[3] Hiester, N. K., and Clark, C. F., "Comparative Evaluation of Ablating Materials in Arc Plasma Jets," TR 111, April 1, 1965–March 31, 1968, Contract NASr-49(15), NASA CR-1207, 1968, Stanford Research Inst.

[4] Hiester, N. K., Clark, C. F., and Vojvodich, N. S., "Ablative Characterization of Seven Heat Shield Materials—A Review of the NASA-Stanford Research Institute Round-Robin Program," *AIAA/ASME 10th Structures, Structural Dynamics and Materials Conference*, AIAA, New York, 1969, pp. 57–68.

[5] Buckingham, E., "On Physically Similar Systems: Illustrations of the Use of Dimensional Analysis," *Physical Review*, Vol. 4, p. 345, 1914.

[6] Rayleigh, L., "The Principle of Similitude," *Nature*, Vol. 95, 1915, p. 66.

[7] Rosensweig, R. E. and Beecher, N., "Theory for the Ablation of Fiberglass-Reinforced Phenolic Resin," *AIAA Journal*, Vol. 1, Jan. 1963, pp. 1802–1809.

[8] Chapman, A. J., "An Experimental Investigation of Several Ablation Materials in an Electric Arc-Heated Air Jet," TND 1520, April 1963, NASA.

[9] "Postlaunch Report for Mission AS 202 (Apollo Spacecraft 009)," MSC-A-R-66-4, May 6, 1966, NASA; also "Postlaunch Report for Mission AS 202 (Apollo Spacecraft 011)," MSC-A-R-66-5, Oct. 12, 1966, NASA.

Solar Electric Missions To Ceres

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THE continuing interest shown in missions to the asteroid belt can be justified by considering the uniqueness of this region as a source of valuable information regarding the origin of the solar system and as a possible hazard for future probes enroute to the outer solar system. For studying the latter problem an asteroid belt flythrough mission is sufficient to initiate a program of accurately determining relevant properties of the region between roughly 2 and 4 a.u. However, for studying the asteroid belt itself, it appears obvious that no such effort can be considered worthwhile unless it includes at least a close flyby of one of the asteroids as a primary mission objective.

In this Note, flyby and rendezvous missions to the largest asteroid—Ceres—are examined for a late 1976 launch opportunity. The purpose of the Note is to demonstrate the application of combined chemical and solar electric propulsion (SEP) systems to a specific mission of particular scientific value and to establish approximate weight breakdowns for payload-optimized spacecraft.

The scientific return from a close approach to, or soft landing on, Ceres would be considerable, for size is the only physical property of the major asteroids which is known with reasonable certainty. A target such as Ceres, which is located in the asteroid belt, may deserve priority over easier targets such as Eros which pass closer to Earth. In this way, the information gained from what will, in any event, be a major mission effort can be maximized since an asteroid belt survey can be performed simultaneously.

Approximate orbital elements and some physical properties of Ceres are shown in Table 1. The orbital parameters are subject to large (compared to the major planets) periodic fluctuations due primarily to Jupiter, so it is desirable to use osculating (time variable) elements even for preliminary studies. For this purpose, an ephemeris tape was generated from data computed by Michielsen and Krop[1] similar to the tabular data for Ceres and Vesta found in the NASA Planetary Flight Handbook.[2]

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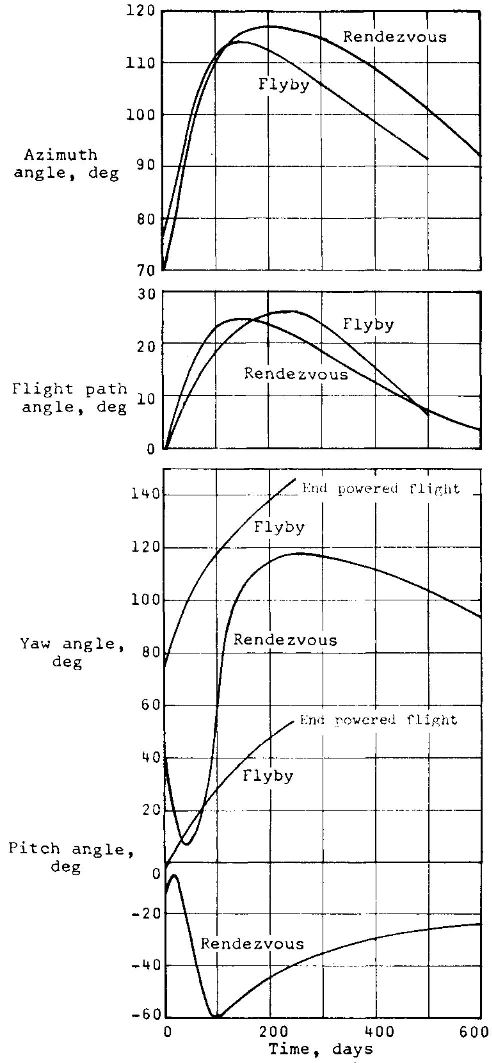

Fig. 1 Spacecraft orientation (azimuth and flight-path angles) and thrust vector orientation (yaw and pitch angles) for flyby and rendezvous missions to Ceres.

Table 1 Orbital and physical characteristics of Ceres

Inclination to ecliptic (deg)	10.607
Longitude of ascending node (deg)	80.514
Longitude of perihelion (deg)	152.367
Mean motion (deg/day)	0.2141
Mean circular velocity (EMOS)	0.6011
Eccentricity	0.0759
Sidereal period (yr)	4.60
Semimajor axis (a.u.)	2.7675
Perihelion (a.u.)	2.5574
Aphelion (a.u.)	2.9776
Mean anomaly at epoch (deg)[a]	279.880
Diameter (statute miles)	480.0
Mass (fraction of Earth)[b]	2.2×10^{-4}
Escape velocity (m/sec)[c]	680.0
Albedo	0.06
Rotational period (hr)	9.08

[a] Epoch June 11, 1957.
[b] Assuming asteroid density equal to Earth's density. Based on Lunar density, the mass would be multiplied by approximately 0.6.
[c] Based on mass as given.

The SEP system assumed for this asteroid mission is characterized by a specific impulse of less than or equal to about 4000 sec, a specific mass, α, of the entire propulsion system of 30 kg/kw, and a thruster efficiency of less than or equal to about 72%. These can be compared with operating parameters for actual thrusters in Ref. 3. Fuel tanks are assumed to weigh 10% of the fuel (3% for actual hardware and 7% for contingency). Multiple ion engines are required, consuming up to 5 kw of electrical power apiece, and the thruster array must provide continuous thrust for up to 600 days. As the available solar power drops, some units are shut off, and these serve as standbys for the units which are still operating. The power level for the Ceres missions has been restricted to no more than 50 kw; there seems to be little doubt that such a solar array is feasible if required.[4] The thrust vector is free to vary in an optimum way. The Princeton University program TOPCAT[5] has been used to generate the trajectories for this study. The launch date is Nov. 18, 1976, and trip times are 500 and 600 days for the flyby and rendezvous missions, respectively. For the flyby mode an Atlas (SLV3C)/Centaur/SEP launch vehicle was assumed, with the total energy required for the mission being divided optimally between the chemical and SEP systems. For the rendezvous mode a Titan IIID(1207)/SEP with and without Centaur was assumed. In this case the chemical system provides virtually all of the energy required to reach the target, while the SEP system performs the required plane change and retrobraking maneuver to match velocity vectors with the target. The contribution of the SEP system in each flight mode is clarified by Fig. 1, which shows azimuth and flight-path angles for the spacecraft and pitch and yaw angles for the thrust vector, referenced to a heliocentric Earth-equatorial coordinate system.

Numerical results for flyby and rendezvous trajectories to Ceres are summarized in Table 2. For the flyby mode the payload, as defined later in Table 3, is 1087 lb. The optimum high thrust contribution is an injection energy C_3 of 7.84 km^2/sec^2; an initial solar power of 7.05 kw is optimum for the SEP power supply. For the rendezvous mission Table 2 shows that the use of a Centaur on the Titan IIID(1207) approximately doubles the payload from 802–1621 lb. The C_3 values are 30.5 and 49 km^2/sec^2 for the small and large payloads, respectively. The injection energy for the large payload is higher than the optimum value to force the optimum power supply below 50 kw. The power supply can be reduced further with little loss in payload—an additional 50% decrease to 25 kw results in a payload loss on the order of 10%.[6] For any of the missions shown, reduction in the SEP power supply specific mass from 30 kg/kw–20 kg/kw results in a payload increase of about 17%.

Weight breakdowns for the Ceres flyby and large payload rendezvous spacecraft are shown in Table 3. The flyby spacecraft has 387 lb allotted for experiments and sensors while the rendezvous spacecraft has 221 lb of experiments and sensors plus a 500-lb lander capsule which could separate from the main spacecraft and soft-land with a small chemical rocket.

In conclusion, it has been shown how SEP systems can be used to good advantage for missions to Ceres. For the Ceres flyby mode, the SEP stage approximately replaces available high-energy stages, and hence does not demonstrate a compelling case for an advanced propulsion system. However, since the Ceres rendezvous mission requires Saturn V capability for an approximately equivalent payload,[7] SEP in this case turns a nearly impractical mission into one which is quite practical from the standpoint of hardware availability.

Table 2 Trajectory and spacecraft data for optimized Ceres flyby and rendezvous missions

	Flyby (Atlas-(SLV3C)/Centaur/SEP)	Rendezvous (TitanIIID-(1207)/Centaur/SEP)	Rendezvous (TitanIIID-(1207)/SEP)
Mission data			
Launch date	Nov. 18, 1976	Nov. 18, 1976	Nov. 18, 1976
Trip time (days)	500.0	600.0	600.0
Thrust time/trip time	0.491	1.0	1.0
Final radius (a.u.)	2.84	2.90	2.90
Transfer angle (deg)	214.0	236.0	236.0
Flyby velocity (km/sec)	4.89	0.0	0.0
Spacecraft weight (lb)			
Initial injected weight	2094.0	7700.0	3580.0
Payload	1087.0	1621.0	802.0
Propulsion system			
Initial solar power (kw)	7.05	49.0[a]	20.0
Injection C_3 (km^2/sec^2)	7.84	49.0[b]	30.48
Ion engine I_{sp} (sec)	3126.0	4082.0[b]	3444.0
Engine efficiency (%)	66.0	72.0	69.0
Specific mass (kg/kw)	30.0	30.0	30.0
F/M_o (m/sec^2)	0.000321	0.000510	0.000496

[a] Smaller than optimum to reduce initial solar power to less than 50 kw.
[b] Larger than optimum to reduce initial solar power to less than 50 kw.

Table 3 Spacecraft weight breakdowns for Ceres missions

Subsystem	Flyby mode Weight, lb	Rendezvous mode Weight, lb
a) Spacecraft		
Power and propulsion	467	3252
Fuel	300	1870
Tank	30	187
Structure	210	770
b) Payload		
Shielding	100	200
Guidance and navigation	200	200
Stability and control	150	200
Communications	200	200
Communications (relay)	...	20
Data management	50	80
Experiments/sensors	387	221
c) Lander capsule		
Structure	...	125
Propulsion[a]	...	75
Power (RTG)	...	25
Communications	...	25
Experiments/sensors	...	250
Total	2094	7700

[a] Chemical retrorocket, vernier control.

References

[1] Michielsen, H. F., and Krop, M. A., "Ephemerides for the Four Major Asteroids for the Period 1969-2000," LMSC/D004914, July 1969, Lockheed Missiles and Space Co., Sunnyvale, Calif.

[2] Ross, S., ed., "Planetary Flight Handbook, Part 5—Trajectories to Jupiter, Ceres, and Vesta," *Space Flight Handbook*, Vol. III, SP-35, 1966, NASA.

[3] Pawlik, E. V. and Macie, T., "Electric Propulsion System Performance Evaluation," AIAA Paper 69-236, Williamsburg, Va., 1969.

[4] Barber, T. A., Goldsmith, J. V., and Edberg, J. R., "Spacecraft Electric Propulsion—Now?", *Aeronautics and Astronautics*, Vol. 6, No. 6, June 1968, pp. 38–47.

[5] Lion, P. M., Campbell, J. H., and Shulzycki, A. B., "TOPCAT 1: Trajectory Optimization Program for Comparing Advanced Technologies," Aerospace Engineering Rept. 717s, March 1968, Princeton Univ., Princeton, N.J.

[6] Brooks, D. R., "Solar-Electric Missions to Specific Targets in the Asteroid Belt," AIAA Paper 70-1120, Stanford, Calif., 1970.

[7] Roberts, D. L. and Narin, F., "Early Missions to the Asteroids," *Advances in Space Sciences and Technology*, edited by F. I. Ordway, III, Vol. 9, Acacemic Press, N.Y., 1967, pp. 123–160.