a description of the associated impacts, discussion of solar array articulation options, and predicted differences in cost, or cost deltas, for the all-solar design.

D.2.1 Assumptions

D.2.1.1 Launch Vehicle Fairing. The baseline launch vehicle is the Titan IV (SRMU)/Centaur. The spacecraft dynamic envelope inside the current payload fairing is 180 inches (4.6 m) in diameter and 240 inches (6.1 m) in height. This limits the size of the structure that can be stowed. The PLF height is currently 66 ft (20.1 m) and can be increased by 10 ft (3.1 m) sections, if required. The maximum size fairing available is 86 feet (26.2 m) in length. Stowed solar arrays are assumed to fit within the given dynamic envelope, as are any needed changes to spacecraft configuration.

D.2.1.2 Radiation Environment. The radiation environment encountered by Cassini during its mission is assumed to be similar to that encountered in a similar duration geosynchronous orbit about the Earth. The degradation due to radiation is possibly overestimated by assuming geosynchronous orbit fluences. The difference, however, is small (around five percent), and is not considered.

Project requirements (JPL, 1991c) state that for all electronic devices a radiation design margin (RDM) of two must be applied. For the solar array, this is accomplished by doubling the expected radiation fluence seen by the arrays in geosynchronous orbit (see Subsection D.2.2.2).

D.2.1.3 Sunpoint Accuracy. The degree to which the solar arrays are pointed to the Sun affects their power output. Maximum power is available when the plane of the solar arrays is normal with the spacecraft's line-of-sight with the Sun (considered 0° off-Sun). Therefore, the degree to which the solar arrays are Sun-pointed will depend on the angle of the Sun and Earth as seen by the spacecraft. It has been assumed that the spacecraft will remain Sun-pointed for most of the orbital tour.

A discussion of articulated vs. non-articulated solar arrays (Subsection D.2.5) addresses the issue of off-Sun pointing.

D.2.1.4 Conductive Coating. Fields and particles instruments science return may be affected if solar panel surfaces are not kept at an equipotential, so as not to deflect particles. This may require that a conductive coating be applied to the cells to maintain the necessary equipotential. Such coatings have been developed and flown on rigid solar arrays, but none have ever been developed for a flexible foldout array, such as APSA. A conductive coating will have a small impact upon the conversion efficiency of the array, but it is sufficiently small in magnitude that it was ignored in this study.

D.2.1.5 Field-of-View. The solar arrays are expected to cause a field-of-view (FOV) problem for thermal control and the science instruments. It is assumed that the thermal control design will allow for the change in configuration and that this will not have a dramatic effect on the mass and power performance of the solar arrays. Scan-platform-mounted instruments (assumed in this appendix) will lose part of their range of motion due to the presence of the arrays. Body-fixed instruments, such as the magnetometer, will need to be located so as to minimize the impact of restricted FOVs. It is assumed that the mission design will accommodate potential FOV constraints for the science instruments.

D.2.1.6 Ring Particle Damage. The ring planes of Saturn are known from previous missions to have orbiting particles. There is a significant probability that some particles may impact the spacecraft when Cassini crosses the ring planes. The RTG-baseline vehicle allows for the spacecraft to withstand this type of impact. It is assumed that proper alignment of the arrays during ring plane crossings—minimal cross-sectional area facing the particles—can minimize solar cell damage. No allowance has been made for any array damage due to Saturn ring plane crossing.

D.2.2 Silicon APSA Performance Characteristics

This section details the performance characteristics of the silicon APSA array and the mission-specific factors which impact APSA performance.

D.2.2.1 Nominal Performance. The nominal (Earth orbit) performance parameters of a silicon APSA array were published in a TRW report (TRW, 1990). Based on a prototype detailed design with silicon cells, the end-of-life (EOL) performance for a 12.6-year geosynchronous mission is:

Specific power (12.6 y, 1 AU) = 90 W/kg

Power density (12.6 y, 1 AU) = 93 W/m^2

(i.e., where specific power is the ratio of power generated per unit mass, and power density is the ratio of power to area.)

D.2.2.2 Radiation Degradation. As previously mentioned, an RDM of 2 is required for electronic parts. The RDM for the solar cells is applied by doubling the expected radiation fluence seen by the arrays when determining its degradation effects. In this case, radiation fluence is linear over time. Therefore, to account for an RDM of 2, the power factors are scaled proportionately (Tada, 1982; Anspaugh, 1989). Accounting for the necessary RDM, the performance factors for Cassini become:

Specific power (12.6 y, 1 AU, RDM 2)
= Specific power (25.2 y, 1 AU)
= 82.3 W/kg (at Earth)

Power density (12.6 y, 1 AU, RDM 2)
= Power density (25.2 y,1 AU)
= 85 W/m^2 (at Earth)

D.2.2.3 Insolation. After insertion into Saturn orbit, the Cassini spacecraft's distance from the Sun varies from 9.0 to 9.3 AU during the nominal mission. This variation has very little effect on the amount of solar radiation received by the arrays; for the purposes of this study, the variation is negligible. The insolation for the Cassini spacecraft at Saturn remains essentially constant at about $1/(9.0)^2$, or 1.2 percent (0.012) of that at Earth during the mission.

D.2.2.4 Low Intensity, Low Temperature. At large distances from the Sun, silicon cell performance is different from that obtained in Earth orbit. Two factors are responsible for this change in performance — low intensity solar radiation and low operating temperature.

As the distance from the spacecraft to the Sun increases, the intensity of the solar radiation impinging upon the solar arrays is reduced, and the operating temperature of the cells is also reduced. Solar cells operating under these conditions provide different current and voltage levels than those provided in Earth orbit. In ideal cells, the net effect of these changes would produce an increase in the operating efficiency of the cells. However, tests have shown that most silicon cells exhibit anomalous behavior under low intensity/low temperature conditions. Due to mechanisms which are not fully understood, some cells show drastically reduced efficiency when operating under these conditions. This anomalous behavior, known as LILT degradation, is random, and it cannot be predicted which cells will exhibit this behavior.

Significant testing has been performed to characterize LILT effects for distances up to 5 AU, but almost no testing has been done to characterize LILT effects for larger distances. The LILT effects for cells at Saturn have been extrapolated from existing data. Table D-1 shows the estimated performance of silicon cells under the conditions which would be encountered at Saturn.

D.2.2.5 Additional Acceleration Loading. The APSA array structure is designed to withstand acceleration loading up to 0.01 g. The Cassini mission must handle 0.05 g during 490-N engine bums. The acceleration loading for the worst-case spacecraft mass is calculated as follows:

$$\begin{aligned} \text{g loading} &= [\text{Main engine thrust}] \times [\text{Dynamic load factor}] \\ &\quad \div [\text{RTG baseline mass - RTG/RHU component mass}] \\ &\quad \div [\text{m/s}^2 \text{ to g conversion factor}] \\ &= 490 \text{ N} \times 2/(1920-221) + 9.81 \text{ m/s}^2/\text{g} \\ &= 0.05 \text{ g} \end{aligned}$$

To handle the additional g loading, the solar array must be reinforced, leading to a decrease in the specific power. For Cassini, this specific power derating factor is estimated to be 0.88.

Table D-1. Estimated Change in Silicon APSA Power Output Due to LILT: Cassini EOM at Saturn*

Low intensity at low temperature (LILT) power degradation	0.6
Low temperature conversion efficiency (voltage enhancement)	1.2
LILT factor (power degradation x conversion efficiency)	0.72

* There is no test data for distances of 9.0 AU and greater. These values were extrapolated from curves for 1 t0 5 AU (TRW, 1986).

D.2.2.6 Estimated Silicon APSA Performance. Utilizing these factors, the specific power and power density for Saturn operating conditions are determined as follows:

Specific power = [Specific power (12.6 y, 1 AU, RDM 2)]
x [Insolation factor]
x [LILT factor] x [Structural beef-up factor]
= 82.3 W/kg x 0.012 x 0.72 x 0.88
= 0.62 W/kg

Power density = [Power density (12.6 y, 1 AU, RDM 2)]
x [Insolation factor] x [LILT factor]
= 85W/m^2 x 0.012 x 0.72
= 0.73 W/m^2

D.2.3 Spacecraft Design Modifications

This section addresses the first order design modifications required to implement an all solar design using silicon solar cells.

D.2.3.1 Silicon APSA Size. The size of the solar array is governed by the amount of power required to perform the mission under the worst anticipated conditions.

Since one of the ground rules is to minimize mission impact, the nominal power required to operate the spacecraft and instruments is assumed to be equal to the power provided at the end of the mission by the three RTGs. Additional power is required for the electric heaters used to replace the RHUs which are eliminated when all radioactive isotopes are removed from the design. 117 RHUs were included in the original Orbiter and Probe designs,[1] and must be replaced with 117 W of electric heaters. The panels will also be required to provide 45 W of power to recharge the

[1] The current best estimate is between 130 and 230 RHUs. This study used the original figure of 117.

secondary batteries after occultations (discussed further in Subsection D.2.3.2). The total amount of power to be provided by the arrays, not including RTG waste heat supplied to the propellant tanks (Subsection 4.2.2.1), is therefore:

675 W + 117 W + 45 W = 837 W

Using the predicted APSA performance at Saturn, the total resulting mass and area estimates are:

Mass: (837 W) / (0.62 W/kg) = 1350 kg

Area: (837 W) / (0.73 W/m^2) = 1147 m^2

Adding 5 m^2 (53.7 ft^2) for blank leader (attachment) panels to the calculated area, and utilizing a four-panel design (See Subsection D.2.3.6), each wing is 288 m^2 (3097 ft^2), or approximately 3.5 x 82.3 m (11.5 x 270 ft). The total area of all four wings, 1152 m^2 (12,387 ft^2), is equivalent to about 5.3 tennis courts. This area would be larger if RTG waste heat were included.

This design does not include any excess array area to account for premature failures of part or all of a panel. In the baseline RTG design, three RTGs are included for power generation, with no spare RTG for redundancy. It has been assumed that a similar no-redundancy policy would be applied to the solar panels in an all-solar design.

The aspect ratio (length/width) of each panel is 23.4. This is well outside the mass-efficient range of the APSA array, aspect ratios from four to ten. Because no information exists about the design of an APSA array with a non-optimum aspect ratio, the estimates for APSA performance are based on optimum aspect ratio performance. However, the mass growth contingency for the solar array (see Subsection D.2.4.1) has been increased to account for the expected growth of the array due to non-optimum effects.

D.2.3.2 Secondary Batteries. The need to provide power during periods when the solar arrays are not illuminated mandates the addition of batteries to the spacecraft that are not needed for the RTG design. These so-called secondary batteries must be sized to provide sufficient power during worst case conditions, namely, long-duration Saturn occultations. Table D-2 lists typical Cassini off-Sunpoint events and required energy storage.

NiH_2 batteries were chosen for energy storage because they are more mass-efficient than standard NiCd batteries. Nine 50-amp-hour NiH_2 batteries are required to meet the energy storage requirements for large Saturn occultations at 81% depth of discharge (DOD). An additional battery is needed for redundancy according to Project fault protection requirements. Each battery weighs 34 kg (75 lb) and requires a battery control assembly (BCA), which weighs 2 kg (4.4 lb). The total mass increase due to these batteries is therefore:

10 batteries x 34 kg/battery + 10 BCAs x 2 kg/BCA = 360 kg

Table D-2. Typical Cassini Off-Sunpoint Events for All-Solar Option

Event	Required Duration (h)	Required Power (W-h)	Required Energy (W-h)	A-h	DOD for 950-A-h (%)	No. of Events
Saturn Orbit Insertion	3	837	2250	75	17	1
Saturn Occultations - Large** - Average	 14.5 10	 837 837	 10,875 7500	 326.5 250	 81 56	 13 10
Titan Occultation	0.3	837	225	7.5	1.6	6
Trajectory Correction Maneuvers	0.2	837	150	5	1.1	200

* Depth of discharge
** Driver for determining battery size requirements

However, this is not the only impact. Immediately following each occultation, the batteries must be recharged in preparation for the next discharge cycle. Assuming a worst-case period of 10 days between large occultations, approximately 45 W of additional power must be provided by the solar arrays to recharge the batteries.

The extra power is factored into the solar array sizing, and raises the solar array power requirement from 792 W to 837 W.

D.2.3.3 Power Regulation. The power subsystem for the all-solar Cassini spacecraft is a highly regulated 30-Vdc bus. A peak power tracker is used instead of an RTG shunt regulator to process the maximum power available from the solar array.

D.2.3.4 Additional Propellant and Changed Propellant Tank Size. Additional hydrazine propellant will be required for spacecraft propulsive maneuvers due to the addition of solar arrays. Based on the solar array size, an estimated 348 kg (767 lb) of additional monopropellant hydrazine is required to counteract solar torques. The increase in tank size to hold this additional propellant relates to an increase in dry spacecraft mass, which is not considered in this study, but is typically small in relation to the mass of the propellant required. More frequent thruster firings will be required to counteract the increased solar torque (either to reorient the spacecraft or unload the reaction wheels), and must be incorporated into the mission design.

D.2.3.5 Reaction Wheels. Due to the substantially increased spacecraft inertial resulting from the large solar arrays (see Subsection D.2.4.2), larger reaction wheels are incorporated into the spacecraft design. These larger wheels offer improved performance over the baseline wheels, but cannot match the turn times of the RTG-powered design.

D.2.3.6 Spacecraft Configuration. Figure D-1 simplistically illustrates the Cassini spacecraft configuration using the 1152 m^2 (12,387 ft^2) silicon APSA solar arrays. Note that the solar array mast canister is not included in Figure D-1, but scan platforms are.

Insufficient area exists on the spacecraft to mount 10 batteries. This means that a configuration change is necessary for additional surface area to mount these batteries, driving up the stack height, mass, and cost.

D.2.4 Spacecraft Effects

Using the array size calculated in Section D.2.3.1, the silicon solar arrays are incorporated into the spacecraft design. This subsection describes the additional changes and effects on the spacecraft design of incorporating silicon solar arrays.

D.2.4.1 Mass Changes. Table D-3 summarizes the mass effects of using silicon APSA solar arrays rather than RTGs. This table includes an option of articulating two of the panels (See Subsection D.2.5). The mass increase to incorporate the design changes is so large that this spacecraft is not feasible based on current U.S. launch vehicles and upper stages.

The component masses (i.e., current best estimates) are only part of the mass estimation process. The Cassini Project practice is to produce a best estimate of the mass for a component and calculate from that a contingency mass based upon component maturity to provide an allowance for mass growth as the item matures from a design concept to a final, space-qualified product. The contingency mass allocations are:

- 0% : Fixed -- absolute value allocated;
- 1 % : Measured -- actual hardware;
- 5 % : Calculated -- completed detailed design;
- 10% : Estimated -- conceptual design, first-order calculation;
- 20 % : Analogous -- functional equivalence to known hardware.

The mass values shown in Table D-3 have contingency added based on the maturity/knowledge of the hardware to allow for mass growth during the development life cycle. The deleted RTGs and associated masses and contingencies are what are currently carried in the RTG baseline. The silicon APSA array performance characteristics (see Subsection D.2.2.1) incorporate a 5% growth factor in addition to the contingencies noted above.

D.2.4.2 Increased Turn and Settling Times. The moments of inertia will be greatly increased for all axes on Cassini due to the addition of the four solar arrays. The turn times also show large increases and are nearly the same for BOM and EOM because the arrays are so large (i.e., the change in inertia due to propellant depletion has little impact). The estimated times for a full 360° turn about each axis are shown in Table D-4. These turn times are first-order estimates.

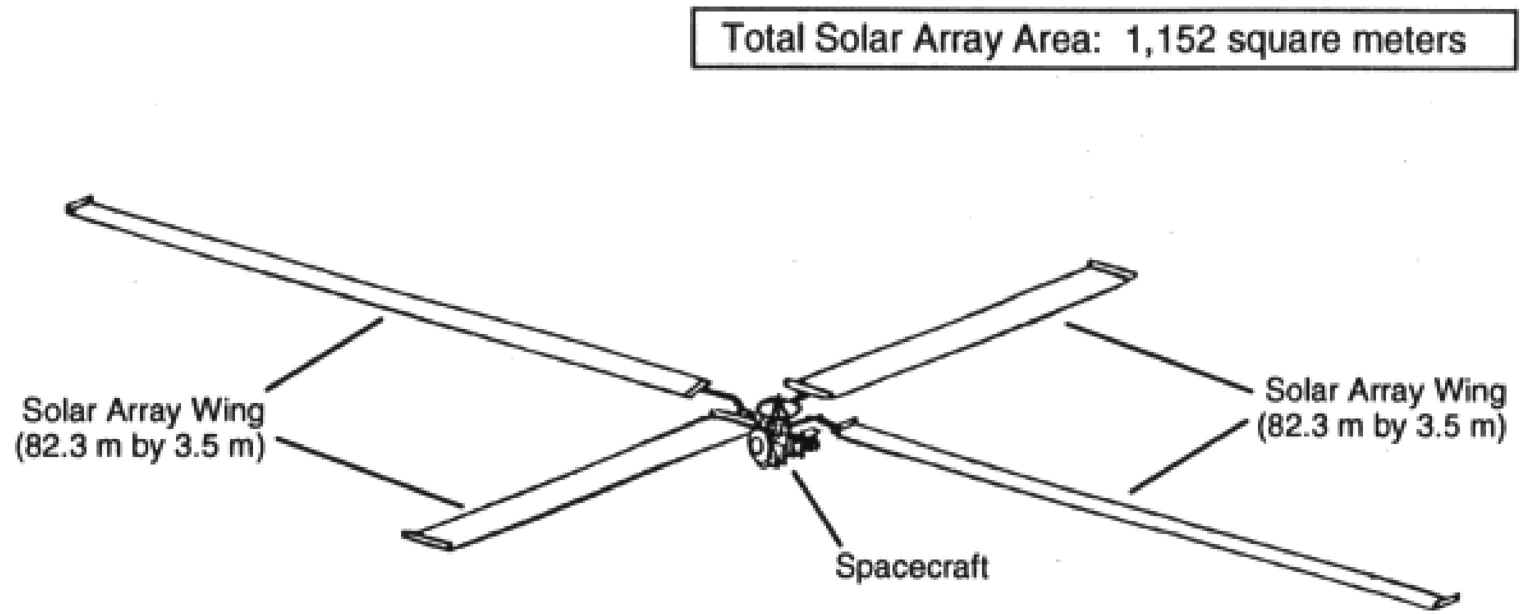

Figure D-1. Silicon APSA All-Solar Configuration for the Cassini Spacecraft

Table D-3. Mass Changes Table for Silicon All-Solar Cassini (kg)

Item	Current Best Mass Estimate	Contingency Mass
NO ARTICULATION OPTION		
Added silicon cell solar arrays	1350	203*
Added solar array support structure	121	24
Added solar deploy mechanism	10	2
Added HPSP ballast	15	8
Added peak power tracker (replaces shunt regulator)	4	0.8
Added UARS reaction wheels	48	4.8
Added ten 50-A-H batteries	340	34
Added ten battery control assemblies	20	4
Added replacement heaters and cabling (replaces RHUs)	1	0.2
Deleted three RTGs	-168	-1.7
Deleted RTG support structure and thermal blankets	-13	-2.6
Deleted RTG harness and launch support	-21	-4.1
Deleted shunt regulator (replaced by power tracker)	-4	-0.8
Deleted 117 RHUs and housings	-15	-1.5
Deleted baseline reaction wheels	-38	-1.9
Net Spacecraft Dry Mass Delta	1650	268
Total Spacecraft Dry Mass Delta (with contingency)	1918	
OPTION WITH ARTICULATION		
Added two solar array (SADA) actuators	45	0.5
Added two solar array drive electronics	13	0.2
Net Spacecraft Dry Mass Delta	1708	269
Total Spacecraft Dry Mass Delta (with contingency)	1977	

*15% contingency is carried to account for the non-optimum aspect ratio of the APSA array (see Subsection D.2.3.1).

One way to decrease turn times is to use larger reaction wheels. As a representative example, off-the-shelf reaction wheels used on the Upper Atmospheric Research Satellite were evaluated (Table D-4, last column); they provided better performance than those used in the RTG baseline. Table D-5 summarizes the comparison between the different reaction wheels (values are for one wheel). Such UARS reaction wheels are incorporated into the spacecraft design because of their faster spacecraft turn times.

D.2.4.3 Attitude Control. Pointing accuracy and stability were not thoroughly evaluated in this study, since an extensive simulation would be necessary and would require detailed information about array stiffness. However, the addition of large solar panels will have large impacts upon the pointing accuracy and stability of the spacecraft.

Table D-4. All-Solar Si Cassini 360° Estimated Turn Times* (hours)

Axis	BOM**	EOM***	Using UARS
X	145 (1.11)	144 (0.77)	48
Y	179 (0.99)	178 (0.49)	59.7
Z	288 (0.87)	288 (0.67)	96

* Turn times for RTG-powered spacecraft are given in parenthesis.
** Spacecraft inertia is approximately the same at BOM and EOM. Therefore, BOM and EOM turn times are almost identical.
*** The EOM values are generally smaller because the spacecraft is carrying 1ess propellant after rendezvous, but are about the same for the all-solar Cassini.

Table D-5. Comparison of Reaction Wheels (WARS vs. Baseline)

	UARS	Baseline
Angular Momentum (ft-lb-s)	60	20
Mass(kg)	12	9.5
Power; steady state (W)	24	22
Power; peak (W)	160	135

D.2.4.4 Launch Adapter. Due to the large increase in spacecraft mass, the launch vehicle adapter must be reinforced to withstand the larger launch loads. The mass of the reinforced adapter has been estimated to be approximately 400 kg.

D.2.4.5 Thrusters. Attitude control thrusters are normally fired in pairs. Turns about the X and Y axes are uncoupled; turns about the Z axis are coupled. Due to the addition of the arrays, the use of thrusters is of concern; the thruster's plumes would impinge on the solar arrays. For Cassini, using four solar panels, roll control is dramatically compromised.

D.2.4.6 Thermal Control. Several thermal control problems are caused by the addition of the four large solar panels:

(a) The field of view of engineering bus bays adjacent to the solar panels are blocked by the solar arrays; heat rejection capability is reduced by one-half.

(b) The dynamics of the solar panels as they move from shadow to sunlight conditions is unknown (important for attitude control).

(c) The high-precision scan platform is limited in elevation due to thermal constraints on the instrument radiators with less deep-space viewing.

These effects, as well as other thermal impacts, would necessitate a change of the spacecraft thermal design. The effects of such a redesign are not known.

D.2.5 Array Articulation Options

This subsection discusses the benefits and drawbacks associated with fixed and articulated solar arrays. The solar panels were sized assuming they were Sun-pointed (Subsection D.2.3.1). Off-Sun maneuvers will effect the amount of power generated by the arrays, unless the arrays are articulated to remain pointed at the Sun.

In the fixed-panel option, the solar arrays are rigidly mounted to the spacecraft structure. The active area of the solar arrays is aligned with the X-axis so that the solar arrays are normally Sun-pointed, keeping the batteries charged. When the spacecraft points away from the Sun (e.g., for a maneuver or science data gathering), the solar array power output decreases as a cosine angle loss, up to about 60° from the Sun. The power deficit caused by the reduced solar array output is then made up by the batteries. Each time the spacecraft must go off-Sunpoint, the batteries must be recharged after returning to a Sun-pointed direction. While body fixing the arrays is the simplest solution, there will be mission design and operational impacts due to constraints on off-Sun pointing (e.g., to minimize battery cycling).

Articulating the arrays will relieve many of the operational constraints on the mission—battery charging and discharging can be kept to a minimum and the mission profile will have more freedom in allowing off-Sun events. Placing single degree of freedom actuators on two of the four solar panels will allow two of the four wings to remain Sun-pointed while the spacecraft rolls 360° around the articulated axis. The additional mass of two actuators and drive electronics is 59 kg (130 lb), including contingency mass. However, there will be cases where a single degree of freedom on two of the arrays will not provide any advantage; off-Sun maneuvers that place the long axes of the articulated panels in a Sun-pointed direction will not benefit from articulation. To provide more flexibility in spacecraft pointing, more articulation can be added to the panels. For example, all four panels could be articulated in one axis, or some or all of the panels could be articulated in two axes (two-axis articulation may be difficult due to the large size of the arrays). If all four solar wings were articulated in one axis, four actuators and drive electronics would be required and 117 kg (258 lb) of additional mass would be added to the spacecraft.

Determining the appropriate amount of articulation (if any) would require a substantial amount of a effort in spacecraft, mission, and operational design. For this study, articulation has been left as an open item, although the mass impact of adding articulation to two of the for arrays is shown in Table D-3.

D.2.6 Cost

A cost estimate for the required silicon APSA arrays is shown in Table D-6. These costs represent the resources required to manufacture and test the minimum amount of hardware required for the design. In reality, spare components would need to be procured to prevent schedule slips during fabrication and test. The amount of spare components would need to be negotiated, and all additional components would represent increased cost.

Table D 6. Unarticulated All-Solar Cassini Si Array First-Order Cost Estimates (FY93 $)

• Total Estimated Cost: Four flight arrays**** $1520/W (BOL power) producing 148 kW** at BOL*** (148 kW) x ($1520/W)=$225 M	$225 M
Breakdown: • 1,065,000 thin cells ($38 cell price for one 2 x 4 cm cell) • Procurement cost estimates consider: - Subassembly costs, manufacturing, quality assurance - Array design, circuit layout, structure, etc. - Engineering, engineering tests - Testing: array, acceptance - Procurement vendor overhead, G & A*	 $40 M $185 M

* No JPL support fees are included in cost estimate.
** Power required at 1 AU (BOL) to meet mission requirements.
*** Historical cost of $1086/W basis with the assumption that thin cells costs increase overall costs ~20%, and large array size will cause ~20% overall cost increase (i.e., handling, tooling, fixtures, etc.)
**** Minimal sparing.

D.3 CASSINI GaAs SOLAR ARRAY

The second part of the study examines GaAs cell technology for power conversion. As for the silicon option, this section is broken into six major subsections: assumptions, APSA performance, spacecraft design changes, performance impacts, solar array articulation, and predicted cost deltas. Many of the items discussed for the GaAs option are identical to those in the silicon study.

D.3.1 Assumptions

The assumptions used in the GaAs analysis are identical to those used in the silicon analysis. See Subsection D.2.1 for more detail.

D.3.2 GaAs APSA Performance

This section details the performance characteristics of the GaAs APSA array and the mission specific factors which impact APSA performance.

D.3.2.1 Nominal Performance and Radiation Degradation. Based upon the predicted performance of a GaAs APSA array in geosynchronous orbit, the end of life performance for the Cassini GaAs APSA array is expected to be:

Specific power (12.6 y, 1 AU, RDM 2) = 96 W/kg

Power density (12.6 y, 1 AU, RDM 2) = 124 W/m^2

(These estimates include the required RDM = 2 margin.)

D.3.2.2 Insolation. The insolation for the GaAs array at Saturn is the same as the silicon array—1.2 percent (0.012) of that at Earth.

D.3.2.3 Low Intensity. Low Temperature. Unlike silicon cells, GaAs cells do not experience a significant change in conversion efficiency due to low temperature operation. Limited test data is available about low intensity lighting degradation for GaAs cells. The limited tests that have been performed have shown that GaAs cells do not seem to suffer from the anomalous LILT behavior exhibited in silicon cells. Based upon this limited information, this study assumes that low intensity degradation will not play a role in GaAs APSA performance. Table D-7 shows the estimated performance of GaAs cells under the conditions which would be encountered at Saturn.

D.3.2.4 Additional Acceleration Loading. The specific power derating factor for the GaAs array is the same as for the silicon array—0.88. Subsection D.2.2.5 provides more detail about acceleration loading.

D.3.2.5 Estimated GaAs APSA Performance. Utilizing these factors, the specific power and power density for Saturn operating conditions are determined as follows:

Specific power = [Specific power (12.6 y, 1 AU, RDM 2)] x
[Insolation factor] x [LILT factor] x
[Structural beef-up factor]
=96W/kg x 0.012 x 0.95 x 0.88
= 0.96 W/kg

Power density = [Power density (12.6 y, 1 AU, RDM 2)] x
[Insolation factor] x [LILT factor]
= 124W/m^2 x 0.012 x 0.95
= 1.41 W/m^2

D.3.3 Spacecraft Design Modifications

This section addresses the first order design modifications required to implement an all solar design using GaAs cells.

Table D-7. Estimated Change in GaAs APSA Power Output Due to LILT: Cassini EOM at Saturn*

Low intensity at low temperature (LILT) power degradation	1.0
Low temperature conversion efficiency (voltage change)	0.95
LILT factor (power degradation x conversion efficiency)	0.95

* There is no test data for distances of 9.0 AU and greater. These values were extrapolated from curves for 1 to 5 AU (Tada, 1982; Anspaugh, 1989).

D.3.3.1 GaAs APSA Size. The power requirement for GaAs arrays is the same as for silicon arrays -- 837 W (see Subsection D.2.3.1 for details). Using the performance estimates for a GaAs APSA array at Saturn, the predicted mass and area of the arrays is:

Mass: (837 W)/(0.96 W/kg) = 872 kg

Area: (837 W)/(1.41 W/m^2) = 593 m^2

This mass estimate is used to calculate the total change in the spacecraft dry mass necessary to accommodate GaAs solar arrays. Adding 5 m^2 (53.7 ft^2) for attachment (blank leader) panels to the calculated area, and utilizing a four-panel design (See Subsection D.3.3.6), each wing is 149.5 m^2 (1607 ft^2), or approximately 3.5 x 42.7 m (11.5 x 140 ft). The total area of all four panels is 598 m^2 (6430 ft^2), 2.7 times the area of a tennis court or about 12 percent of a football field.

The aspect ratio (length/width) of each panel is 12.6. This is outside the mass-efficient range of the APSA array, with aspect ratios from four to ten. Because no information exists about the design of an APSA array with a non-optimum aspect ratio, the estimates for APSA performance are based on optimum aspect ratio performance. However, the mass growth contingency for the solar array (See Table D-8) has been increased to account for the expected growth of the array due to non-optimum effects.

D.3.3.2 Secondary Batteries. The energy storage requirements are the same as those in the silicon study. Therefore, ten 50-amp-hour NiH_2 batteries are required to meet the energy storage/fault protection requirements for the mission. The total mass increase is 360 kg (794 lb). Subsection D.2.3.2 presents more detail about secondary battery sizing.

D.3.3.3 Power Regulation. See Subsection D.2.3.3.

D.3.3.4 Additional Propellant and Changed Propellant Tank Size. Based on the solar array size, an estimated 199 kg (439 lb) additional monopropellant hydrazine is required to counteract solar torques. See Subsection D.2.3.4 for more detail.

D.3.3.5 Reaction Wheels. Despite lower spacecraft inertial than the silicon design, larger reaction wheels are still recommended to reduce spacecraft turn times. Therefore, UARS wheels have been incorporated into the design. See Subsections D.2.3.5 and D.3.4.2 for further information.

D.3.3.6 Spacecraft Configuration. Figure D-2 simplistically illustrates the Cassini spacecraft configuration using the 598 m^2 (6430 ft^2) GaAs APSA solar arrays. Note that the solar array mast canister is not included in this figure, but scan platforms are.

D.3.4 Spacecraft Effects

Using the array size from Subsection D.3.3.1, the GaAs solar arrays are incorporated into the spacecraft design. Additional changes and effects on the spacecraft design resulting from this incorporation are described below.

D.3.4.1 Mass Changes. The changes necessary to equip the spacecraft with GaAs solar arrays instead of RTGs are shown in Table D-8. This table includes an option of articulating two of the panels (see Subsection D.2.5). The mass increase to incorporate the design is so large that this spacecraft is not feasible based on current U.S. launch vehicles and upper stages.

The mass values shown in the mass changes table (Table D-8) have contingency added based on the maturity/knowledge of the hardware to allow for mass growth during the development life cycle. A discussion of the amount of contingency used for various levels of design maturity is presented in Subsection D.2.4.1. The deleted RTGs and associated masses and contingencies are what are carried in the RTG baseline. The GaAs APSA array performance characteristics (See Subsection D.3.2.1) incorporate a 5% growth factor in addition to the contingencies noted in D.2.4.1.

D.3.4.2 Increased Turn and Settling Time Estimates. The moments of inertia about two axes will be less for the GaAs solar array than the Si array, due to the reduction in wing length. The 360° turn times for the three spacecraft axes are shown in Table D-9. Turn times for UARS reaction wheels are also shown (the baseline RTG turn times are shown in parentheses for comparison).

D.3.4.3 Attitude Control. See Subsection D.2.4.3.

D.3.4.4 Launch Adapter. Due to the large increase in spacecraft mass, the launch vehicle adapter must be reinforced to withstand the larger launch loads. The mass of the reinforced adapter has been estimated to be approximately 350 kg (772 lb).

D.3.4.5 Thrusters. See Subsection D.2.4.5.

D.3.4.6 Thermal Control. See Subsection D.2.4.6.

D.3.5 Array Articulation: See Subsection D.2.5

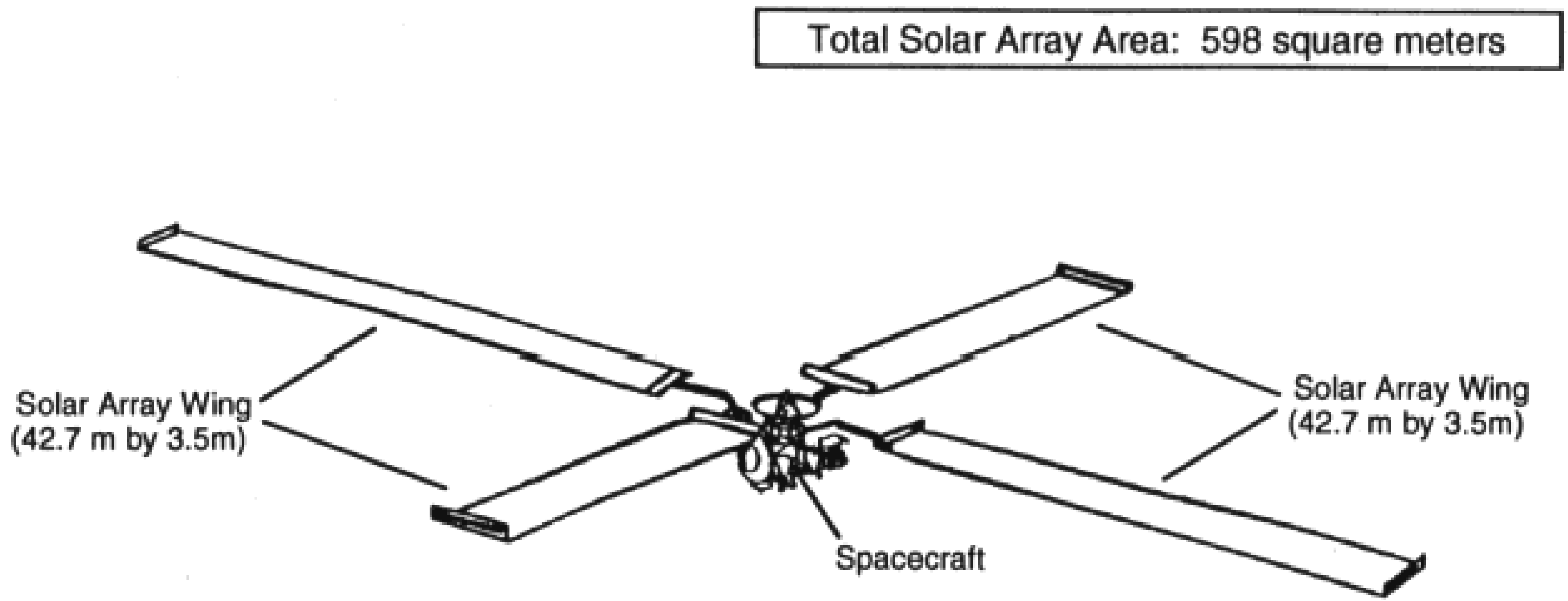

Figure D-2. Gallium Arsenide/Germanium (GaAs/Ge) APSA All-Solar Configuration for the Cassini Spacecraft

Table D-8. Mass Changes Table for GaAs All-Solar Cassini Spacecraft (kg)

Item	Current Best Mass Estimate	Contingency Mass
NO ARTICULATION OPTION		
Added silicon cell solar arrays	872	131*
Added solar array support structure	95	19
Added solar deploy mechanism	10	2
Added HPSP ballast	15	8
Added peak power tracker (replaces shunt regulator)	4	0.8
Added UARS reaction wheels	48	4.8
Added ten 50-A-H batteries	340	34
Added ten battery control assemblies	20	4
Added replacement heaters and cabling (replaces RHUs)	1	0.2
Deleted three RTGs	-168	-1.7
Deleted RTG support structure and thermal blankets	-13	-2.6
Deleted RTG harness and launch support	-21	-4.1
Deleted 117 RHUs and housings	-15	-1.5
Deleted shunt regulator	-4	-0.8
Deleted baseline reaction wheels	-38	-1.9
NET SPACECRAFT DRY MASS DELTA	1146	191
TOTAL SPACECRAFT DRY MASS DELTA (with contingency)	1337	
OPTION WITH ARTICULATION		
Added two solar array (SADA) actuators	45	0.5
Added two solar array drive electronics	13	0.2
NET SPACECRAFT DRY MASS DELTA	1204	192
TOTAL SPACECRAFT DRY MASS DELTA (with contingency)	1396	

* 15% contingency is carried to account for non-optimum aspect ratio of the APSA array (refer to Subsection D.3.3.1).

Table D-9. All-Solar GaAs Cassini 360° Estimated Turn Times* (hours)

Axis	Using Baseline Reaction Wheels	Using UARS Reaction Wheels
X	27.6 (1.11)	9 (0.77)
Y	33.5 (0.99)	11.2 (0.49)
Z	53.4 (0.87)	17.8 (0.67)

* Turn times for RTG-powered spacecraft are given in parentheses.

D.3.6 Cost

A cost estimate for the GaAs APSA arrays is shown in Table D-10. Total spacecraft cost deltas are presented in Table D-13.

D.4 ALL-SOLAR CASSINI SPACECRAFT OPTIONS COMPARISON

D.4.1 Mass

Mass is one of the major spacecraft resources that must continually be maintained and controlled. The mass increases associated with both types of solar arrays (Si and GaAs) are summarized in Table D-11.

D.4.2 Increased Turn Times

Table D-12 summarizes the turn times for the original baseline (RTG), Si ASPA, and GaAs APSA designs.

Table D-10. Unarticulated All-Solar Cassini GaAs Solar Array First-Order Cost Estimates (FY93 $)

• Total Estimated Cost: Four flight arrays**** $1925/W (BOL power) producing 99 kW** at BOL*** (99 kW) x ($1925/W) = $191 M	$191 M
Breakdown: • 574,000 thin cells ($135 cell price for 2 x 4 cm cell) • Procurement cost estimates consider: - Subassembly costs, manufacturing, quality assurance - Array design, circuit layout, structure, etc. - Engineering, engineering tests - Testing: array, acceptance - Procurement vendor overhead, G&A*	 $77 M $114 M

* No JPL support fees are included in cost estimate.
** Power required at 1 AU (BOL) to meet mission requirements.
*** Historical cost of $1086/W basis, with the assumption that thin cells costs increase overall costs ~20 percent, and large array size will cause ~20 percent overall cost increase (i.e., handling, tooling, fixtures, etc.). Higher GaAs cell costs also incorporated.
**** Minimal sparing.

Table D-11. Cassini Solar Options Total Mass Increase Summary (kg)

Mass increase for solar option vs. RTG baseline*	Spacecraft Power Type			
	Silicon APSA Array		GaAs/Ge APSA Array	
	Articulation	No Articulation	Articulation	No Articulation
Without Contingency	1861	1817	1344	1296
With Contingency	2146	2105	1551	1504

*Original, or nominal, RTG spacecraft used as the reference for computing mass changes for original all-solar option.

Table D-12. Cassini All-Solar Options Estimated 360° Turn Times Comparison

Axis	360° Turn Times, hours		
	Power Type		
	RTG	Si APSA*	GaAs APSA*
X	1.11/0.77	48/48	9/9
Y	0.99/0.49	60/60	11/11
Z	0.87/0.67	96/96	18/18

*UARS reaction wheels used.

D.4.3 Cost

Table D-13 shows spacecraft, testing, and ground support cost deltas for the two array options. Some costs cannot be understood without significant design effort, such as spacecraft redesign costs, cost impacts on subsystem designs due to the presence of large solar arrays, increased operational costs, etc., and are not included.

D.5 Conclusion

The masses of the silicon APSA and the GaAs APSA arrays are both substantially higher than the original, RTG-powered Cassini dry spacecraft mass allocation. Both options are too massive to be launched on the baseline Titan IV (SRMU)/Centaur vehicle.

In addition, the all-solar Cassini spacecraft is much more complex than the RTG-powered spacecraft. Operational complexity will increase with the addition of solar arrays; specific examples of this include longer turn and settling times, a more comprehensive mission design plan, and spacecraft thermal blockages. Fault protection will be more complex with the larger solar array because its larger inertial characteristics determine allowable no-power input times (spacecraft on batteries) for off-Sun-pointing maneuvers, Sun acquisition, and slewing to a Sun-pointed configuration.

Table D-13. Unarticulated All-Solar Cassini Cost Deltas* (FY93 $)

Affected Spacecraft Subsystem	Silicon APSA	GaAs APSA
1. Power regulator: Design development and text a. Hardware build, spares, etc.	 1,357,500 162,900	 1,357,500 162,900
2. Batteries: 10-50 A-H Ni-hydrogen Integ/test cells into batteries	 5,000,000 500,000	 5,000,000 500,000
3. Solar Panels: Silicon APSA GaAs/Ge	225,000,000	191,000,000
4. RTG H/W (3)	(117,100,000)**	(117,100,000)**
Testing Effects		
1. Solar panel simulator	81,500	81,500
2. Solar panel testing & integration deploy & illumination	1,086,000	1,086,000
3. Battery charging at the launch pad	159,600	159,600
4. High-power cables through T-O umbilical	54,300	54,300
5. Flight operations (software modifications)	434,400	434,400
TOTAL COST DELTA	$116,736,200	$82,736,200

*These numbers have been rounded to reflect the large degree of uncertainty inherent in the cost estimates. Additionally, the operations cost to include the solar array constraints in each maneuver will be high.

**In the baseline, the Project plans to hold the GLL/ULS spare RTG (F5) as a spare for Cassini. The spare converter from GLL/ULS (E2) will be fueled (F2) for Cassini. Two new RTGs will be built for Cassini (F6 and F7). The total RTG cost to NASA under this plan is $117.1 M (FY93 $). This RTG cost does not reflect DOE’s costs.

APPENDIX E

VARIATIONS ON THE NOMINAL ALL-SOLAR DESIGN

E.1 CONCENTRATOR PHOTOVOLTAIC ARRAY DESIGNS

Use of concentrators was examined for the Cassini mission. Because of its large distance from the Sun (9.0 to 9.3 AU for the nominal mission), a relatively high-concentration ratio concentrator was assumed, since it could produce intensities at the solar cell much closer to what is observed at Earth (or Mars). Considering the solar photovoltaic power supply alone, this would lead to a substantial decrease in the number of solar cells required, yet also provide for cell operation at illumination intensities where significant amounts of existing data are available (i.e., LILT effects are not critical).

A survey of concentrator efforts indicates that few designs are directly available for space applications. Those which have achieved minimal development hardware status include two reflective and one refractive designs. Reflective designs have been developed primarily for military applications and are hardened for natural and human-made threats. As a result, these designs—the minicassegranian (first design) and Solar Linear Array Technology System, or SLATS (second design)— are quite massive, with specific powers (power/ mass) at Earth approximately 1/6 that of the planar APSA design. The deployed area of the concentrator designs are generally somewhat smaller than the planar design, since concentrator systems tend to operate at a higher efficiency than unconcentrated cells. Part of this improvement is, however, mitigated by non-ideal optics and material reflectances. A size reduction of approximately 10 to 20% compared to a planar array is probably feasible.

The third design, the minidome fresnel system, has been developed primarily for the natural space environment and is projected to be less massive, although the technology is still not mature enough for confident projections of the performance of a large array system.

Because of the lack of suitable data for large-scale performance characteristics of any of the concentrator systems, a number of strawman estimates/ assumptions were made to facilitate a hypothetical concentrator design. In particular, it was assumed that designs could be developed capable of launch stowage and orbital deployment that would have a specific power approximately 1/3 that of the planar APSA array (50 W/kg at Earth). The areal power density would be roughly 15% higher than a planar array. The concentration ratio would be about 50 to 1.[1] (The latter figure is not used by any of the systems discussed above; it represents a mid-range value that is used here for costing purposes.)

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[1] Note that at a 50:1 concentration ratio, the concentrator arrays must be pointed to the Sun within an accuracy of ±l-2%. New technology may be able to extend this range to about ±4% with additional expenditure of time and money. Outside of this pointing, the output rapidly goes to zero because of the concentrator cell characteristics (i.e., the concentrated solar beam will be displaced away from the photovoltaic cells.

Concentrator costs (FY93 dollars) were estimated very coarsely from the planar array value by considering the impact of a reduced quantity of solar cells (approximately two percent of the planar quantity) and the increased complexity of the concentrator hardware. A rough estimate is appropriate, since there is presently no well-developed and tested concentrator design available (as mentioned above) that is directly applicable to the Cassini mission. Cell costs were calculated for the reduced quantity of cells required, although the individual cell cost was increased to $65 for silicon and $385 for GaAs to account for the non-standard design and fabrication processing of the concentrator cells. The use of a concentrator array will also significantly increase costs for nonrecurring design and analysis, array structure fabrication, and testing. It was estimated that the increase would be roughly $25 M for either the silicon or GaAs array. This leads to a total estimated cost of $159 M for a silicon concentrator array (compared to $225 M for a planar silicon array) and $163 M for a GaAs concentrator array (compared to $191 M for a planar GaAs array). It does not include any costs that could be incurred for more precise pointing, whether by actuators or spacecraft precision pointing. As is evident, the first-order cost estimate differential is negligible for the silicon array and significant for the GaAs array, due to the impact of cell cost (low for silicon and high for GaAs) on the overall array cost, and other estimated costs.

There are a number of concerns that need to be resolved prior to using any concentrator. Concentrators for Earth use are designed to remove the excess cell heat due to concentration so that the cell can operate at a safe temperature. The thermal requirements will vary widely from 0.63 AU to 9.3 AU EOM. In order to prevent overheating at 1 AU, the concentrator array will have excessive thermal mass for use at 9.3 AU. The behavior of the optics, reflective and refractive, over the mission lifetime are not known. However, small changes in the mirror/lens condition (i.e., yellowing, aging, sagging) can lead to relatively large power losses. Alignment of the concentrator elements is critical, since these systems will not provide power if the Sun incident angle is off normal by more than two or three degrees. A more difficult issue to estimate is the build-up of dust—interstellar and Saturnian—on the optics. Concentrator performance depends on clear, unobscured optics. On the Earth, build-up of dust on concentrator optics reduces array performance nonlinearly and significantly. The stowage of a concentrator array for launch, considering the large array area needed for the Cassini mission, is a formidable problem. Concentrator elements are generally two orders of magnitude thicker than the planar array blanket.

Finally, ground testing of large area concentrator arrays is expected to be much more complicated than for planar arrays (which in themselves require the use of expensive specialized testing and fixtures). As one example, since concentrator element alignment is critical, it will be necessary to verify that the wing alignment in the Earth's 1 g environment is the same as in 0 g of space. For a very large array wing, this will entail substantial development of ground support fixtures that will not introduce array wing distortions. Also, vibration testing of the concentrator array would be necessary to verify the optical alignments after launch.

APPENDIX F

LOWER MASS, REDUCED SCIENCE ALL-SOLAR DESIGN

To assess the sensitivity of spacecraft mass with respect to science capability, a second design point was chosen which reduces spacecraft mass by reducing science capability. This design is based on the redesigned Cassini, in which the science platforms are body-fixed, eliminating the deployable booms and platform actuators. In addition, the power available to science instruments is further reduced by 50%, beyond the Cassini redesign, to reduce the size of the solar arrays, at the expense of the elimination of some instruments from the payload or a reduction in data gathering opportunities for each instrument.

F.1 ANALYSIS

The analysis methodology used in the second design study was the same as that used for the first. Estimates of technical performance (e.g., solar array specific density, reaction wheel momentum storage, etc.) were also identical to those used in the first study, as were all design assumptions, except as noted. For a detailed description of the methodology, performance, and assumptions used in the first study, see Section 4.2 and Appendix D.

The analysis for the fixed platform design examines only the use of GaAs cells in the solar array, as the purpose of the second design was to minimize mass. As illustrated in Appendix D, a design utilizing silicon cells would be substantially heavier than a design using GaAs.

F.1.1 Solar Array Sizing

The performance predictions used to size the array were:

Specific power = 0.96 W/kg
Power density = 1.41 W/m^2

The average power required for the highest power science mode for Cassini at Saturn is approximately 564 W, including 229 W for the instrument payload. Reducing the science allocation by half to further reduce spacecraft mass yields a requirement of 450 W for the engineering and science subsystems. The redesigned Cassini also used 101 W of RHU heater power, which must be replaced by electric heaters when all radioactive isotopes are removed from the design.[1]

[1] Although the current best estimate of the number of RHUs for Cassini is between 130 and 230, the figure 101 was used for this reduced-science estimate.

The original analysis stated that 45 W of power are required for bakery charging (Section D.2.3.2). It may be possible to reduce this power to a lower level, around 10 W, if the Saturn tour phase of the mission can be appropriately redesigned. This assumption would require further analysis before confirmation, but because an aggressive mass reduction approach was used for this design, it was assumed that 10 W would be sufficient.

The use of a lower battery charging power may impact science observation times. Due to the long turn times for the all-solar design (Subsection F.1.4), science observation times may need to be reduced to permit sufficient Sun-pointed time to recharge the batteries. Articulated arrays could partially alleviate this problem, but would result in increased spacecraft mass (Subsection F.1.5). A more detailed study would be needed to estimate the impacts to observation times. The total power requirement at Saturn, not including RTG waste heat (Subsection 4.4.1), is therefore:

561 W = 450 W + 101 W + 10 W

The solar array mass and area is therefore:

Mass: (561 W) / (0.96 W/kg) = 585 kg
Area: (561 W) / (1.41 W/m^2) = 397 m^2

(The mass and area would be greater if RTG waste heat amounts were included.)

F.1.2 Battery Sizing

Similar to the original all-solar analysis, the second design point will require batteries to provide power during periods when the solar arrays are not illuminated by the Sun. The battery size is determined by the largest Saturn occultation (14.5 hours). Seven 50 A*hr batteries are required to provide power during the long Saturn occultation. One additional battery is required to meet redundancy requirements. A battery control assembly (BCA) is also required for each battery, at 2 kg each. The total mass increase due to batteries is therefore:

Mass: 8 batteries * 34 kg/battery + 8 BCAs * 2 kg/BCA = 288 kg

F.1.3 Dry Mass Delta

The change in dry mass of the spacecraft for this option is shown in Table F-1. See Appendix D (Subsection D.2.4.1) for a description of how contingencies were estimated.

Table F-1. Mass Changes for Reduced Science, All-Solar Cassini (kg)

Item	Current Best Mass Estimate	Contingency Mass
FIXED ARRAY		
Added GaAs solar array	585	88
Added solar array structure support	70	14
Added solar deploy mechanism	10	2
Added UARS reaction wheels	48	5
Added peak power tracker (replaces shunt regulator)	4	0.8
Batteries	272	27
Battery control assemblies	16	3.2
Added replacement heaters and cabling (replaces RHUs)	1.0	0.2
Deleted three RTGs	-168	-1.7
Deleted RTG support structure and thermal blankets	-13	-3
Deleted RTG harness and launch support	-21	-4.1
Deleted 101 RHUs and housings	-13	-1.3
Deleted shunt regulator	-4	-0.8
Deleted baseline reaction wheels	-38	-1.9
NET SPACECRAFT DRY MASS DELTA	749	127
NET SPACECRAFT DRY MASS DELTA, with contingency	876	
ARTICULATED ARRAY		
Added two solar array actuators	45	0.5
Added two solar array drive electronics	13	0.2
NET SPACECRAFT DRY MASS DELTA	807	128
NET SPACECRAFT DRY MASS DELTA, with contingency	935	

F.1.4 Turn Time Estimates

The addition of solar arrays increases the time required for spacecraft sums. Because the instruments are body-fixed, these longer times will increase the time needed for imaging mosaics, cause loss of instrument observation time during periods of communication with Earth, and have other impacts upon science observations. The turn times for this design, using both the baseline and UARS reaction wheels, are illustrated in Table F-2.

F.1.5 Array Articulation

The articulated platform design of Appendix D implements a single degree of articulation for two of the four solar panels, requiring two solar array actuators. However, removing the ability to point the instruments independent of spacecraft orientation may impact the ability of the arrays to remain normal to the Sun during science observation periods. Additional actuators may be required for the other two panels to maintain the necessary orientation of those panels, and it may be necessary to articulate some or all of the panels in two axis. The addition of more actuators will increase spacecraft mass.

F.1.6 Propellant Impacts

Based upon the solar array size, an estimated 145 kg (320 lb) additional monopropellant hydrazine is required to counteract solar torque.

F.1.7 Conclusion

The total spacecraft flight mass (without propellant for ΔV maneuvers) is summarized in Table F-3. The total flight mass, including propellant, is presented in Table 4-6, page 4-26. As shown in this first-order analysis, the spacecraft is too massive to be launched using the baseline launch system. Consideration of more detailed effects would increase spacecraft mass and make such a vehicle even more unfeasible. The scientific cost associated with this lower mass design is assessed in Section 4.4.

Table F-2. Estimated 360° Turn Times for Reduced Science, All-Solar Cassini (hrs)

Axis	Using Baseline Reaction Wheels	Using UARS Reaction Wheels
X	7.9	2.8
Y	9.8	3.5
Z	15.7	5.2

Table F-3. Dry Flight Mass of Reduced Science, All-Solar Cassini (kg)

Spacecraft Allocation	2150
Probe	306
Probe Support Equipment	46
Hydrazine Fuel (monopropellant)	207
Additional Dry Mass For All-Solar Option*	881
TOTAL	3590

*Two solar panels articulated in one axis.

APPENDIX G

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Cassini Environmental Impact Statement
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Earth Swingby Plan Supplement