Helios Mission Support

P. S. Goodwin
Mission Support Office

This article relates the historical factors that led to the establishment of Project Helios; describes the project management relationships between the United States and the Federal Republic of West Germany, and the role of the Deep Space Network; and gives a brief description of the spacecraft and its telecommunications subsystem.

I. Background

The Space Act of 1958 authorized the National Aeronautics and Space Administration (NASA) to conduct programs of international cooperation with other nations in the peaceful exploration of space. To implement this article in the NASA Charter, the U.S. National Academy of Science, in behalf of NASA, introduced an offer of international space cooperation to the Committee on Space Research of the International Council of Scientific Unions. This, in turn, led to discussions between United States and Federal Republic of West Germany officials on the subject of a possible advanced cooperative project in space. An agreement in principle was made between the German Minister for Scientific Research, Gerhard Stoltenberg, and NASA Administrator James E. Webb in September of 1966 to carry out a cooperative solar probe project—provided that a mission of mutual interest could be defined. This decision was made official during Chancellor Erhard's visit with President Johnson in November 1966. This project was subsequently named *Helios* after the ancient Greek goddess of the sun.

To implement this agreement, NASA and the West German Minister for Science established a *Helios* Mission Definition Working Group in July 1968. This group's effort culminated in the publishing of the *Helios* Program "Mission Definition Group Report," dated April 1969. This report recommended that two solar probes be launched toward the sun—one each in calendar years 1974 and 1975—to achieve a perihelion distance from the sun of approximately 0.3 AU. Each spacecraft would carry ten experiments to perform fields and particle measurements in the region between the earth and the sun. The target perihelion distance of 0.3 AU was selected because it was a heretofore unexplored region of interplanetary space and was within the estimated extrapolation of the state-of-the-art for the high-temperature solar cells necessary to generate spacecraft power. The size and weight of the proposed *Helios* spacecraft was recommended to be compatible with the *Atlas/Centaur* launch vehicle, with the addition of a TE-364-4 solid propellant third-stage.

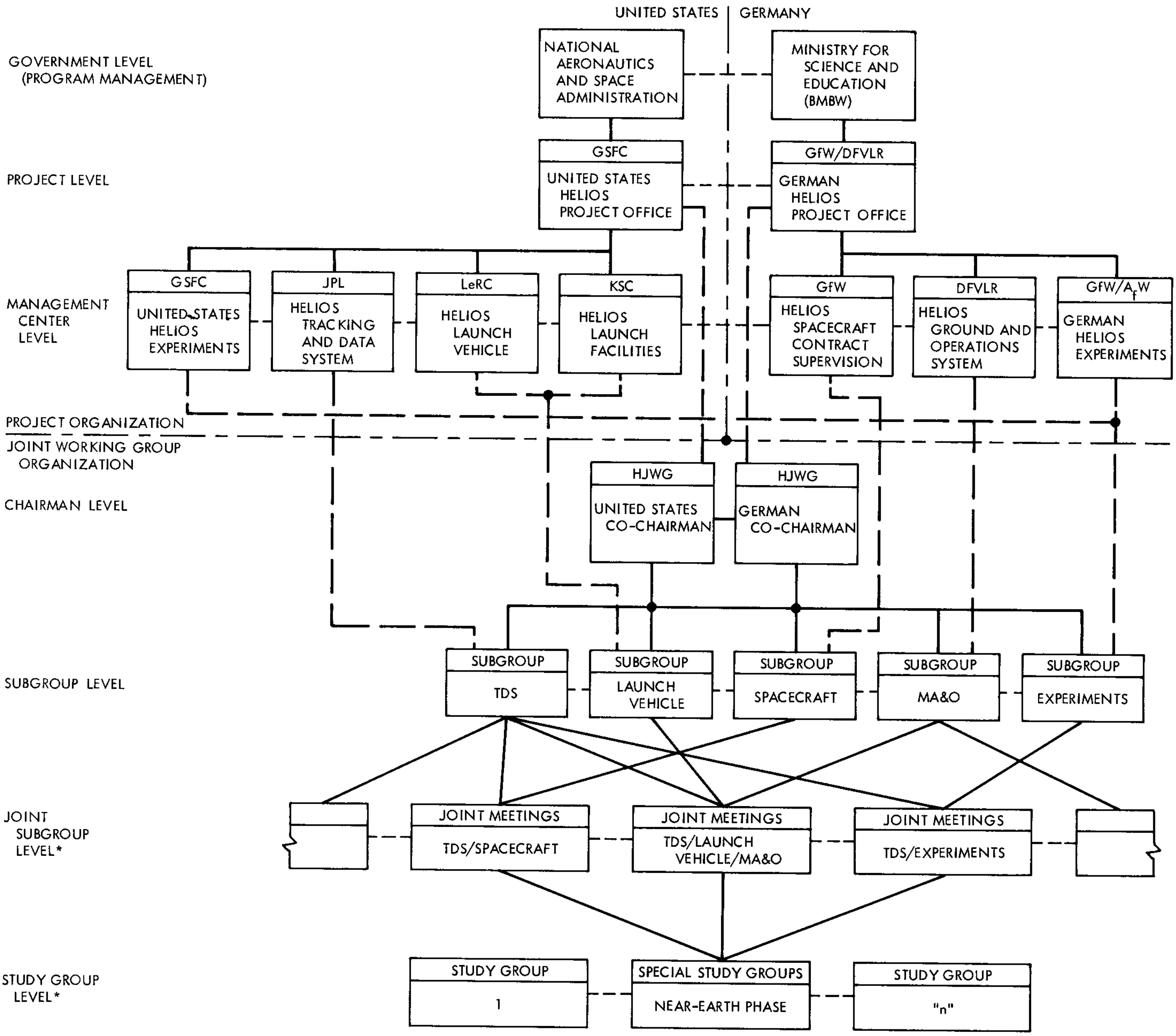
The publication of the *Helios* Mission Definition Report was followed by a "Memorandum of Understanding" between NASA Administrator Thomas Paine and German Science Minister Gerhard Stoltenberg in June 1969, and by joint statements by President Nixon and Chancellor Kiesinger in August 1969. These documents ratified the activities to date and established the basis for the future relationships between the two participating countries. The *Helios* Project would be managed jointly by both countries, with each co-manager being responsible for those elements of the cooperative project that were assigned to his particular country. In brief, the West German Project Manager would be responsible for the design, development, and fabrication of the spacecraft and the mission design and operations, while the U.S. Project Manager would be responsible for the launch vehicle, the launch facilities, and the tracking and data system. Of the ten onboard scientific experiments, seven were to be of German origin, and three of U.S. origin—however, the integration of all ten experiments into the spacecraft would be the responsibility of the West Germans. Each country would be responsible for providing the funding necessary to accomplish its portion of the program. Coordination of the effort was to be achieved via semiannual Joint Working Group Meetings to be held alternately in each country. The first *Helios* Joint Working Group Meeting was held in September 1969 in Bonn, West Germany. Subsequently, two additional Joint Working Group Meetings have been held: the second in April 1970 at Goddard Space Flight Center, Greenbelt, Maryland; and the third in October 1970 in Bonn. The next *Helios* Joint Working Group Meeting is scheduled for late April 1971 at Goddard Space Flight Center.

II. *Helios* Project Management Organization

In addition to the scientific objectives mentioned above, a second and very important objective of the *Helios* Project is to develop a broad governmental–educational–industrial technological base within the Federal Republic of Germany to conduct space research. Therefore, the West German participation in the *Helios* Project is not solely restricted to the development of the spacecraft and the mission design, but also includes the development of German tracking facilities, a German Control Center, and a full mission operations organization to conduct the mission. In addition, the international agreement provides for the cross-training of a significant number of West German specialists at various NASA installations to learn U. S. techniques pertaining to space exploration. These factors, together with the international character of the project, account for the "committee-like" structure of the *Helios* Project Management Organization (Fig. 1).

The upper lefthand quadrant of Fig. 1 depicts the familiar NASA flight project organizational structure wherein NASA Headquarters assigns the project management responsibility to one of its field centers, with functional support in specific areas being provided by other NASA field centers. The significant difference here is that only a portion of the elements comprising a total flight project are represented. The missing elements appear on the West German side of the interface in the upper righthand quadrant of Fig. 1—along with some new elements due to the factors mentioned above. The West German *Helios* Project management is seen to parallel and complement the U.S. *Helios* Project organization so that, in total, the top half of Fig. 1 represents the formal international project organization for Project *Helios*. An important advantage of this formal structure is that it provides a clear and distinct division of responsibility between the two countries in the administration, technical supervision, and financial management of the *Helios* Program.

As mentioned above, the technical coordination of the two countries' efforts is accomplished via the *Helios* Joint Working Group activities. The *Helios* Joint Working Group organizational structure is depicted in the bottom half of Fig. 1. In accordance with the international agreement, the Joint Working Group Meetings are co-chaired by the U. S. and West German *Helios* Project managers, respectively. Reporting to them are the chairmen of the various technical subgroups that support the project. These subgroup chairmen are the same individuals who have been assigned the equivalent functional responsibility in the formal project organization depicted in the top half of Fig. 1. However, the subgroup panel membership within each of these technical subgroups is fairly evenly divided between the U. S. and West German representatives in order to achieve internationally optimum solutions to problems facing the project. During the semiannual *Helios* Joint Working Group sessions, these subgroups meet both individually to resolve problems within their own areas of specialization, and jointly to resolve problems associated with the interface between the areas of technical responsibility. When necessary, the activities of the subgroups are augmented by special task or study groups assigned to investigate in detail a particular aspect of the program. A recent and successful example of the latter was the establishment of a special study group to develop a typical near-earth phase sequence of events following launch in order to determine that the spacecraft design as currently contemplated will fulfill all operational constraints upon the mission. This study group was chaired by the Mission Analysis and Operations Subgroup and its membership was comprised of representatives from each of the other subgroups. Once it has accomplished this initial task, the Near-Earth Phase Study Group will probably become



* ONLY PRESENT ACTIVITIES PERTAINING TO THE TDS ARE DEPICTED

A_fW ARBEITSGEMEINSCHAFT FUER WELTRAUMFORSCHUNG (COOPERATIVE FOR SPACE RESEARCH)

BMBW DER BUNDESMINISTER FUER BILDUNG AND WISSENSCHAFT (FEDERAL MINISTER FOR EDUCATION AND SCIENCES)

DFVLR DEUTSCHE FORSCHUNGS-UND VERSUCHSANSTALT FUER LUFT-UND RAUMFAHRT E.V. (GERMAN RESEARCH AND EXPERIMENTAL INSTITUTION FOR AEROSPACE)

GfW GESELLSCHAFT FUER WELTRAUMFORSCHUNG, mbH (SOCIETY FOR SPACE RESEARCH)

GSFC GODDARD SPACE FLIGHT CENTER

HJWG HELIOS JOINT WORKING GROUP

KSC KENNEDY SPACE CENTER

LeRC LEWIS RESEARCH CENTER

MA&O MISSION ANALYSIS AND OPERATIONS

TDS TRACKING AND DATA SYSTEM

Fig. 1. *Helios* Project Management Organization

dormant until the final launch trajectories are developed. At that time, the group is expected to reconvene to prepare the final near-earth phase sequence of events.

The technical decisions reached during the *Helios* Joint Working Group Meetings are reviewed by the co-chairmen. Upon their approval, these decisions are routed via the respective project office to the formal organization (top portion of Fig. 1) for implementation. In the case of decisions affecting the U. S. Tracking and Data System (TDS), these are routed to the Jet Propulsion Laboratory as the cognizant NASA field center for *Helios* TDS management. The NASA TDS function for *Helios* has two major subdivisions: (1) support from the near-earth phase facilities, and (2) support from the Deep Space Network (DSN). The near-earth phase facilities have TDS responsibility from launch up to that portion of the trajectory wherein the DSN has continuous visibility of the spacecraft, at which time the DSN assumes responsibility for the TDS function. The near-earth phase facilities are composed of selected stations from the NASA MSFN and STADAN Networks and from the Air Force Eastern Test Range (AFETR). The selection of stations from these facilities is a function of the particular trajectory flown and the individual project's data requirements. The selection is individually negotiated by the JPL Cape Kennedy Field Station for each flight project, but in general is composed of both tracking stations (including radars) and telemetry reception stations. The near-earth phase facilities may or may not have a requirement to send commands to the spacecraft—depending upon the criticality of the flight sequence during the near-earth phase of the particular mission. However, the AFETR at all times retains responsibility for range safety. Data from the near-earth phase stations are fed via a combination of NASCOM and/or AFETR communications circuits back to Cape Kennedy, Florida, where flight operations are conducted for the near-earth phase of the mission. During this period, radiometric data are fed to the Cape Kennedy Real-Time Computer Complex, where early trajectory computations are performed. Near-earth phase telemetry data are displayed at the Mission Control Center at Cape Kennedy and are also routed to the DSN's Space Flight Operations Facility (SFOF) in Pasadena, California. When the spacecraft reaches the pre-determined point in the trajectory, the TDS responsibility is transferred to the DSN. By this time, mission operations responsibility has been transferred from Cape Kennedy to the SFOF.

The other part of the NASA Tracking and Data System is the Deep Space Network. Its function in supporting *Helios* is to provide near-continuous tracking, telemetry, and command support from initial acquisition through the end of the primary mission. In accomplishing these objectives, the DSN will request and receive TDS support from the stations being implemented in West Germany. Also, as noted before, the West Germans have responsibility for mission operations. During Phase I, the period from launch through 2 to 4 weeks following launch, the West German Mission Operations Team will reside in the SFOF. At the conclusion of Phase I, *Helios* mission operations will be transferred from the SFOF to the German Control Center in Oberpfaffenhofen (near Munich), West Germany. This will initiate Phase II, which will conclude at the end of the primary mission. During Phase II, the German Control Center will act as a remote terminal to the SFOF for the conduct of mission operations; however, network operations will continue to be conducted from the SFOF since the DSN retains TDS responsibility through the end of Phase II. Because of these additional operational interfaces, the DSN has representative membership in the TDS and MA&O Subgroups (center portion of the lower half of Fig. 1). In addition, the DSN has been assigned the responsibility for training over a dozen West German specialists. These factors explain why the DSN's support to the *Helios* Project is slightly more complicated organizationally than the support provided to a typical U. S. flight project. However, the organizational structure depicted in Fig. 1 has been functioning very satisfactorily to date and there is no anticipation of the need for a significant change to occur during the lifetime of the project.

III. Spacecraft Description

A. Physical Concept

A description of the current configuration of the spin-stabilized *Helios* spacecraft is helpful in understanding the support requirements that the spacecraft design places upon the Deep Space Network. The present *Helios* spacecraft concept differs only slightly in appearance from one of the two proposed configurations in the Mission Definition Report. The report had assumed that the spacecraft would be launched by an *Atlas/Centaur/*TE-364-4 launch vehicle combination. By the time of the second *Helios* Joint Working Group Meeting (April 1970), it became possible for the *Helios* Project to consider employing a *Titan-IIID/Centaur/*TE-364-4 launch vehicle combination. Since this was a possibility rather than a firm commitment, the *Helios* project managers decided that instead of permitting the spacecraft design to increase significantly in either size or weight, they would target the *Helios* mission profile to have a perihelion closer to the sun—i.e., 0.25 AU instead of 0.3 AU. This decision markedly altered the thermal design of the spacecraft (which now must withstand 16 solar constants[1] at perihelion instead of the previous 11 solar constants—due to the inverse square law effect), but would not preclude flying the new configuration to the original 0.3-AU perihelion in the event it was necessary to return to the *Atlas/Centaur*/TE-364-4 launch vehicle. As a result, the alternate cylindrical shape for the main body of the spacecraft recommended in the Mission Definition Report was discarded in favor of the spool shape shown in Fig. 2. The conical sections of the spacecraft body in the current design reflect unwanted solar energy away from the spacecraft structure while at the same time provide for a reasonable angle of incidence between the sun rays and the solar cells that are mounted on these conical surfaces. Also, because of the greater solar energy, a fewer number of solar cells were required. Therefore, the current design contains a mixture of 50% solar cells and 50% second surface (i.e., front surface) mirrors to reflect excess solar energy from the conical surfaces. The cylindrical midsection, containing no solar cells but all of the spacecraft electronics and the bulk of the onboard scientific experiments, remains basically unchanged from that depicted in the Mission Definition Report. The maximum spacecraft design weight of 260 kg (570 lb) is compatible with either a *Titan* or *Atlas* for the first stage of the launch vehicle combination. The overall clearance dimensions of the spool-shape spacecraft configuration can be accommodated by using either of the more recently designed *Viking*-Mars or *Intelsat IV* shroud systems (depending upon launch vehicle selection) rather than an extended version of the *Surveyor*-type spacecraft shroud contemplated during the Mission Definition Group's studies. Therefore, the present *Helios* spacecraft design configuration meets the objectives set forth in the Mission Definition Report when using the *Atlas/Centaur*, while at the same time permits trajectories to be flown closer to the sun—i.e., 0.25 AU and possibly 0.2 AU—when launched aboard a *Titan/Centaur* launch vehicle combination. For the present, planning is pro-

[1]A solar constant is defined as the average solar energy received at earth distance from the sun, i.e., the solar flux at 1.0 AU.

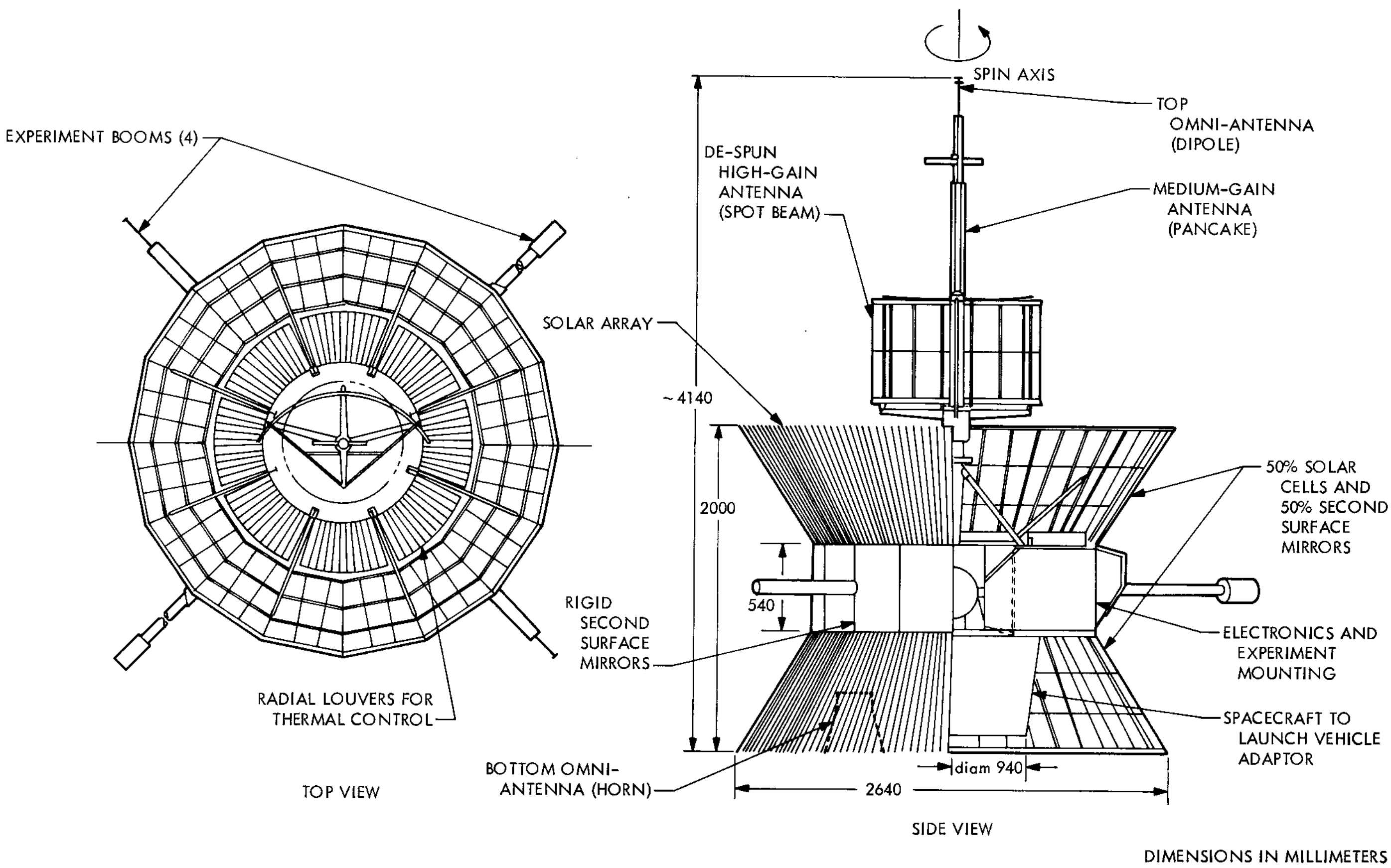


Fig. 2. *Helios* spacecraft

ceeding on the assumption that the *Titan/Centaur* launch vehicle will be used; however, these plans are constrained to always maintain compatibility to use the *Atlas/Centaur* vehicle. As might be expected, these launch vehicle constraints impact the thermal and structural design of the spacecraft much more significantly than they do the telecommunications system design.

B. Antenna System

Following injection of the spacecraft into its heliocentric orbit, and DSN acquisition, the spacecraft is positioned by command such that its spin axis (Fig. 2) is pointing to the pole of the ecliptic. This maneuver provides proper illumination of the solar cells and positions the spacecraft's antennas such that their patterns will intercept the earth throughout the trajectory of the spacecraft. There are three parts to the *Helios* spacecraft antenna system: (1) a two-antenna, quasi-omni-directional system; (2) a medium-gain antenna system whose "pancake" pattern lies in the plane of the ecliptic; and (3) a high-gain or "spot-beam" antenna that is mechanically oriented to point directly at the earth throughout the spacecraft's heliocentric orbit. These antennas are individually described below.

The omni-directional antenna system is composed of a linearly polarized dipole antenna atop the central mast (Fig. 2), and a circularly polarized horn antenna pointing downward and mounted inside of the bottom solar cell conical skirt. These two antennas are hard-wired to a fixed ratio power splitter, thence to one of the two onboard transponders (Fig. 3). The division ratio of the power splitter is adjusted such that the combination will provide a nearly uniform combined antenna pattern almost completely surrounding the spacecraft. The omni-antenna system finds its primary use during the period from shroud ejection following launch up through the completion of the orientation maneuver that positions the spin axis of the spacecraft toward the pole of the ecliptic. Since the latter maneuver is completed by the time the spacecraft reaches a distance from earth equivalent to the moon's orbit, the omni-directional antenna system might be considered the "near-earth" antenna system. However, this is not to imply that the omni-directional antenna system will not be used during later phases of the mission when signal margins permit.

The medium-gain antenna consists of a single longitudinal helix antenna mounted on the mast beneath the omni-antenna but above the high-gain antenna. This antenna produces a linearly polarized, pancake-like pattern which, following the above-mentioned maneuver, directs its maximum radiation in all directions within the plane of the ecliptic while minimizing its radiation toward the poles of the ecliptic. It is approximately 15 deg wide in the direction perpendicular to the ecliptic and has a downlink gain of 8 dB. It is the prior knowledge of this pattern that enables the final positioning of the spacecraft spin axis toward the pole of the ecliptic. The medium-gain antenna is connected through a second diplexer to the second transponder aboard the spacecraft. However, the transmitter portion of the second transponder is shared between the medium-gain antenna and the high-gain antenna described below.

The *Helios* spacecraft high-gain antenna is a cylindrical, parabolic-shaped reflector that is mechanically de-spun to counter the spacecraft body's spin-rate of 60 rpm. The de-spin angular velocity is adjusted to exactly counter-match the spacecraft body's spin velocity in order to achieve a fixed beam direction in space. The direction that the spot-beam points within the plane of the ecliptic is a phase angle adjustment with respect to the pulses received from the onboard sun sensor that clocks the speed of the spool-shaped main body of the spacecraft. Since the angle between the sun and the earth changes as the spacecraft traverses its elliptical heliocentric orbit, the phase angle for the high-gain antenna must be updated by earth command during the mission. The high-gain antenna beam is again linearly polarized with a beamwidth of 5½ deg within the plane of the ecliptic, and 14 deg normal to the plane of the ecliptic. The downlink gain of this antenna is 23 dB. It is used to transmit high-rate telemetry from the spacecraft to the earth, and operates in conjunction with the spacecraft receiving on the medium-gain antenna, using the second transponder mentioned above. Since the second transponder contains a turnaround ranging loop, it is possible to employ either the DSN lunar or planetary ranging system in conjunction with the spacecraft medium-gain antenna system. However, at perihelion distances and beyond, the spacecraft's radio system configuration will be such that it will receive ranging signals via the medium-gain antenna and return them to earth via the high-gain antenna.

C. Radio System

The foregoing *Helios* spacecraft antennas are connected to the onboard radio system, as depicted in Fig. 3. As mentioned previously, the spacecraft contains two complete transponders. Despite the fact that these transponders operate on the same frequencies (both uplink and downlink), they are not completely redundant

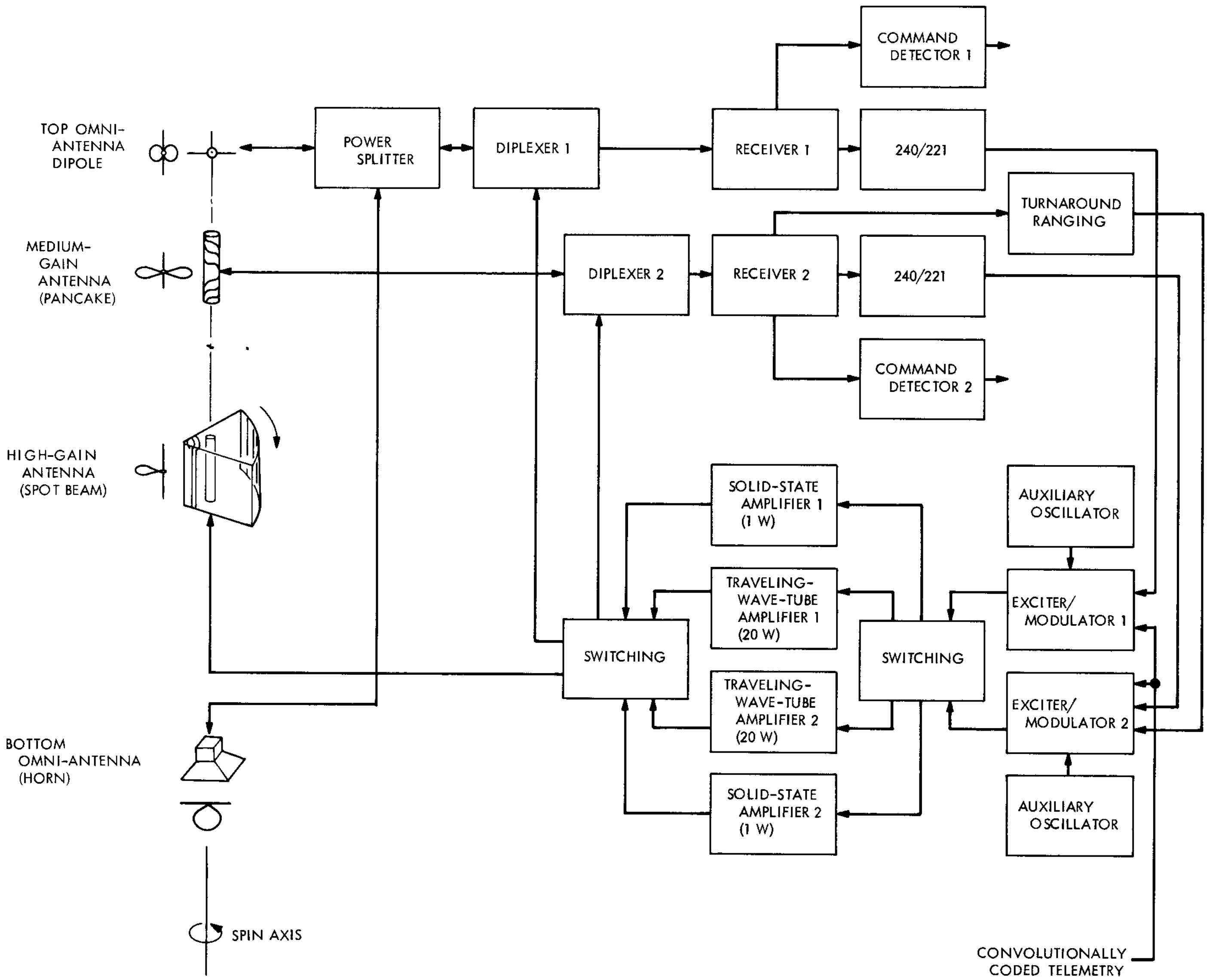


Fig. 3. *Helios* spacecraft radio system

in that the transponders are partially hard-wired and partially switchable. To explain the degree of redundancy provided by the present *Helios* spacecraft radio system design, it is best to describe the signal flow under various mission conditions.

From launch through the completion of the maneuver that orients the spin axis to the pole of the ecliptic, the spacecraft is operating with the omni-directional antenna system and with transponder 1. Prior to separation of the spacecraft from the TE-364-4 third-stage, uplink and downlink communication is via the top omni-antenna (since the bottom omni-horn-antenna is masked by the third-stage rocket motor). At separation, the spacecraft is spinning with the spin axis oriented along the velocity vector of the outgoing trajectory, which lies in the plane of the ecliptic. Since, under these circumstances, the medium-gain antenna pattern intercepts but a small segment of the earth, the omni-antenna system is used for initial DSN acquisition. The two-antenna omni-system provides nearly spherical antenna pattern coverage except for a null directly ahead of the spacecraft. Since the "look-angle" from the spacecraft to an earth station will traverse this forward null very rapidly as the spacecraft departs the vicinity of the earth, the existence of this forward null does not impose a serious initial acquisition constraint. Following separation, onboard sun sensors cause the spacecraft's attitude control system to perform the first of two orientation maneuvers—namely, to position the spacecraft's spin axis such that the spacecraft's solar cells receive maximum energy from the sun. During this (step I) maneuver, the spacecraft's spin axis remains basically in the plane of the ecliptic. Since, depending upon the trajectory, the initial DSN acquisition of the injected spacecraft can occur prior to, during, or subsequent to the completion of the step I maneuver, a nearly uniform spacecraft omni-directional system is needed. Following completion of the step I maneuver, the spacecraft's attitude remains fixed while the *Helios* Mission Operations Team in the SFOF evaluates the condition of the various onboard systems, using spacecraft telemetry. Following this, selected scientific experiments aboard the spacecraft are activated in the near-earth science mode. The near-earth science phase is complete by the time the spacecraft reaches lunar distance from the earth. At this time, the science instruments are deactivated and the spacecraft is readied for the second orientation maneuver (step II). During the step II maneuver, the spin axis is commanded to precess in a direction that will orient it to the pole of the ecliptic. As this is done, the medium-gain antenna's pancake-like pattern will slowly begin to intercept the earth. At this time, the spacecraft is commanded to operate with transponder 2 in the diplex mode via the medium-gain antenna. Using the received signal strength at the DSS as an indicator, the spacecraft is commanded to continue precessing its spin axis until the pancake antenna pattern causes a maximum signal strength indication to be achieved on earth. The maximum is determined by causing the spacecraft to precess beyond its optimum orientation and then return to the optimum position—at which time the spin axis will be oriented toward the pole of the ecliptic. Following completion of the step II maneuver, the de-spin velocity and phase angle of the high-gain antenna are adjusted by command to direct the "spot beam" toward earth. Since the spinning spacecraft will be self-stabilized (gyroscopically) in this position (unless unexpectedly perturbed), no further orientation maneuvers should be necessary. All ten onboard scientific instruments are then activated and commanded to return data via the high-gain antenna at the maximum information rate permitted by the telecommunications system capability through the remainder of the mission. The foregoing orientation maneuver considerations explain in part the rationale behind the radio system block diagram shown in Fig. 3. Other aspects will be discussed in the paragraphs that follow.

The benefits achieved by having redundant components aboard the spacecraft must be traded off against the reliability of the switching circuits necessary to utilize the redundant component. Since these studies are still in process, the *Helios* Project spacecraft radio system depicted in Fig. 3 is subject to change with time. Nevertheless, an understanding of the basic design philosophy of the *Helios* spacecraft radio system as presently conceived is helpful in assessing its operational impact upon the DSN. For instance, it is interesting to note that the redundant components associated with the uplink from the DSN to the spacecraft have separate, hard-wired paths, while the redundant components in that portion of the transponder associated with the downlink are connected to switching matrices which permit the use of a number of alternate paths or operating modes in the event of a failure along one of the downlink paths.

Let us consider first the uplink portion of the spacecraft radio system. As mentioned previously, both transponder loops operate on the same center frequency. Also, both receiver 1 and receiver 2 are continuously energized in order to ensure an uplink command capability to the spacecraft. Since these receivers are hardwired to separate antennas, the selection of the uplink transponder path is determined by spacecraft logic circuits that select the stronger of the two signals from the receiver automatic gain control values. Prior to the completion of the step II orientation maneuver, it is anticipated that the stronger signal will be via the omnidirectional antenna and receiver 1 since the medium-gain antenna pattern will not have yet been oriented toward earth. Following the step II maneuver, the reverse should be true since the medium-gain antenna has an uplink advantage of approximately 6 dB over the omni-directional system. The transition characteristics between these two antenna/receiver paths are dependent upon both the actual trajectory being flown and the sequences employed in the orientation maneuvers. For these reasons, it is quite possible that future studies will show the need for some kind of mode control within the spacecraft receiver loops during the orientation maneuvers in order to prevent the interruption of two-way lock due to unwanted antenna switching. However, with respect to commands, the following precautions have already been taken: (1) The subcarrier frequency for command detector 2 has been made intentionally different from that for command detector 1; and (2) since the command bit rates for the two channels are submultiples of their subcarrier frequency, the bit rates to the two command detectors are different. These two features ensure that commands will be executed only via the pre-selected channel. The last major distinction between the two uplink channels aboard the spacecraft is that the turnaround ranging loop components are connected only to receiver 2. This is because uplink signal-to-noise ratio considerations via the omni-antenna system very rapidly diminish the usefulness of that path once the spacecraft has been injected into heliocentric orbit. Since the *Helios* spacecraft reaches a distance from earth equal to the moon's orbit approximately 9 h after launch, it was felt that the additional complexity and weight associated with a ranging loop via the omni-antenna was not justified.

Next, let us consider the downlink portion of the *Helios* spacecraft radio system. In contrast to the uplink portion of the system, the active downlink path through the radio system is under ground-commandable mode-control. As implied by Fig. 3, a considerable number of paths or modes are possible in the downlink portion of the radio system. For the moment, only the more significant modes or combinations will be discussed. First, the downlink portion of the radio system may be operated in either the non-coherent or the coherent mode in association with one of the uplink receivers. The non-coherent mode utilizes onboard auxiliary crystal oscillators to derive the transmitter's frequency. This mode is used principally prior to the initial DSN acquisition and again during solar occultation periods. The coherent mode employs the uplink frequency (as received aboard the spacecraft) multiplied by the ratio of 240/221 in order to translate it to the nominal downlink frequency. The coherent mode is required for two-way doppler tracking and for ranging.

The coherent mode is the primary mode of operation during the cruise phase of the mission. Regardless of whether the non-coherent or coherent mode is selected, only one exciter/modulator is activated at one time. The active exciter/modulator feeds one of four possible amplifier circuits, only one of which is active at a given time. The four power amplifier circuits are composed of redundant modules of two classes of amplifiers: (1) 1-W solid-state amplifiers whose primary service is during the near-earth phase of the mission, and (2) 20-W traveling-wave-tube amplifiers used during the main phase of the mission. Following the spacecraft power amplifiers is a switching matrix circuit which will permit the downlink to be sent to earth via any one of the three antenna systems described in *Section III, Part B*. Modulated onto this downlink carrier signal is the spacecraft's single-channel, convolutionally coded telemetry—which contains a preprogrammed commutation of both science and engineering data. Also on the downlink carrier signal is the turnaround ranging modulation from receiver 2, which is sent to exciter/modulator 2 to be returned to earth via either the medium-gain or the high-gain antenna system. This completes, for the moment, the basic functional description of the *Helios* spacecraft radio system.

IV. Conclusion

This article has attempted to provide the reader with a basic understanding of the international character of the *Helios* project—how it became established and how it is organized. In addition, the reader has been given an overview of the spacecraft configuration and a brief description of its onboard radio system. The next article will discuss the launch windows and probable launch dates for the two proposed *Helios* spacecraft missions, the nominal post-injection trajectories and attitude maneuvers of the spacecraft as they relate to the support to be provided by the DSN, and a discussion of the DSN tracking load imposed by support to other flight projects during the *Helios* time-period. These two articles will then form the baseline for detailed descriptions of the various subsystems within the *Helios* spacecraft radio system and their interfaces with the DSN, starting with the third article. However, since the *Helios* spacecraft configuration and mission profile design are subject to evolutionary improvement between the present time and the launch of the first spacecraft in the summer of 1974, future articles will also include any evolutionary design changes that affect the spacecraft/DSN telecommunications link, or the interface between the DSN and the *Helios* Project's Mission Operations Team.

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These documents are on file in The *Helios* Project Office, Goddard Space Flight Center, Greenbelt, Md., and in The *Helios* Project Office, Gesellschaft fuer Weltraumforschung, mbH, Bonn–Bad Godesberg, West Germany.

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