Interplanetary Missions for the Late Twentieth Century

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A set of missions that is feasible during the remainder of this century and which will significantly contribute to our understanding of the solar system is presented. Targets include comets, asteroids, the outer planets and their satellites, and Mars. Mission opportunities for these targets are described in terms of launch vehicle, propulsion, and flight-time requirements, as well as other mission constraints such as launch payload margin and launch period objectives. Examples of encounter designs are also given.

I. Introduction

THIS paper discusses some of the planning which the National Aeronautics and Space Administration (NASA) has done for planetary exploration missions in the 1990's. The focus is on mission opportunities to the planetary bodies that are of greatest scientific interest. These are the periodic comets, asteroids, Saturn and Titan, and Mars. The principal technical constraints to planning these missions are the timing of the launch opportunities, based on the motion of the planets, and the limits on the propulsive capability, in particular, the available upper stages for launches from the Space Shuttle.

The most important constraint on planning for future planetary missions is budgetary. By the early 1980's, NASA's planetary program faced an uncertain future as fiscal constraints had lead to infrequent starts of increasingly expensive projects. NASA's plan for re-establishing a stable base for the planetary program is based on the 1983 report of the Solar System Exploration Committee (SSEC).[1]

The missions described in this paper were studied at the Jet Propulsion Laboratory during FY82 in support of the Mariner Mark II development project. Mariner Mark II would be a reconfigurable planetary spacecraft capable of flying several different missions at the end of this century. The costs for a series of Mariner Mark II missions would be reduced by the reuse of hardware designs, software, and ground operations systems.

A description of appropriate mission opportunities for the late twentieth century is the heart of this paper. Mission design guidelines and assumptions used to select the mission opportunities are described in Sec. II. Mission opportunities are discussed in Sec. III, and examples of encounter designs are given in Sec. IV.

II. Guidelines and Assumptions

Missions

The following six types of missions, reflecting the interest of the scientific community, are considered:

1) Comet Rendezvous
2) Comet Flyby/Sample Return
3) Near-Earth and Main Belt Asteroid Rendezvous
4) Saturn Orbiter/Titan Flybys
5) Titan Flyby/Titan Probe
6) Mars Orbiter

Launch Vehicle

Launch vehicle performance is a key constraint in determining the mission opportunities. This study assumes the use of the Space Shuttle with the largest upper stages available for injecting the spacecraft on the heliocentric trajectory. These upper stages are the Inertial Upper Stage (IUS) and two versions of the wide-body Centaur vehicle, which are being developed for launches from the Space Shuttle by NASA and the U.S. Air Force. The two Centaur versions are the Centaur G (20 ft long) and the more capable Centaur F (28 ft long). Figure 1 displays payload performance vs launch energy per unit mass, C_3, for the IUS and Centaur upper stages. Performance curves for the systems with or without the addition of a Star-48 kickstage are also displayed, along with typical capture requirements for the types of mission considered.

Performance Margin

Redesign can be kept to a minimum if the original design provides enough margin. This feature can be seen in Fig. 1 where the required launch energy, launch energy/injected payload combinations fall well below the launch vehicle capability curves. Where possible, a launch vehicle margin of 20% has been designed for and is a limiting factor in identifying candidate mission opportunities. Launch vehicle margin for a given C_3 is the percentage of the launch vehicle capability not required by the mission.

Launch Vehicle-Spacecraft Adapter Mass

The interface structural mass required to mate the spacecraft to the launch vehicle, called the adapter mass, is assumed to be 5% of the required launch vehicle injected mass at any particular launch energy and is included in performance calculations. Historically, the adapter mass has been between 2 and 5%.

Spacecraft

The spacecraft mass is assumed to be 600 kg, exclusive of propellant and propulsion system. The Titan Flyby/Titan Probe mission will carry an additional 200-kg Galileo-class atmospheric probe. The heat shield mass for the Titan probe is considerably lighter than the Jupiter probe. The Comet Flyby/Sample Return mission includes an additional 400 kg of payload for an Earth-return capsule (about 200 kg), the sample return collection device/dust shield (about 150 kg), and about 50 kg has been allotted for interface requirements. The resulting net spacecraft mass for the Comet Flyby/Sample Return is 1000 kg.

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Propulsion System

Deep space propulsion is an important cost factor for missions requiring large post-launch ΔV capability. Such missions include Orbiters and those requiring gravity-assist or similar maneuvers. The mission analysis in this paper assumes a single-stage, Earth-storable propulsion system with a specific impulse, I_{sp}, of 310 s. The other factors in the propulsion system model are a tankage factor of 15.7% and rocket engine inerts of 66 kg. With this model, the equation for the net spacecraft mass is:

$$M_N = M_0\{(1+A)\exp[-(\Delta V/gI_{sp})]-A\}-B$$

where M_N is the net spacecraft mass exclusive of propellant and propulsion system; M_0 the initial mass; A the tankage factor, $=0.157$; ΔV the impulse in km/s; B the engine inerts, $=66$ kg; and g the Earth surface gravity.

III. Mission Opportunities

Mission opportunities for the set of six mission types are identified and described below.

Comet Rendezvous Missions

Rendezvous targets are chosen on the basis of their interest to comet scientists and their accessibility. Table 1 lists suitable periodic comets along with the time of perihelion passage and viewing conditions from Earth before perihelion passage.

It is possible to find a rendezvous opportunity for all comets given in Table 1. Trajectory modes considered are direct (D), Earth gravity-assist (ΔVEGA), and Jupiter gravity-assist (JGA). A direct trajectory is preferred because of its shorter flight time. A ΔVEGA trajectory is approximately 2 yr longer than the direct trajectory because the first revolution of the transfer orbit (about a 2-yr period) must be shaped optimally for the return to Earth for the gravity-assist. Rendezvous mission opportunities and the associated mission parameters are summarized in Tables 2a and 2b on a yearly basis starting in 1989. Only one mission per year is selected. Where there is more than one mission in a year, the one considered to be most attractive is given; hence, the absence of D'Arrest and Tuttle-Giacobini-Kresak (TGK) launch opportunities. On a yearly basis, missions that can be launched with a Centaur and an IUS Two-Stage/Star-48 are displayed separately. Note that missions possible with an IUS Two-Stage/Star-48 can always be carried out with a Centaur.

The arrival date at the comet is chosen 50 days before time of perihelion T_P, unless it is better to arrive earlier from the standpoint of performance. In each case, the latest launch year which allows delivery of a 600-kg net spacecraft mass is chosen. In some cases, it is possible to launch a year or two earlier and increase the performance margin. All mission parameters are chosen to minimize the total ΔV, which is the sum of launch ΔV and the post-launch ΔV (ΔV_{PL}). Included in the latter is a 150-m/s allowance for navigation. The flyby altitude constraint at Earth for Earth gravity-assist trajectories is 300 km.

Enroute to a comet, there is a good change of encountering one or more asteroids. As an example, the spacecraft comes within 0.1 a.u. of nine asteroids during the Honda-Mrkos-Pajdusakova (HMP) (1995) mission. The ΔV required for targeting to one or more of these bodies is to be determined. However, the flyby velocities are all greater than 10 km/s, a difficult challenge to achieving meaningful scientific return.

Table 1 Comets for Rendezvous and Flyby/Sample Return missions

Comet	Abbreviation	Perihelion passage	Perihelion, au	Viewing from Earth
Encke	ENK	2/9/94	0.33	Good
		5/23/87		Fair
		9/5/00		Fair
Tempel 2	T2	3/17/94	1.48	Poor
		9/8/99		Good
Honda-Mrkos-Pajdusakova	HMP	9/13/90	0.54	Very good
		12/30/95		Excellent
		4/6/01		Poor
Kopff	KPF	7/2/96	1.57	Excellent
Wild 2	W2	5/7/97	1.58	Fair
D'Arrest	DA	7/27/95	1.35	Excellent
Churymov-Gerasimenko	C-G	1/18/96	1.30	Excellent
Tuttle-Giacobini-Kresak	TGK	7/28/95	1.05	Fair
Additional comets for Sample Return Mission				
Tuttle	TUT	6/27/94	1.00	Poor
Borrelly	BOR	10/31/94	1.40	Good

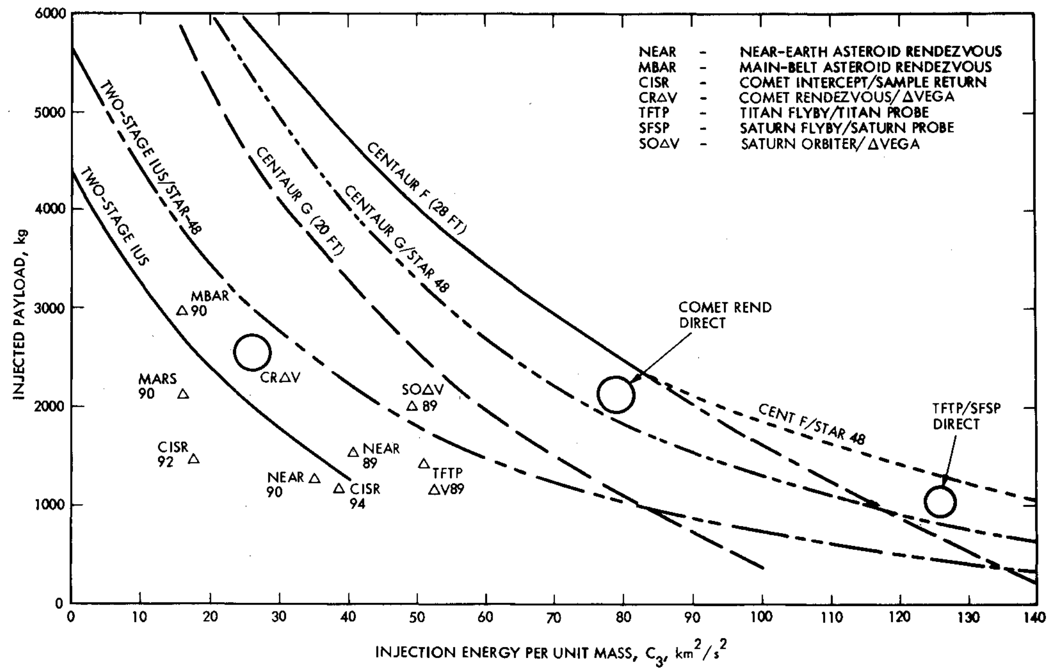

Fig. 1 Launch vehicle performance capability for Shuttle upper stage.

Comet Flyby/Sample Return Missions

In addition to the comets selected for rendezvous missions, two comets, Tuttle and Borrelly, are possible Flyby/Sample Return (F/SR) mission candidates. These are included in Table 1 but are not suitable rendezvous targets for the launch vehicles considered.

Searches for opportunities to flyby a comet and then to return to Earth have been conducted. Table 3 summarizes all available opportunities for all of the comets included in Table 1 (excluding those with perihelion date beyond the year 2000). The opportunity search was restricted to a comet encounter time between $T_P - 20$ days to $T_P + 20$ days. This requirement is imposed to perform sample collection while the comet is active, and to obtain adequate dust samples without having to fly too close to the comet. The trajectory with a positive performance margin for arrival at perihelion was chosen as the reference trajectory. For a 1000-kg spacecraft, the Two-Stage IUS appears to be more than adequate.

For most trajectories a deep space maneuver is not needed. In some cases, the phasing of the Earth and the comet is slightly off and will require a small corrective maneuver. This extra ΔV is included in the calculation of the launch vehicle mass margin shown in Table 3.

Dual Comet Missions (Rendezvous and F/SR)

It has been suggested[2] that the scientific return of an F/SR mission can be enhanced if the gathering of the sample is aided by the presence of a rendezvous spacecraft. Since the rendezvous spacecraft will arrive at the comet earlier than the F/SR spacecraft and will have explored the comet to some extent, it can provide navigation information and guide the F/SR spacecraft more precisely to a desired region of the comet.

Dual-mission opportunities of this type are available for five of the F/SR missions in Table 3 by combining with a rendezvous mission from Table 2.

Near-Earth Asteroids Rendezvous Missions

In addition to the scientific benefits from visiting examples of all the types of minor planets,[3] missions to near-Earth objects can return valuable information for future prospectors.[4] A list of opportunities to rendezvous with the more accessible near-Earth asteroids was compiled. Reference 5 served as a starting point for the search. Recently discovered and still unnumbered objects were included and proved to be some of the best targets in terms of propulsive requirements. Table 4 displays some opportunities with launch dates from

Table 2 Comet rendezvous mission opportunities

Target (apparition)	Launch date	C_3, km^2/s^2	Flight time, yr	Total ΔV, km/s	Launch vehicle mass margin for 600-kg net mass spacecraft, kg(%)
A. Launch vehicle: Space Shuttle/Two-Stage IUS/Star-48					
Kopff (1996)	8/89	26.6	6.71	2.80	664 (22)
C-G (1996)	1/90	25.4	5.90	3.00	501 (16)
Tempel 2 (1999)	9/92	26.6	6.86	2.70	776 (26)
HMP (2001)	12/93	25.6	7.15	2.92	589 (19)
Encke (2000)	2/94	27.4	6.45	2.67	757 (26)
Tempel 2 (2005)	9/97	26.7	7.33	2.78	685 (23)
B. Launch vehicle: Space Shuttle/Centaur					
Temple 2(1994)	7/89	72.7	4.49	2.64	687 (24)
HMP (1995)	11/90	76.0	4.97	2.68	481 (18)
Kopff (1996)	7/91	74.3	4.83	2.30	927 (34)
Wild 2 (1997)	3/92	71.2	4.23	2.57	841 (29)
Tempel (1999)	7/94	77.1	4.97	2.21	861 (33)
HMP (2001)	11/95	78.2	5.25	2.47	579 (23)
Encke (2000)	3/96	85.7	4.34	2.05	579 (26)

Notes: ΔVEGA (2^+) trajectories, Encke (2000) also requires JGA.
Data for one-day launch period.
One-stage Earth-storable propulsion ($I_{sp} = 310$ s).
Total ΔV includes 0.15 km/s for interplanetary navigation and post-rendezvous maneuvers.

Table 3 Comet Flyby/Sample Return mission opportunities (Launch vehicle: Space Shuttle/Two-Stage IUS)

Target (apparition)	Launch date	C_3, km^2/s^2	Flight time, yr	Arrival time, days from perihelion	Flyby V_∞, km/s	Launch mass margin for 1000-kg net mass spacecraft, kg (%)
HMP (1990)	8/89	23.0	3	+20	28	1010 (46)
HMP (1990)	5/90	35.7	3	−15	14	294 (20)
Borrelly (1994)	3/91	17.8	4	0	18	1384 (54)
Tempel 2 (1994)	4/92	17.5	3	0	11	1038 (40)
Tuttle (1994)	10/92	26.6	3	−15	33	627 (32)
TGK (1995)	1/94	28.1	4	0	9	476 (25)
HMP (1995)	3/94	38.3	2	0	12	160 (12)
Wild 2 (1997)	11/96	19.4	3	0	10	956 (39)
Tempel 2 (1999)	3/99	15.5	3	0	11	1512 (55)

Notes: One-day launch period.
One-stage Earth-storable propulsion ($I_{sp} = 310$ s).
Spacecraft net mass of 1000 kg includes 200-kg return capsules, 150 kg for sample collector/dust shield, and 50 kg for interface requirements.

1988 to 1996. Particular interest was expressed by some scientists in the Eros opportunity launching in 1989. Eros is one of the largest asteroids traveling near the Earth that can be reached using the Shuttle/Two-Stage IUS/Star-48.

Main Belt Asteroid Rendezvous Missions

Ballistic missions to rendezvous with main belt asteroids have received little attention in the past. They have been regarded as a class of missions most suited for the application of a Solar Electric Propulsion Stage (SEPS),[6-8] mainly because multiple asteroid rendezvous missions are feasible with the SEPS. Since the SEPS development has been postponed indefinitely, ways to explore asteroids with the currently available launch vehicles have been found. A companion paper[9] describes the analysis of ballistic rendezvous with diverse groups of asteroids distributed in the solar system from 2 to 5 a.u. from the sun. One conclusion from that analysis is that no main belt asteroid rendezvous is possible without a gravity-assist of some sort with current launch vehicles. A large number of asteroids in all regions can be visited with various types of gravity-assists. A single or double Mars gravity-assist (MGA, M^2GA) is effective in reaching asteroids in the inner belt (2-2.5 a.u.) and midbelt (2.5-3 a.u.). For asteroids residing closer to Jupiter's orbit, JGA becomes useful.

Once the spacecraft is in the main belt, flybys of other asteroids are probable. Thus an asteroid rendezvous mission can be expanded to include flybys with modest additional ΔV expenditure. If there is a substantial performance margin for a particular single asteroid rendezvous mission, a multiple asteroid rendezvous opportunity can be found. Multiple asteroid rendezvous missions are energetically more demanding and opportunities are rare; therefore, they are less likely to involve asteroids of significant interest.

Three types of missions are considered: single rendezvous, rendezvous with flybys, and multiple rendezvous. For the single rendezvous mission, examples of short flight-time missions and missions involving asteroids of significant interest are emphasized (Table 5a). Two examples of rendezvous with flybys and multiple rendezvous each are given in Tables 5b, 5c, and 6. The examples shown are for 1990, 1992, and 1994 launch opportunities, and the encounters are mostly

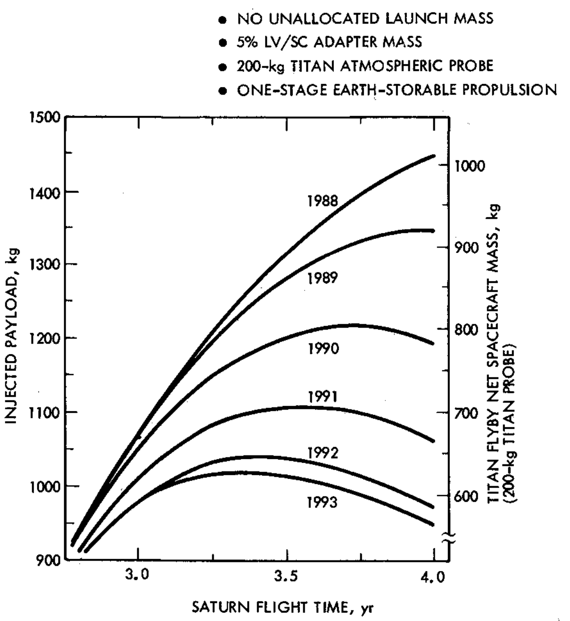

Fig. 2 Injected payload for Saturn direct trajectories, Shuttle/Wide Tank Centaur/Star-48.

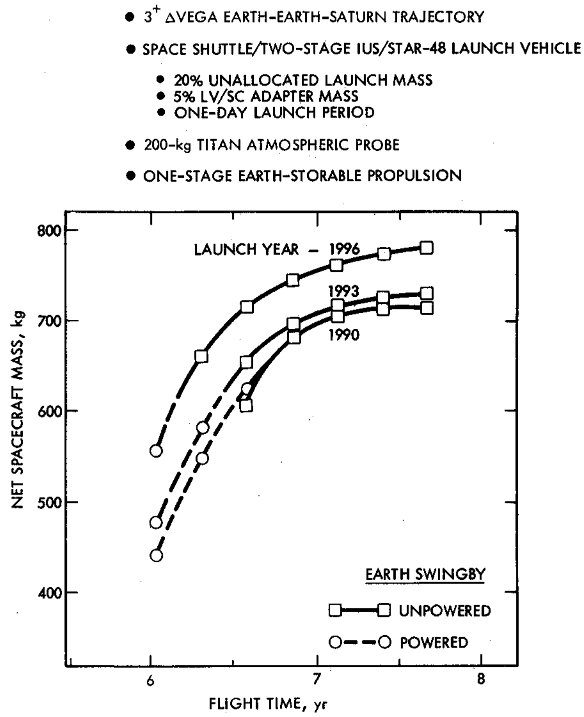

Fig. 3 Titan Flyby/Titan Probe mission delivery capability for ΔVEGA trajectories.

Table 4 Near-Earth asteroid rendezvous mission opportunities (Launch vehicle: Space Shuttle/Two-Stage IUS)

Target	Launch date	C_3, km^2/s^2	Flight time, yr	Total ΔV, km/s	Launch vehicle mass margin for 600-kg net mass spacecraft, kg(%)
Anza	10/88	47.0	2.44	1.36	723 (38)[a]
Eros	1/89	40.3	1.89	1.95	650 (30)[a]
Ivar	7/90	48.7	1.77	1.77	429 (23)[a]
1980AA	1/91	29.2	1.95	1.68	437 (24)
Anteros	5/92	31.8	1.40	1.13	581 (35)
Ivar	8/93	44.4	2.10	1.50	750 (37)[a]
1980 PA	11/94	27.4	1.80	1.86	432 (22)[a]
Anteros	5/95	41.0	2.00	1.58	854 (39)[a]
Eros	1/96	40.8	1.91	1.89	674 (31)

Notes: Data for one-day launch period, except Eros 1989 launch is 10 days.
One-stage Earth-storable propulsion (I_{sp} = 310 s).
Total ΔV includes 0.115 km/s for interplanetary navigation and post-rendezvous maneuvers.
[a] Star-48 Kickstage required.

Table 5A Main-Belt asteroid rendezvous mission opportunities
Single rendezvous examples – Mars gravity assist trajectories

Asteroid	Type	Radius, km	launch date	C_3, km^2/s^2	Flight time, yr	Total ΔV, km/s	Space Shuttle upper-stage, margin for 600-kg net mass spacecraft, kg (%)
Vesta	U	277	9/90	15.4	4.81[a]	3.30	IUS/Star-48, 851 (22)
Fortuna	C	31	8/90	26.0	1.49	2.70	IUS/Star-48, 820 (27)
Hertha	M	40	8/90	19.3	1.84	3.03	IUS/Star-48, 903 (26)
Vesta	U	277	10/92	12.8	2.76	4.06	Centaur F, 2375 (33)
Flora	S	80	10/92	20.1	4.37	2.87	IUS/Star-48 1036 (30)
Massalia	S	70	10/92	24.9	3.90	2.96	IUS/Star-48, 537 (17)
Metis	S	84	11/94	26.6	1.45	3.10	Centaur F, 3123 (54)
Gisela	S	16	10/94	10.1	2.37	3.66	IUS/Star-48, 726 (16)
Euterpe	S	59	11/94	28.8	4.67	2.53	IUS/Star-48, 822 (29)

Notes: One-day launch period.
One-stage Earth-storable propulsion (I_{sp} = 310 s).
Total ΔV includes 0.15 km/s for interplanetary navigation and post-rendezvous maneuvers.
[a] Dual Mars gravity-assist required.

Table 5B Asteroid rendezvous/flybys mission, Example 1: 1992 Medusa rendezvous plus flybys

Event	Type	Radius, km	Date	Flight time, yr	ΔV, km/s	Flyby speed, km/s
Launch			10/6/92	0	(C_3-19.1km^2/s^2)	—
Mars Flyby			4/21/93	0.54	0	4.95
ΔV_1			11/25/94	2.14	0.567	—
313 Chaldaea Flyby	C	54	12/23/95	3.21	0	5.03
149 Medusa Rendezvous	U	13	7/15/96	3.77	1.867	0
Departure from Medusa			2/23/97	4.38	0.161	—
ΔV_2			8/20/98	5.87	0.239	—
21 Lutetia Flyby	M	57	9/12/99	6.93	0.038	3.59
8 Flora Flyby	S	80	9/1/01	8.9	0	2.65
454 Mathesis Flyby	CU	44	7/29/02	9.81	0	3.27

Performance: Total post-launch ΔV requirement (includes navigation ΔV of 0.15 km/s per rendezvous and 0.025 km/s per flyby) 3.122 km/s
Total post-launch ΔV capability: Two-Stage IUS/Star-48 3.643 km/s
Centaur F 4.586 km/s

Table 5C Asteroid rendezvous: flybys mission, Example 2: 1990 Euterpe rendezvous plus flybys

Event	Type	Radius, km	Date	Flight time, yr	ΔV, km/s	Flyby speed, km/s
Launch			9/1/90	0	(C_3-19.7 km^2/s^2)	—
Mars Powered Flyby			2/18/91	0.47	0.186	4.73
ΔV_1			7/3/91	0.84	1.779	—
207 Hedda Flyby	C	31	9/7/91	1.02	0	4.83
ΔV_2			1/23/92	1.40	0.562	—
113 Amalthea Flyby	U	24	4/26/93	2.65	0	4.88
27 Euterpe Rendezvous	S	59	3/31/95	4.58	0.451	0
Departure from Euterpe			10/30/95	5.16	0.025	—
ΔV_3			8/27/97	6.99	0.034	—
136 Austria Flyby	MEU	21	11/3/98	8.17	0	5.66
751 Fiana Flyby	C	56	8/3/99	8.92	0	5.28

Performance: Total post-launch ΔV requirement (includes navigation ΔV of 0.15 km/s per rendezvous and 0.025 km/s per flyby) 3.286 km/s
Total post-launch ΔV capability: Two-Stage IUS/Star-48 3.551 km/s
Centaur F 4.475 km/s

with the inner belt asteroids. A search for opportunities for later launch years and missions involving M^2GA modes (which are required for asteroids beyond the inner belt) constitutes a follow-on analysis.

Saturn Orbiter/Titan Flyby Missions

The Saturn Orbiter/Titan Flyby mission would deliver a single spacecraft into orbit about Saturn for a long duration study of the planet, its satellites, and the magnetosphere. An orbital tour of the Saturn system would be facilitated by repeated close encounters with Titan, which would allow intensive study of this most interesting satellite.[10] Because of the large propulsion requirements to achieve an orbit around Saturn, direct trajectories are not practical. The assumed set of launch vehicle upper stages cannot place the required net spacecraft mass of 600 kg in the desired orbit. For this reason various gravity-assist techniques were investigated, including ΔVEGA trajectories to Saturn (available in any launch year), JGA to Saturn for launches in 1997 and 1998, and ΔVEGA JGA trajectories for launches in 1995 and 1996. The op-

Table 6 Multiple asteroid rendezvous mission example: 1990 launch with Mars gravity-assist trajectory

Asteroid Type	Radius, km	Launch date	C_3, km^2/s^2	Flight time,[a] yr	Stay time,[b] days	Total ΔV,[c] km/s	Space Shuttle Upper Stage[d] margin for 600-kg net mass spacecraft
Anahita S	26	9/90	14.9	4.87	98	2.41	IUS/Star-48
Brunsia	9			2.42		1.16	405 kg (10%)
Totals for mission				7.56		3.57	
Van Gent	7	10/90	20.0	2.55	91	2.57	IUS/Star-48
1950 SH	6			2.37		0.70	466 kg (14%)
Totals for mission				5.17		3.27	
Hertha M	40	8/90	19.3	1.84	703	2.99	Centaur F
1977 NG	4			1.35		0.58	3011 kg (46%)
Totals for mission				5.11		3.57	
Nysa E	34	8/90	17.8	6.14	75	2.38	Centaur F
Massalia S	70			0.84		1.44	2572 kg (39%)
Totals for mission				7.19		3.82	

[a] Flight time to each asteroid and total mission time to second asteroid are shown.
[b] Stay time allowed at first asteroid.
[c] Total ΔV includes 0.30 km/s for interplanetary navigation and post-rendezvous maneuvers.
[d] One-day launch period, one-stage Earth-storable propulsion.

Table 7 Saturn Orbiter flight times (20% launch vehicle margin, 600-kg spacecraft)

Launch opportunity	IUS/Star-48	Centaur G	Centaur G/Star-48	Centaur F
ΔVEGA Launch 1989	–	7.3 yr	6.8 yr	6.6 yr
ΔVEGA Launch 1992	–	7.3 yr	6.7 yr	6.5 yr
ΔVEGA + JS97 Launch 1995	6.8 yr ($20\,R_J$)	6.2 yr ($14\,R_J$)	6.0 yr ($11\,R_J$)	5.9 yr ($10\,R_J$)
ΔVEGA + JS98 Launch 1996	6.3 yr ($51\,R_J$)	5.7 yr ($37\,R_J$)	5.5 yr ($32\,R_J$)	5.4 yr ($30\,R_J$)
Jupiter Assist JS97 Launch 1997	–	–	5.6 yr ($24\,R_J$)	4.6 yr ($19\,R_J$)
Jupiter Assist JS98 Launch 1998	–	–	4.9 yr ($58\,R_J$)	4.1 yr ($50\,R_J$)

Notes: Data for one-day launch period.
Jupiter flyby radius shown in brackets where required.
One-stage Earth-storable propulsion ($I_{sp} = 310$ s).

portunities are identified in Table 7. The table shows the flight time required for delivery of a 600-kg net mass spacecraft to Saturn with sufficient propulsion for navigation, orbit insertion, and the orbital tour. A 20% margin in the launch vehicle injected mass has been maintained. Somewhat shorter flight times would be available with less margin. When a JGA is used, the Jupiter flyby radius is also shown in the table to indicate the possibility of radiation hazard. The Centaur F/Star-48 is not shown in the table because it has no advantage over the Centaur F at the required launch energies.

One important result shown in Table 7 is that a Saturn Orbiter mission can be launched in any year with a Centaur-class upper stage on a ΔVEGA trajectory with an Earth-to-Earth flight time of slightly more than 3 yr (3^+ option), and that this mission is not tied to a JGA opportunity. Two launch years, 1989 and 1992, were chosen as examples of the ΔVEGA performance. The poorest launch year is 1990, with performance improving in subsequent years. The spacecraft can be delivered to Saturn orbit in about 6.5 yr with good design margin using the Centaur F upper stage. The IUS/Star-48 can deliver close to 600 kg, but only if the margin is reduced to zero and the flight time is increased to 8.5 yr.

Shorter flight times are achieved with the ΔVEGA JGA trajectories launched in 1995 and 1996, and any of the upper stages can deliver an Orbiter to Saturn. Flight times as short as 5.4 yr are possible for the 1996 launch opportunity and use of the Centaur F. This is also the only trajectory option for which the IUS/Star-48 can be used with good design margin and for flight times of less than 7 yr.

The direct Earth-to-Jupiter-to-Saturn trajectories launched in 1997 and 1998 have the shortest flight times, but the launch energies are too high for the IUS/Star-48 or the Centaur G upper stages. The Centaur F is the best choice for this mission, with flight times as short as 4.1 yr for the 1998 opportunity. The 1998 launch is also preferred because of the more distant flyby of Jupiter.

Titan Flyby/Titan Probe Missions

The Titan Flyby/Titan Probe (TFTP) mission would deliver a single spacecraft, carrying a Titan atmospheric probe, to the Saturn system. Detailed observations of the planet and Titan would be made by several instruments, and a radar experiment would provide a first look at Titan's surface during a close, 1000-km altitude flyby. Yearly launch opportunites for this mission from 1988 to 1996 have been identified. The complexity and duration of the interplanetary flight path depend on the launch year and the launch vehicle upper stage selected. The upper stages which have been considered are the Centaur F, the Centaur G, and the Two-Stage IUS, all possibly augmented by a Star-48 kickstage.

Direct ballistic trajectories to Saturn using the Centaur-class upper stages with flight times of 3 to 4 yr were con-

Table 8 Centaur delivery capability for direct Earth-to-Saturn trajectories

	Centaur F/Star-48		Centaur G/Star-48	
Launch date	Flight time, yr	Launch vehicle mass margin for 600-kg net mass spacecraft, kg (%)	Flight time, yr	Launch vehicle mass margin for 400-kg net mass spacecraft, kg (%)
2/1988	3.5	349 (27)	4.0	205 (22)
2/1989	3.5	320 (25)	4.0	116 (14)
3/1990	3.5	235 (20)	3.75	21 (3)
3/1991	3.5	143 (13)	3.5	−53
4/1992	3.5	72 (7)	3.5	−104
4/1993	3.5	50 (5)	3.5	−123
5/1994	3.5	78(7)	3.5	−98
5/1995	3.5	145 (13)	3.5	−52
5/1996	3.5	215 (18)	3.5	−5

Notes: Ten-day launch period.
Launch declination penalty included.
200-kg Titan atmospheric probe delivered.
Total $\Delta V = 0.15$ km/s for navigation and deflection.
One-stage Earth-storable propulsion ($I_{sp} = 310$ s).

Table 9 Mars Orbiter mission opportunities (Launch vehicle: Space Shuttle/Two-Stage IUS)

Launch date	C_3, km^2/s^2	Flight time, yr	Total ΔV, km/s	Launch vehicle mass margin for 600-kg net mass spacecraft, kg(%)
7/88	11.9	0.57	2.45	1142 (37)
8/90	15.9	0.98	2.52	716 (26)
9/92	12.3	0.94	2.38	1173 (38)
10/94	9.7	0.86	2.42	1402 (42)
11/96	9.2	0.83	2.61	1267 (38)
12/98	10.5	0.80	2.86	864 (27)

Notes: Ten-day launch period.
One-stage Earth-storable propulsion ($I_{sp} = 310$ s).
Total ΔV includes 0.4 km/s for interplanetary navigation, orbit insertion gravity losses, and orbital maneuvers.

sidered. A Star-48 kickstage is required for even these stages to deliver a 600-kg net spacecraft and a 200-kg Titan probe. The early 1990s are the least favorable time for a direct launch to Saturn because of high launch declinations and the peak launch energy requirements of 1994 and 1995. Figure 2 shows the injected mass and net spacecraft mass delivery capabilities for the Centaur F/Star-48 vs flight time for each launch year. Because of launch declination penalties, 1993 is the worst performance launch year. Table 8 shows the launch vehicle margin available for the two Centaur options for each launch year from 1988 to 1996. For a 600-kg net spacecraft mass and a 3.5-yr flight time, the Centaur F/Star-48 has good margin (greater than 10% of the launch vehicle injected mass capability) for direct transfers in every launch year except 1992 to 1994. The Centaur G/Star-48 has significantly less performance and cannot deliver the required payload. If the spacecraft were simplified to a basic probe carrier, with the removal of most of the nonprobe science, a net mass of about 400 kg could be achieved. This spacecraft could be delivered by the Centaur G/Star-48 only for launches up through 1990 and again after 1996.

Since the Two-State IUS/Star-48 cannot deliver the required spacecraft mass to Saturn on a direct trajectory, ΔVEGA trajectories were investigated. With a flight time on the Earth-to-Earth leg of slightly more than 3 yr (3^+ option), these trajectories can deliver the required mass to Saturn with ample launch vehicle margin. The cost is an increase in flight time of about 3 yr. Figure 3 shows the net spacecraft mass which can be delivered vs flight time for launches in 1990, 1993, and 1996, with a launch vehicle margin of 20%. The 1990 launch year has the lowest performance (corresponding to the 1993 direct trajectories), but the 600-kg net mass spacecraft can be delivered with good margin in 7.6 yr. Shorter flight times are available in subsequent years.

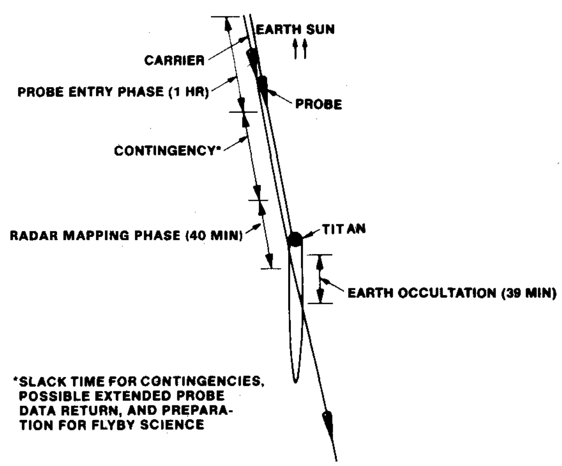

Fig. 4 Titan Flyby/Titan Probe mission encounter design.

Mars Orbiter Missions

The Mars Orbiter mission requires that a single spacecraft be launched to Mars and delivered into a low-altitude, circular, near-polar orbit for geochemical mapping of the entire planet. Opportunities for direct transfers to Mars occur about every 26 months, with launch energy requirements of less than 20 km^2/s^2. The propulsion requirements for reaching a low-altitude orbit at Mars are significant, requiring that the mass injected from Earth be three or more times the net spacecraft mass. For this large payload, a Two-Stage IUS is assumed as the upper stage for a launch from the Space Shuttle. The launch opportunities which were considered repeat about every 2 yr from 1988 to 1998 and are compared in Table 9. The launch and arrival dates in the table are selected to give the greatest net mass delivered into Mars orbit for a 10-day launch period. In all cases except 1988, a Type 2 interplanetary trajectory is selected to maximize net mass. A 600-kg net spacecraft mass can be orbited with any of these trajectories with at least 20% margin in the launch vehicle injected mass. This provides a valuable design margin for a spacecraft which must be compatible with several other missions.

IV. Encounter Design

For each of the mission types considered, strawman mission descriptions have been prepared for a representative launch opportunity. These provide a basic description of the timeline of each mission and its requirements on different systems such as launch vehicle, propulsion, communications, and navigation. When all of these requirements are compared, preliminary design decisions can be made on the capabilities required for different subsystems to carry out a sequence of these missions. In some cases, a particular mission may be a poor candidate for a project using a multimission spacecraft because its requirements would be so different or more demanding than those for the others. For example, the radiation hardening required for a Saturn Orbiter mission with a close flyby of Jupiter for a gravity-assist could increase the cost of electronic parts for all of the other missions.

An essential part of this process is a preliminary encounter design which identifies the capabilities and constraints for observing the targets of each mission. Many of the requirements for the data handling and radio subsystems are determined by the timeline and observation strategy for the encounter period. This section describes the work that has been done on encounter design for two of the mission types, TFTP and Mars Orbiter.

Titan Flyby/Titan Probe Mission

The TFTP encounter sequence was designed to maximize the science return from several instruments during a fast flyby of Saturn's largest satellite. It would return essential data on the surface and lower atmosphere of Titan, while using proven designs for the probe from the Galileo mission. The different science investigations place many, often conflicting, requirements on the encounter sequence. For example, all of the following may be expected:

1) Deploy an atmospheric probe to a specific target in Titan's atmosphere and maintain a 1-h radio link between the probe and the carrier spacecraft to record the descent data to the surface.

2) Make a close flyby of Titan at 1000-km altitude to allow mapping of its surface by a radar instrument on the carrier spacecraft.

3) Fly behind Titan as seen from the Earth to conduct a radio occultation experiment through the satellite's atmosphere.

4) After the Titan flyby, approach Saturn to a distance of 4 radii to allow studies of the planet, the rings, the inner satellites, and the magnetosphere.

This last requirement for a close Saturn flyby determines the inbound location at which Titan must be intercepted and requires that the spacecraft cut across Titan's orbit at a high relative speed of about 12 km/s. This intercept speed could be significantly reduced to about 7 km/s if a more tangential approach was made with a greater minimum approach distance to Saturn (20.3 radii at Titan's orbit). The high approach speed also reduces the time available for close observations of the satellite.

There would be major conflicts in pointing and data handling between the radar mapping experiment and the probe relay sequence, if both of these were conducted near closest approach to Titan. Since the radar must operate near closest approach (for ±20 min) to obtain the best resolution, it was decided to move the probe entry and relay sequence back before a closest approach as shown in Fig. 4. The probe is released 20 days before encounter and enters 90 min before the start of the radar observations. Although the probe to spacecraft range increases during the radio link, the variation of the antenna aspect angles between them is greatly reduced since the spacecraft is traveling almost straight toward the probe. The antenna design can be simplified while still keeping the range less than half that requried for the Galileo probe link at Jupiter. This strategy also requires that the probe enters at a steep flight-path angle near 90 deg so that its axial antenna remains pointed toward the carrier spacecraft. The navigation problem of targeting the probe is lessened since there is a greater margin for error when it is aimed for

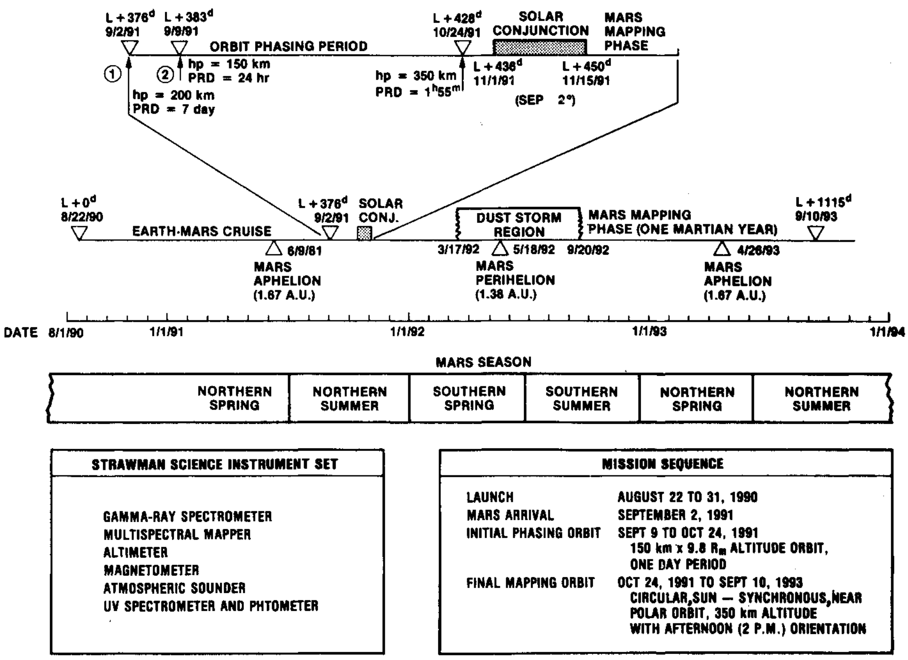

Fig. 5 Example of a Mars Orbiter mission timeline.

the center of Titan. The entry loading is also much less than for the Galileo probe entry at Jupiter.

Thus, separating the probe entry sequence from the close flyby for radar mapping provides a number of advantages. A 30-min timespan between each sequence has been planned for operations contingencies. Using anticipated data rates for a full complement of probe and spacecraft instruments including imaging, infrared and radar observations, the requirements for storing all the Tian encounter data have been determined to be less than half the Galileo tape recorder capacity. The occultation of Titan can be obtained with the proper orientation of the flyby trajectory, and it will begin shortly after closest approach, while the outbound radar data is still being recorded. The Saturn encounter will follow about one day after Titan, allowing time for playback of the radar and probe data.

Mars Orbiter Mission

The encounter sequence for the Mars Orbiter mission was designed to expedite the delivery of the spacecraft to a low-altitude orbit from which geochemical mapping of the surface would be conducted. The principal difficulty in achieving the desired final mapping orbit is obtaining the preferred solar orientation close to noon for the dayside passes. The encounter strategy selected attempts to minimize the delay in reaching the final orbit, while considering performance constraints on the spacecraft and additional science goals.

The desired final orbit for geochemical mapping would be circular at low altitude (300-500 km) and sun synchronous with a solar orientation between noon and 2 p.m. for the dayside passes. The sun-synchronous requirement gives an orbit inclination of about 93 deg. All of the orbit parameters can be achieved with a direct, in-plane, orbit insertion burn except the solar orientation, which is dependent on the orientation of the approach asymptote. For a low-energy approach to Mars, which allows greater mass delivery into orbit, the approach asymptotic velocity relative to Mars tends to point toward the dusk terminator. For an in-plane orbit insertion, the resulting sun-synchronous orbit would lie close to the terminator, rather than at the near-noon orientation. The desired orientation can be reached only in a direct way by an out-of-plane burn at the orbit insertion or a subsequent orbit plane change, either of which is enormously expensive in propellant. An alternative solution is to place the spacecraft in an intermediate elliptical orbit and simply wait for the motion of Mars about the sun to change the phase angle of the orbit. While less propellant is used, the total time to achieve the final mapping orbit is increased. This can be many weeks depending on the particular arrival date. This strategy[11] works because a long elliptical orbit is affected much less by the planet's oblateness. For a one-day orbit at the required inclination, the orbit precesses only 0.014 deg/day as compared to the sun-synchronous rate of 0.524 deg/day.

The use of an intermediate elliptical orbit also fits well with the required orbit insertion strategy. One design for a multimission spacecraft uses a propulsion system with a relatively low thrust of 100 lb. This choice was made after a survey of currently available bipropellant rocket motors.[12] This low thrust means that if the Mars orbit insertion maneuver of over 2 km/s was attempted in one long burn, the gravity losses would be prohibitively large. To reduce these, it was decided to break the orbit insertion into three burns. The first burn at a periapsis altitude of 200 km would brake the spacecraft into a seven-day elliptical orbit to minimize the largest part of the orbit insertion as much as possible. At the next periapse the orbit period would be lowered to one day with a smaller burn, and the spacecraft would wait in this orbit until the proper solar phase angle is reached. The orbit then would be circularized at the desired altitude.

Thus, the use of an intermediate elliptical orbit fits both the need to reduce the gravity losses at orbit insertion and the requirement to wait for the proper solar orientation. An additional science goal also can be addressed from an elliptical orbit, where the periapsis altitude can easily be lowered to the fringes of the atmosphere (about 150 km altitude). A magnetometer on the spacecraft would attempt to make the first measurements of the Martian magnetic field and study its interactions with the solar wind, both during the elliptical orbit phase at varying altitudes and from the mapping orbit.

This encounter strategy was applied to the 1990 launch opportunity to demonstrate its implementation. Figure 5 shows a timeline for the mission, where the spacecraft reaches Mars in September 1991 after a flight time of 1 yr. The arrival date was adjusted to improve the initial solar phase angle, while keeping the approach speed less than 3 km/s to reduce the gravity losses. The launch and arrival dates also maintain a comfortable margin of 20% in the launch vehicle injected mass for delivery of a 600-kg net spacecraft mass. After initial insertion into a seven-day elliptical orbit, the period is reduced to one day, and the spacecraft waits in this orbit for 45 days until the solar phase angle is reduced to 30 deg or a 2 p.m. orientation. The orbit is then circularized to a mapping altitude of 350 km, which provides a repeating ground track every four Martian days. The mapping phase of the orbit then continues for one Martian year to study seasonal variations.

V. Conclusion

The Mariner Mark II and Planetary Observer projects have been the beneficiaries of these mission designs. Mariner Mark II missions will be primarily to the outer solar system with targets including a comet, main belt asteroids, and Saturn and Titan. A multimission spacecraft is being designed with many subsystems remaining unchanged for the entire set of three or five missions. The Planetary Observer missions will use modified, Earth-orbiting spacecraft to further explore Mars, the moon, near-Earth asteroids, and other targets in the inner solar system. The Mars Orbiter and near-Earth asteroid rendezvous missions presented here are being considered by this project.

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