New Concepts for Mercury Orbiter Missions

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The next logical step in the exploration of Mercury is an orbiter mission. A conflict exists between those in the field of planetary sciences who desire a mission with a low circular orbit, and scientists in the fields and particles disciplines, who generally prefer a highly elliptical spacecraft orbit. The thermal environment imposed by the Sun and planet render the low orbit intolerable for spacecraft using previous thermal control methods. A thermal control concept and a spacecraft mission concept have been developed which resolve these problems and promise a scientifically significant mission for the mid-1980s.

Introduction

TRIGGERED initially by the results of the Mariner 10 Mercury flyby mission, there has been a continuing high level of scientific interest in Mercury orbiter missions. Mercury occupies a unique place in solar system physics because of its closeness to the Sun, its high density, and magnetic field. A Mercury orbiter can greatly advance our understanding of the dynamics of magnetospheres by probing a magnetosphere with different parameters than those of Earth. A Mercury orbiter offers an excellent opportunity for comparative analysis of the chemical composition and crustal evolution of the planet with that of the Moon. In fact, as shown in this paper, a typical science complement for the Mercury orbiter is much the same as that proposed for the Lunar Polar Orbiter.[1] Also the planet Mercury, explored by an orbiter, can serve as an excellent natural test bed for solar physics experiments and relativistic theories of gravitation. The search for neutrons from the Sun constitutes one of the major untouched problems in solar astrophysics. A unique opportunity for testing the hypothesis that observed neutron fluxes are of solar origin is presented by using Mercury as a "shutter" to cut off neutron emission each time the spacecraft passes the planet's dark side. With proper metric tracking measurements, three important tests of general relativity could be substantially improved, and one important parameter of the Sun's gravity field, the solar quadrupole moment, could be determined with higher accuracy.

Previous studies have shown, however, that Mercury orbiter mission requirements for launch vehicle and orbit insertion performance, as well as for spacecraft survival in the Mercury environment, are very demanding. The thermal control problem at Mercury is difficult at best, and early studies[2] demonstrated that the usual thermal control techniques and spacecraft configurations were inadequate. Subsequent studies have indicated that a highly elliptical Mercury orbit might be feasible without development of new thermal control techniques.[3] Such an orbit is not of interest to much of the science community because it is difficult or impossible to do good surface science. This segment of the community desires a low circular orbit. However, proposed thermal control techniques suitable for highly elliptical orbits were found to be totally inadequate for this low orbit. Previous studies[3] recognized the problem without trying to solve it, merely allocating generous amounts of mass for thermal control. Because solution of the low-orbit thermal problem is so critical to realizing high-quality science return from a Mercury orbiter mission, it was decided to concentrate the resources of this study toward developing a solution.

On the other side of the science coin is a group of scientists who would find their experiments degraded by the lower orbit. This group includes those interested in fields and particles, solar physics, etc. As a result of this dichotomy of interest, the concept of a dual spacecraft mission which would satisfy both sets of requirements was developed. A mission of this type offers certain operational advantages as well.

There is considerable interest in Mercury landers for purposes such as determining libration, surface sampling, etc. No effort was directed toward landers in this study because both surface landers and penetrators for Mercury have been studied previously.[4-6] Including landers (or penetrators) on a mission such as that described herein presents no problems except for the mass required for delivery and for some integration concerns.

Spacecraft Thermal Control

A fresh approach to the problem of spacecraft thermal survival in low Mercury orbit has led to a promising solution. A circular orbit of 105-km altitude was chosen for analysis because the 1.5-h period is in resonance with Earth's rotational period and because it seemed to be as low as is likely to be practical for an early orbital mission. Since previous studies demonstrated the inapplicability of standard spacecraft thermal control techniques and of previously developed spacecraft configurations, it was decided to place no constraints upon the spacecraft in these areas. From the thermal viewpoint, at an altitude of 105 km, the planet presents as much of a problem as the Sun (if not more). Thus, the thermal protection problem becomes a question of difficult geometry as well as of the sheer magnitude of thermal input. The scientific desire for a nadir-pointing spacecraft is helpful in one sense in that it keeps the planet on one side of the spacecraft. On the debit side, this orbit causes every side of the spacecraft to see the Sun sometime during the mission unless some action is taken. This situation is unacceptable because both the Sun and the illuminated side of the planet expose the spacecraft to radiant energy fluxes of such magnitude that they prevent operation of the electronics at acceptable temperatures. Consider, for example, that the solar irradiance at Mercury perihelion exceeds 10.5 mean-Earth-Suns (Fig. 1), and that the planet effective blackbody surface temperature at the subsolar point reaches nearly 700 K (Fig. 2). In this environment, the thermal control system is called upon to maintain temperatures at a tolerable level for electronics components. Since these temperatures generally

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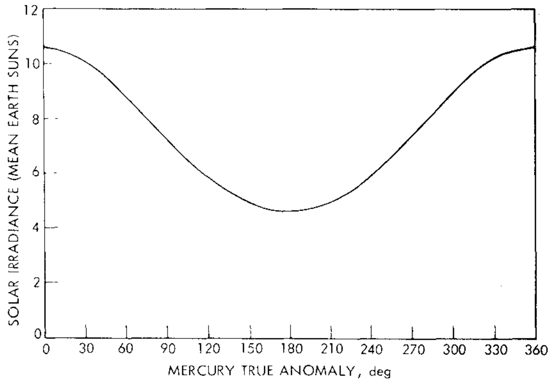

Fig. 1 Solar irradiance.

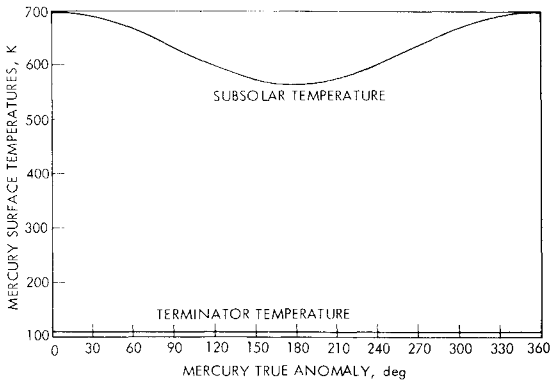

Fig. 2 Mercury surface temperatures.

must be significantly less than 100°C in the interior of the electronics, it is necessary that the heat-rejecting surfaces of these packages function in the 20 to 30°C range at normal heat loads. Some science instruments present even more serious problems in that detectors must be maintained at quite low temperatures (of the order of 120 K), at least during data taking.

The thermal dilemma is solved by rotating the spacecraft about the nadir-zenith axis, as necessary, so that two opposing sides are always parallel to the average solar rays. These faces are dedicated heat radiators which remain parallel to the local vertical at all times. However, the low orbit results in a relatively good view of the planet. For the chosen orbit, the limb is only 16 deg below local horizontal. When over the sunlit side of the planet, unprotected radiators would be swamped by planetary thermal emission. The incident planetary heat flux would, at times, exceed 2970 W/m^2. In the proposed design, this flux is blocked by a planet shade, which consists of a single-layer metal foil. Even though the shade temperature will equal or exceed the planet temperature most of the time, by careful control of surface characteristics the amount of energy re-emitted or reflected to the radiators can be limited to the point that radiators can be designed to operate at no less than 80% of deep-space efficiency, even at high Sun. The thermal inertia provided by radiators of modest mass per unit area allows the spacecraft electronics compartment to function at acceptable temperatures with only minor orbital temperature variations. Many science instruments which must look at the surface will absorb significant radiated energy from the planet. By providing adequate conductive coupling to the radiators, this heat load can be dissipated.

In spite of the obvious benefit it provides, the planet shade presents some problems. First, it must be large. At 105 km, it must extend nearly 3.5 cm for every centimeter of radiator height to fully obscure the planet. Furthermore, the shade becomes quite hot at certain points of the orbit. Since the radiators "see" the shade in lieu of the planet, it becomes an

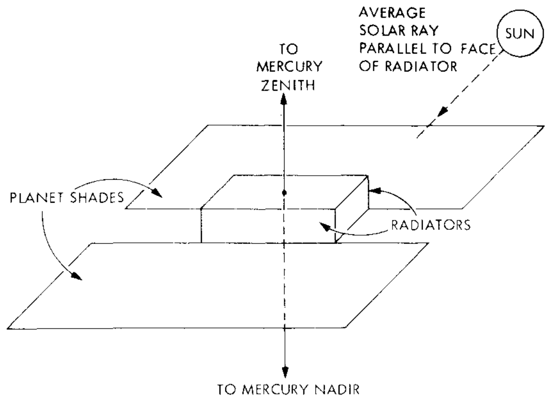

Fig. 3 Thermal control schematic.

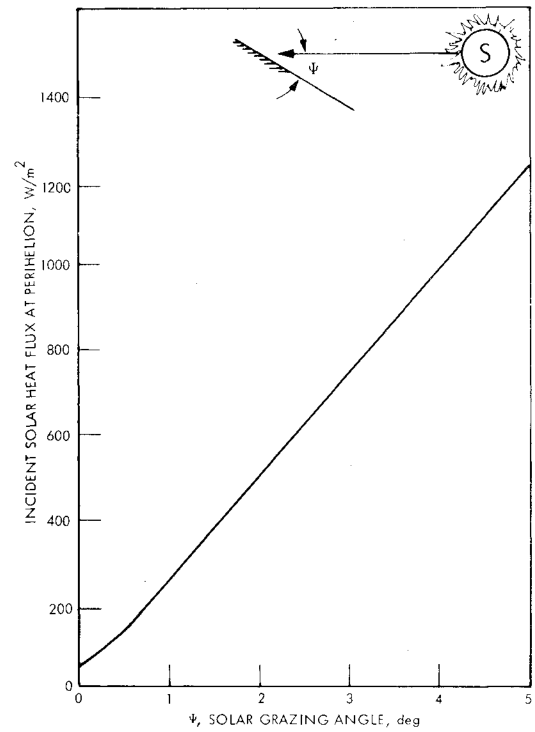

Fig. 4 Solar heat flux vs grazing angle.

IR heat source to them. Fortunately, by using a low emissivity coating on the zenith-facing side of the shade, the effect of this heat source can be reduced to an acceptable level. A schematic of the bus and planet-shade configuration is shown in Fig. 3.

By far the greatest potential heat source to the radiators is the Sun. However, by introducing continual yaw control (supplied, for example, by momentum wheels), it is possible to align the radiator faces so that they are parallel to the average solar rays.§ Of course, it is impossible to maintain perfect yaw control. Indeed, even perfect yaw control would not prevent solar illumination of the radiators because the half-angle of the solar divergence is nearly 0.9 deg at Mercury perihelion. Figure 4 shows the heating potential of incident solar-heat flux as a function of solar-grazing angle. This angle may be interpreted as angular yaw positioning error. Even with the lowest solar absorptivity radiator coating, it is best to avoid even low-angle solar grazing by providing, for example, a small Sun shade around the perimeter of each radiator. In this analysis, it is assumed that neither direct nor reflected solar energy strikes the two radiators.

§The nominal solar rays are parallel to the line passing through the Sun and Mercury mass centers.

Selection of Planet Shade Surface Finishes

Minimum heat input to the radiators was the primary criterion for the selection of the planet-shade surface finishes. As previously mentioned, a low emissivity is desired on the zenith-facing surface to limit its emission, a portion of which is absorbed by the radiators. On either side of the shade, high solar reflectivity (i.e., low solar absorptivity) is one requirement. The other is that solar reflections for the zenith-facing surface must be specular. If solar energy were reflected diffusely, a portion of it would strike the radiators. Materials which have the characteristics just described for the zenith-facing side are polished metals. The polished metal with both the lowest emissivity and highest solar reflectivity is silver which, unfortunately, has a tendency to tarnish during ground handling. Comparison of analytical predictions of net radiator capacity as a function of planet-shade metal finish showed that gold, which has superior resistance to atmospheric degradation, is an acceptable second choice to silver.

In addition, the nadir-facing surface should have a high emissivity to reradiate absorbed energy efficiently. Materials which have this characteristic are metallized Teflon or "white" coatings. However, peak shade-temperature predictions rule out the use of Teflons. At first glance, it would appear necessary to rely on high-temperature "white" coatings. However, solar exposure to this side, except for low-level albedo, occurs only for orbit angles between pole passage and solar occultation. Furthermore, solar-grazing angles are quite small. Because the requirement for low solar absorptivity on this side is not nearly as firm as for the zenith-facing side, even "black" coatings are acceptable.

The numerical results of this study are based on an assumed planet shade with the properties of polished gold on the zenith-facing side ($\epsilon = 0.04$, $\alpha = 0.25$) and a typical "black" coating ($\epsilon = 0.8$, $\alpha = 0.9$) on the nadir-facing side.

Radiator Coating

The primary property required for the radiator is high emissivity. Even though, ideally, sunlight never strikes the radiators, it would be prudent to use a coating with low solar absorptivity. Momentary loss of spacecraft orientation relative to the Sun may cause overheating even with such a coating, but with a "black" radiator coating, chances for survival would be considerably reduced. Silvered Teflon, about 10-mils thick ($\epsilon = 0.77$, $\alpha = 0.09$), is the assumed radiator coating.

Perhaps the major output of this study is the design of a passive thermal control system capable of maintaining a large orbital average radiator capacity at an arbitrarily selected but representative 20°C. Of course, radiator temperature will not really be constant over an orbit. It was assumed that radiator thermal capacitance per unit area would be large enough to damp out significant temperature changes induced by the considerable planet-shield orbital-temperature variations. In Fig. 5, it is seen that the net radiator capacity at 20°C is predicted to be no less than 260 W/m^2. Radiator capacity varies dramatically with temperature as shown in Fig. 6. This figure demonstrates the importance of higher allowable electronic temperatures and underscores the difficulty of cooling cryogenic detectors.

To judge the effectiveness of the design, let us define its figure of merit to be the ratio of orbit average net radiator capacity at 20°C to the capacity which would be obtained, if the same radiators, held at 20°C, were given an exclusive and unobscured view of cold, black space (i.e., no sun, no planet, no planet shade). In the latter situation, the radiators would dissipate about 321.8 W/m^2. Thus, the figure of merit for the design is no less than 0.809 (i.e., 80.9%).

It was stated earlier that the planet shade will experience wide temperature variations during a given orbit. Figure 7 illustrates (for the worst-case orbit) just how dramatic such variations can be.

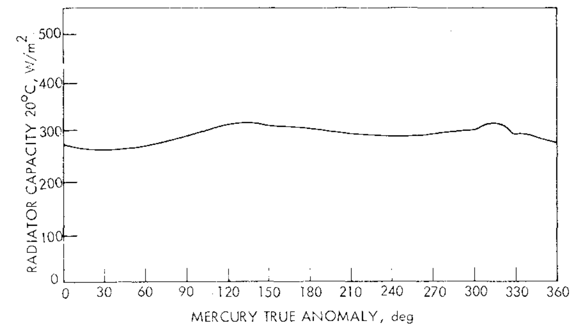

Fig. 5 Radiator capacity.

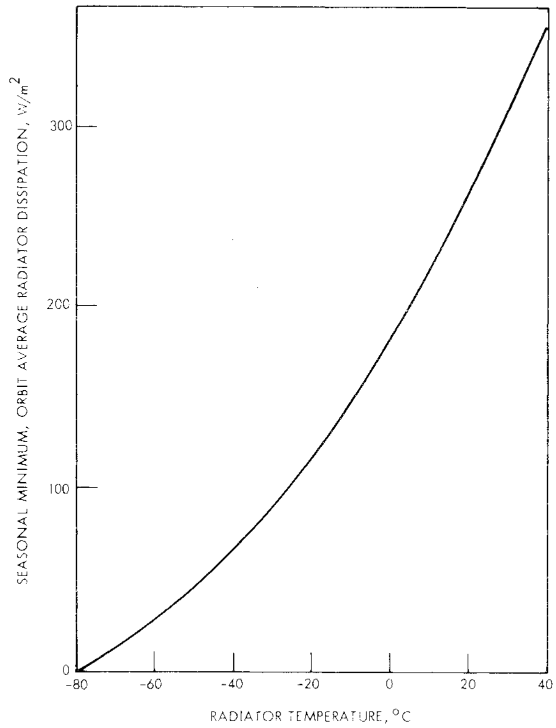

Fig. 6 Orbit average heat rejection capability.

The assumption that radiator mass is adequate to damp out radiator temperature changes is crucial to the success of this thermal control concept. Figure 8 shows temperature excursion vs radiator thickness, if it is assumed that the radiator is aluminum. From the figure, if an excursion of 10°C (e.g., from 20 to 30°C) is allowed per orbit, the radiator must be 0.66-cm thick. In application, however, this means that the mass of the radiator must be equivalent to 0.66 cm of aluminum. This equivalence includes the electronic components attached to the radiator as well as the radiator itself. Viewed in these terms, the requirement becomes quite modest since it implies a mass per unit area of 6.85 kg/m^2. As an example, such a radiator (equivalent to a shear plate on present spacecraft) might have an area of 0.25 m^2 and weigh 20 kg (including radiator and electronics). As can readily be seen, this far exceeds the requirement.

Moderate Temperature Detectors

Science instruments which operate at moderate temperatures (20°C or above) can be controlled rather easily when mounted in the bus and connected thermally to the radiators. This may be accomplished by either direct mounting on the radiator plates or by use of thermal conductors. For high heat loads, heat pipes may be required. For more modest loads, conductive structures, with careful attention to thermal interfaces, should suffice. More detailed consideration of this problem depends upon a better un-

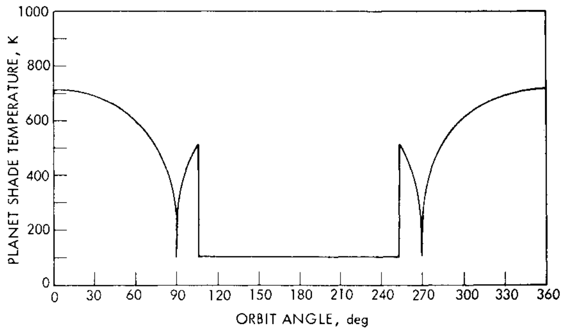

Fig. 7 Planet-shade temperature variations.

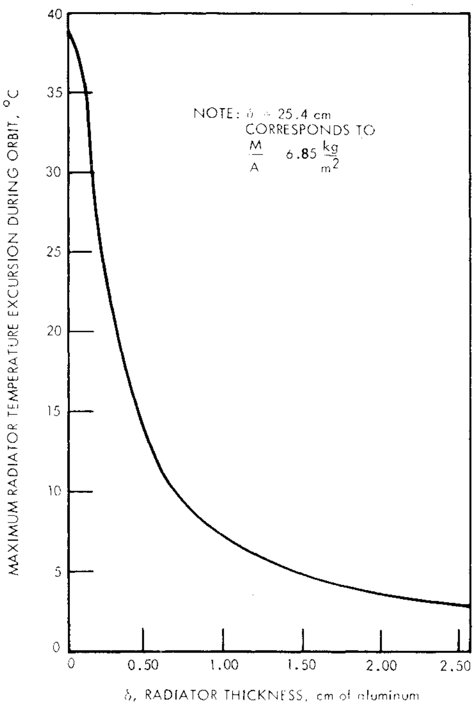

Fig. 8 Temperature excursion vs radiator thickness.

derstanding of the science instrument hardware design. In general, this information is not well developed at this point; hence, no detailed work has been done.

Low-Temperature Detectors

Some science instruments will require detectors operating at fairly low temperatures. Typical examples are charged coupled device (CCD) imaging systems (230 K) and the gamma ray spectrometer (100 K). Attaining these temperatures for long-term operation requires special effort. For imaging systems, dedicated low-temperature radiation segments within the main electronics radiators will suffice to attain the relatively moderate temperatures required. Since the basic instrument chassis is buried in the bus and slaved to the main radiators, the instrument temperature will not significantly exceed room temperature. This renders detector cooling relatively easy.

The gamma ray spectrometer presents a far more difficult problem because, in addition to its 100 K detector requirement, it must be mounted on a boom at some distance from the bus to minimize background noise.

Dual Spacecraft Concept

The mission concept of two spacecraft in two different orbits about Mercury evolved from the distinct orbital preferences of the scientific investigations. For planetology investigations (Table 1), a low, 1.5-h period, 105-km circular orbit is preferred, whereas, for the fields and particle investigations (Table 2), a high, 12-h period, elliptical orbit is preferred. The two spacecraft, when launched from the Space Transportation System (STS) are coupled together and inserted into the high, 12-h period, elliptical orbit with periapsis at 105 km by a large space-storable retropropulsion system. Once on orbit, the large retrosystem inerts are jettisoned. The two spacecraft are then separated. With a solid retrorocket, one of them performs a circularizing periapsis burn, placing it in the low, 1.5-h period, 105-km circular orbit. This dual spacecraft concept yields two very similar spacecraft. Although the scientific payloads are different, the supporting telecommunications, data handling and control, attitude control, temperature control, structure, and configuration are basically the same. The notable differences are due primarily to the different thermal requirements on the two power subsystems. The high orbiter is able to use silicon high-temperature solar cells (previously flown on HELIOS) whereas, because of the severe thermal environment, the low orbiter uses a solar thermoelectric generator (STG) array.

Each spacecraft has a 12-A-h NiCad battery for power during launch and periods of solar occulation. While the two spacecraft are coupled in flight, the high-orbiter solar array provides power to both spacecraft. Once separated in Mercury orbit, the low orbiter is self-powered by its STG array and the high orbiter by its solar cell array. Both arrays move in the plane parallel to the radiator faces to track the Sun. In general, because of the extreme thermal environment, temperature control was the primary consideration in the design of the spacecraft configuration and attitude control for both orbiters. The attitude control for both orbiters is three-axis stabilized during the heliocentric transfer and nadir-pointing/Sun-pointing during on-orbit operations.

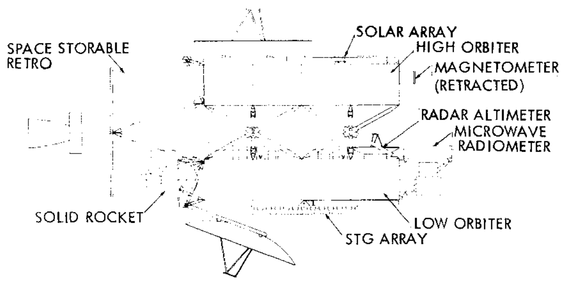

Fig. 9 Earth-Mercury transit configuration.

Table 1 Low orbiter payload

Low orbiter payload	kg	W	Peak, b/s
Altimeter	7	17	625
Heat flow	15	12	120
γ-ray spectrometer	12	6	2,000
x-ray spectrometer	10	10	256
Compositional infrared	10	7	10,000
Imaging	3	4	65,000
Neutral mass spectrometer	5	7	8
Total	62	63	78,000

Table 2 High orbiter payload

High orbiter payload	kg	W	Peak, b/s
Magnetometer	6	4	150
Particles	9	10	400
Solar neutron detector	20	5	50
UV spectrometer	4	2	200
Total	39	21	800

Table 3 On-orbit coupled vehicle mass summary

High orbiter	Mass, kg	Low orbiter	Mass, kg
Science module		Science module	
Science	39	Science	62
Base module		Base module	
Engineering mechanics (excluding propulsion module)	78	Engineering mechanics (excluding propulsion module)	78
Communications	32	Communications	32
Data handling and control	23	Data handling and control	24
Pyrotechnics	5	Pyrotechnics	5
Attitude and articulation control system	48	Attitude and articulation control system	48
Cabling	30	Cabling	30
Power	26	Power	32
Thermal control	10	Thermal control	10
Propulsion module		Propulsion module	
Propulsion – on-orbit (including attitude propulsion system [APS]): inerts (including engineering mechanics), propellant (hydrazine)	44	Propulsion – on-orbit (including attitude propulsion system): inerts (including engineering mechanics); propellant (hydrazine); solid retromotor	245
Propulsion – jettisoned post-Mercury orbit insertion: inerts, propellant (space storable)	(jettisoned)		
Contingency (20% base and science modules)	58	Contingency (20% base and science modules)	64
Total for high orbiter	393	Total for low orbiter	630

Total mass on 12-h orbit	1023 kg
Delivered mass capability	1252 kg
Mass margin	229 kg

Table 4 Power profile for low orbiter

Power parameters	Launch to injection (S-band, low power), W	Transit cruise (S-band, low power), W	Trajectory correction/ orbit insertion maneuvers (S-band, low power), W	Orbit sustenance maneuvers (S-band, high power), W	Terminator orbit (S-band, low power; X-band, high power), W	High Sun orbit (X-band, high power), W	Solar occultation (X-band, high power), W
Power requirements							
Articulation and attitude control	9	9	9	31	48	48	18
S/X-band transponders	37	37	37	87	117	69	69
Telemetry modulation units	3	3	3	3	5	3	3
Data handling and control	18	12	12	12	33	33	31
Temperature control	25	25	25	50	0	8	8
Science	0	0	0	0	63	63	63
Power housekeeping/ conversion losses	6	6	6	11	15	19	12
Battery charging	0	6	0	0	12	95	0
Total load	94	98	92	194	293	338	204
Power available (STG's)							
1.0 AU	N/A	100[a]	[b]				
0.45 AU		100[a]	[b]	350	350	350	N/A
0.3 AU				350	350	350	N/A
Power margin							
1.0 AU	[b]	+2	[b]				
0.45 AU		+2	[b]	[b]	+57	+12	[b]
0.3 AU				[b]	+57	+12	[b]

[a] From outer orbiter solar array.
[b] On battery.

Table 5 Power profile for high orbiter

Power parameters	Launch to injection (S-band, low power) W	Transit cruise (S-band, low power), W	Trajectory correction/ orbit insertion maneuvers (S-band, high power), W	Orbit sustenance maneuvers (S-band, high power), W	Terminator orbit (X-band, high power), W	High-Sun orbit (X-band high power), W	Solar occultation (X-band high power), W
Power requirements							
Inner orbiter	0	100	100	0	0	0	0
Articulation and attitude control	9	31	31	31	48	48	18
S/X-band transponders	37	37	87	87	69	69	69
Telemetry modulation unit	3	3	3	3	5	3	3
Data handling and control	18	12	12	12	33	33	31
Temperature control	30	30	30	50	0	2	8
Science	0	0	0	0	21	21	21
Power housekeeping/ conversion losses	6	13	15	10	10	10	9
Battery charging	0	6	0	0	8	12	0
Total load	103	232	278	193	194	198	159
Power available (solar cell array)							
1.0 AU	N/A	285	285				
0.45 AU		310	310	310	310	310	N/A
0.3 AU				200	200	200	N/A
Power margin							
1.0 AU	a	+53	a				
0.45 AU		+73	a	a	+116	+112	a
0.3 AU				a	+ 6	+ 2	a

[a] On battery.

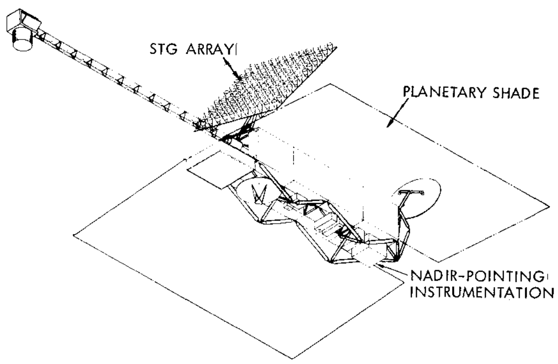

Fig. 10 Low orbiter on-orbit configuration.

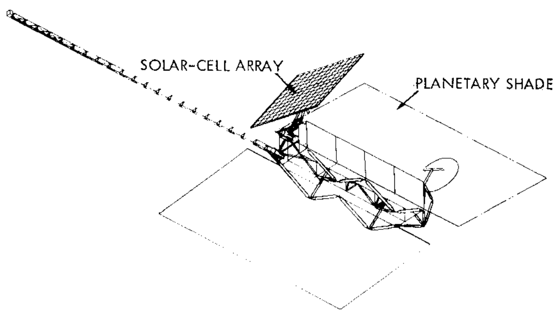

Fig. 11 High orbiter on-orbit configuration.

Figure 9 is an orthographic projection of the two orbiters in their coupled configuration during the Earth to Mercury transfer. During coupled flight following insertion into Mercury orbit, the configuration remains the same except that the large space-storable propulsion system has been jettisoned. Table 3 is the mass breakdown of the coupled vehicle on orbit. Note that, although these spacecraft designs have not been optimized for minimum mass, there still remains a good mass margin for the 1986 multi-Venus flyby opportunity,[3,7] that may allow inclusion of an additional small rough lander. It is anticipated that additional study and refinement would result in still greater mass margin.

Descriptions of Spacecraft

The two orbiters are composed of the following basic subsystems: 1) structure, 2) telecommunications, 3) data handling and control, 4) pyrotechnics, 5) attitude and articulation control, 6) cabling, 7) power, 8) temperature control, and 9) propulsion.

The structure, telecommunications, data handling and control, pyrotechnics, attitude and articulation control, cabling, and temperature control subsystems are essentially identical for both the low orbiter and the high orbiter. The power subsystems are basically the same with the exception of the power source. Whereas both orbiters have a hot gas (hydrazine) mass expulsion system for attitude propulsion and orbit sustenance (150 m/s), the low orbiter also contains a solid retromotor for the circularization burn and additional hydrazine for thrust vector control.

Figure 10 shows the low-orbiter configuration, and Fig. 11 the configuration for the high orbiter. (See Table 3 for the summary mass breakdowns.) A functional block diagram for the low orbiter is shown in Fig. 12. The block diagram for the high orbiter is basically the same, except for the science payload, and the power interface between the two spacecraft. Thermal blankets and insulation are deleted for clarity in the figures. Tables 4 and 5 present power profiles for the high and low orbiters, respectively. Note that the high-orbiter profile includes power for the low orbiter during the early portions of the mission.

Conclusions

From the results of this study, it may be concluded that the severe and conflicting requirements of a Mercury orbiter mission can be resolved through use of the dual spacecraft

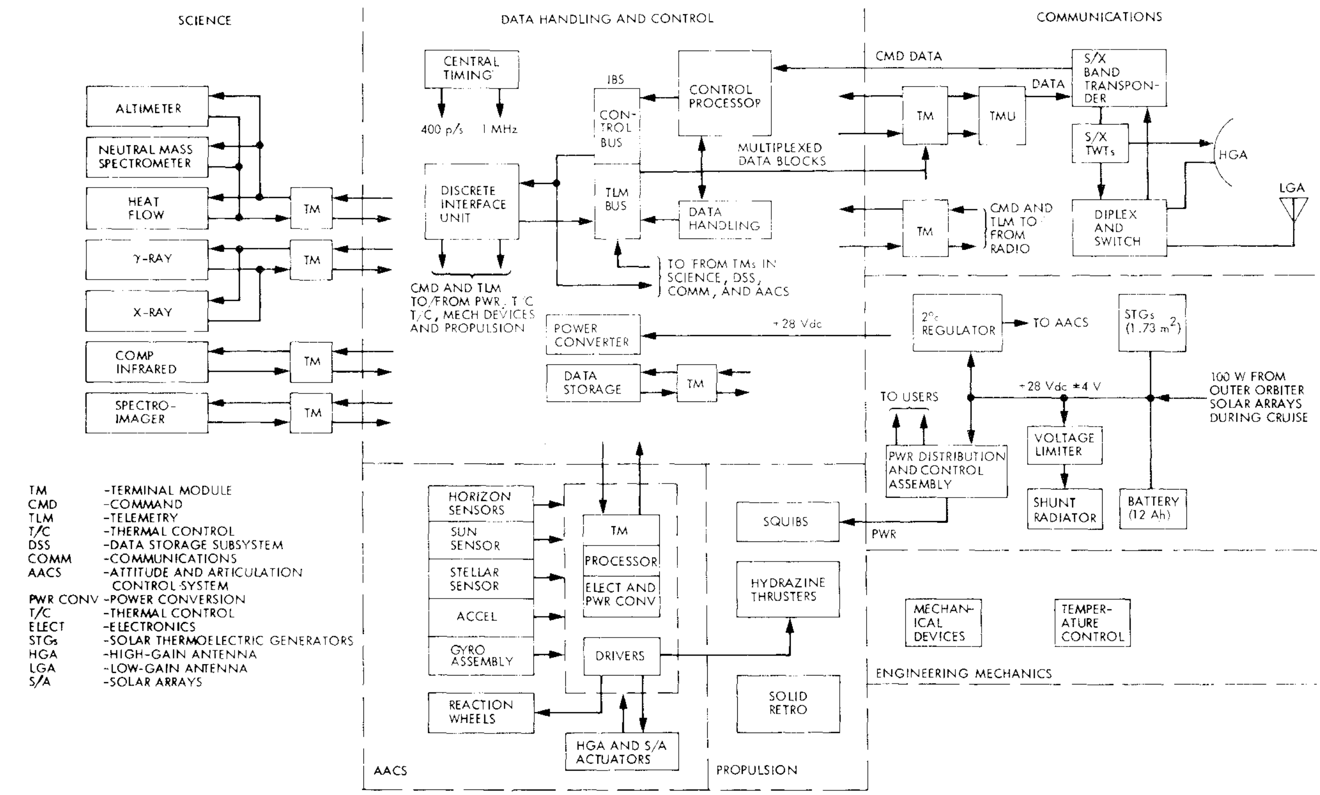

Fig. 12 Low orbiter functional block diagram.

concept that has been defined, and that the concept appears workable for a ballistic mission using multiple Venus swingbys. Dual spacecraft missions are possible in 1986 and 1994 with ample mass margins. The spacecraft design herein is by no means optimized, and such optimization might allow sufficient mass margin for inclusion of a small rough lander vehicle.[4] A single orbiter mission using the low orbiter only is feasible for any of the multi-Venus flyby opportunities in the next two decades, even assuming the weight growth required to allow the low orbiter to fly the interplanetary cruise part of the mission solo.

Thermal control of the low orbiter appears quite feasible based upon the techniques described herein. Some work needs to be done in some areas of technology, but it does not appear that any "breakthroughs" are required.

In summary, there is little question but that a significant and worthwhile Mercury orbiter mission can be conducted in the coming decade.

Acknowledgment

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