NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

Technical Report 32-953

Mariner Mars 1964 Basic Structure, Design and Development

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Contents

Contents (contd)

Figures

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Abstract

This report summarizes the design and development of the *Mariner* Mars 1964 spacecraft basic structure. Specific design considerations and the evolution of the structural design are discussed. The detail design is described, and the fabrication, assembly, and quality control procedures are presented. Finally, conclusions and recommendations are given that may be of value in structural design for future projects.

Mariner Mars 1964 Basic Structure, Design and Development

I. Introduction

The purpose of this report is to present the salient aspects of the design and development of *Mariner IV* basic structure. In this context, *design* includes the concern for operational details and consideration of fabrication and quality control problems, as well as the shaping and sizing of the structural elements to carry the imposed loads efficiently. The subject parameters are covered, as follows:

(1) The considerations that enter into spacecraft structural design

(2) The structural design process, including loading criteria and analysis techniques

(3) The methods of fabricating and testing the basic structure

(4) Problems encountered during the program and the resulting solutions

Finally, recommendations about spacecraft structural design, based on *Mariner IV* experience, that may be applicable to future designs are presented.

II. Spacecraft Description

The 575-lb *Mariner IV* spacecraft (Fig. 1) was launched by an *Atlas–Agena D* on November 28, 1964 on a 229-day trip to Mars. The objectives of the mission were to obtain (1) scientific data on space – both interplanetary and near Mars, (2) close-up television pictures of the Mars surface, and (3) Mars atmospheric data – by the occultation of the spacecraft RF signals. All of the objectives of the mission were successfully completed.

The spacecraft was attitude stabilized using the sun and the star Canopus as celestial references. All of the electronics were mounted in an octagon-shaped electronics compartment. Four solar panels, deployed after separation from the *Agena*, and two fixed antennas – a high-gain and a low-gain – were attached to the sunward side of the spacecraft. At the base of the spacecraft was the rotatable science platform containing the planet sensing instruments and the television camera. Other scientific instruments were mounted around the periphery of the spacecraft.

A midcourse trajectory correction was required several days after launch. This was done by a monopropellant

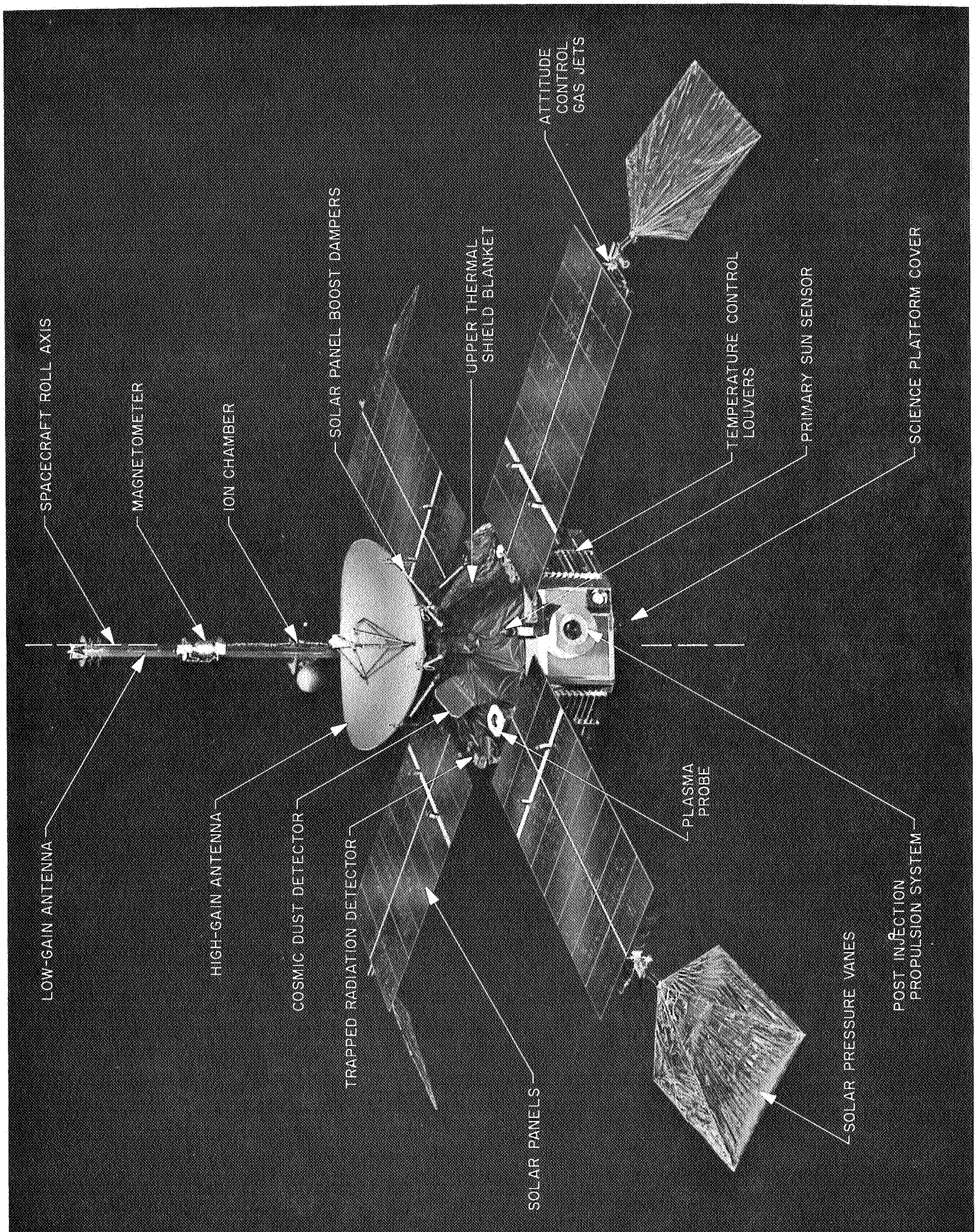

Fig. 1. ***Mariner IV*** **spacecraft in cruise configuration**

propulsion unit mounted in one bay of the spacecraft electronics compartment. The basic structure provided support for all major spacecraft components. The *Mariner* basic structure consisted of three elements:

(1) The primary, or octagonal, structure, was approximately 4½-ft diam × 18-in. high; around the periphery were mounting provisions for electronic equipment and the propulsion system; at the base were booster-attachment points. All of the peripheral spacecraft hardware, such as thermal shields, solar panels, sun sensors, etc., attached to this primary structure.

(2) The secondary structure, contained within the primary structure, supported the science platform and the two attitude-control-gas pressure vessels.

(3) The superstructure, mounted on the sunward side of the primary structure, supported the high-gain antenna, the cosmic dust detector, the spring-dampers supporting the solar panels during boost, and a spring-damper supporting and aligning the low-gain antenna. It also provided the support for the upper thermal shield.

III. Spacecraft Structure

A. Design Considerations

The basic structure of a spacecraft has the following functional requirements:

(1) Tie the spacecraft subsystem elements together in a coherent manner to meet the flight, ground handling, and prelaunch testing requirements.

(2) Provide structural integrity to the complete spacecraft during ground tests, boost, and propulsion maneuvers.

(3) Provide mechanical alignment between spacecraft components.

In meeting these requirements, the principal challenge to the basic structure design was to *minimize weight*. Since weight was a critical factor in determining the feasibility of the mission, this meant that a light, efficient structure was required — one that would satisfy the basic functional requirements and that could be fabricated and tested within the project schedule.

The design of the *Mariner IV* basic structure was accomplished concurrently with the determination of the spacecraft mechanical configuration. Through the spacecraft's basic structure, the mechanical configuration functionally integrates the spacecraft subsystems. Through the proper shaping of a spacecraft mechanical configuration, each subsystem is enabled to perform its functions in a workable fashion. Also, the spacecraft system can be properly exercised in the prelaunch phases of the development test program and in launch and flight environments. Because of this need for a system approach to shaping a spacecraft configuration, such responsibility was assigned to the project group responsible for the design of the spacecraft structure. However, the project group was supported by specialists who participated in the structural design, performed the structural and dynamic analyses, and developed and directed the structural test program.

The largest gains in arriving at a lightweight spacecraft structure do not come from refinement of structural details, but by a configuration that provides the most direct load paths consistent with other spacecraft requirements. For example, rather than attaching the solar panels to the basic spacecraft structure by struts or arms, the solar panels were mounted directly to the basic octagon structure. This direct load path offset any weight advantages that may have been gained by the most sophisticated weight reduction techniques on a less efficient structure. In general, a compact configuration leads to a lightweight and efficient basic structure design that is capable of being built and assembled on schedule.

As in any design work, configuration definition is an iterative process. A complete spacecraft layout is generated in an attempt to meet the requirements for: electronic equipment volume; temperature control; cabling; antenna, sensor, and science instrument look angles; location and method of attachment of articulated subsystems; structural integrity; accessibility; and *cg* control. Structural design approaches are generated for each possible configuration. Typically, 20 or 30 configurations may be studied before a workable spacecraft configuration is selected. The preliminary design phase of the *Mariner* Mars 1964 project resulted in several competing configurations.

B. Evolution of Structural Design

1. Basic structure design approaches.[1] One of the requirements imposed on the spacecraft mechanical design

[1]For additional information regarding the factors influencing the spacecraft mechanical configuration, see R. J. Spehalski, *Mariner Mars 1964 Mechanical Configuration*, Technical Report 32-933, Jet Propulsion Laboratory, Pasadena, California, September 1, 1966.

was that of using the 6- × 6-in. electronic subassembly profile (Fig. 2) that had been used in *Ranger* and *Mariner II*. It was felt, and later confirmed, that integrating the electronic assembly into the structure would provide significant weight reductions without compromising the design flexibility and operational accessibility. Structural designs were actively pursued involving four structural configurations.

a. Cylindrical configuration. This structure consisted of a cylindrical barrel to which the standard subassemblies would be individually mounted from the inside to the structure. Although structurally efficient, this approach was abandoned because of operational problems from the lack of subassembly accessibility during tests and operations.

b. Square configuration. Previous JPL spacecraft designs used a rectangular, machined pan in which a number of electronic subassemblies were mounted. These pans, or chassis, were then bolted to basic spacecraft structural frames. Here, an approach was pursued where four large electronic chassis, approximately 2 ft long, were used to shape the basic structure into a square. The electronic subassemblies inserted into these chassis, and the chassis, in turn, were bolted together to form a square spacecraft structure. The objective was to use the electronic chassis as structure. This approach was abandoned, primarily, because the size of electronic assemblies made them unwieldy to handle, and the total structure was inefficient.

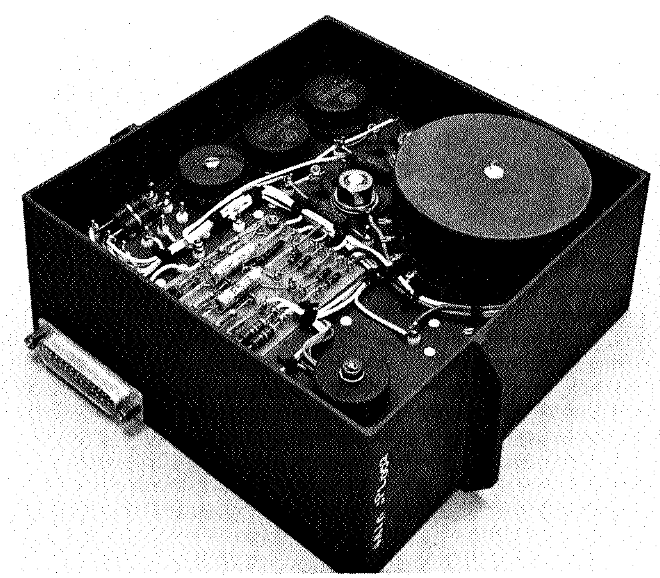

Fig. 2. Typical 6- × 6-in. electronic subchassis

c. Cruciform configuration. During the configuration studies, a structure was devised in which individual trays of electronic subassemblies would be inserted into a spacecraft structure that in planform looked like a cross. Although this approach represented a very efficient structure, it was abandoned because of insufficient electronic packaging volume and lack of growth potential.

d. Octagon configuration. The final design of the spacecraft primary structure was an outgrowth of a design that was fabricated and assembled for the *Mariner A* Venus–*Centaur* spacecraft (Fig. 3). The chosen design used integrally machined octagonal rings, which were separated vertically by longerons. The electronic chassis, which housed two vertical rows of subassemblies, bolted to openings formed by the longerons and rings. Structurally, these chassis were similar to those used on the *Ranger* and *Mariner II* designs. The rings and the longerons formed the sides of the chassis and the outer face

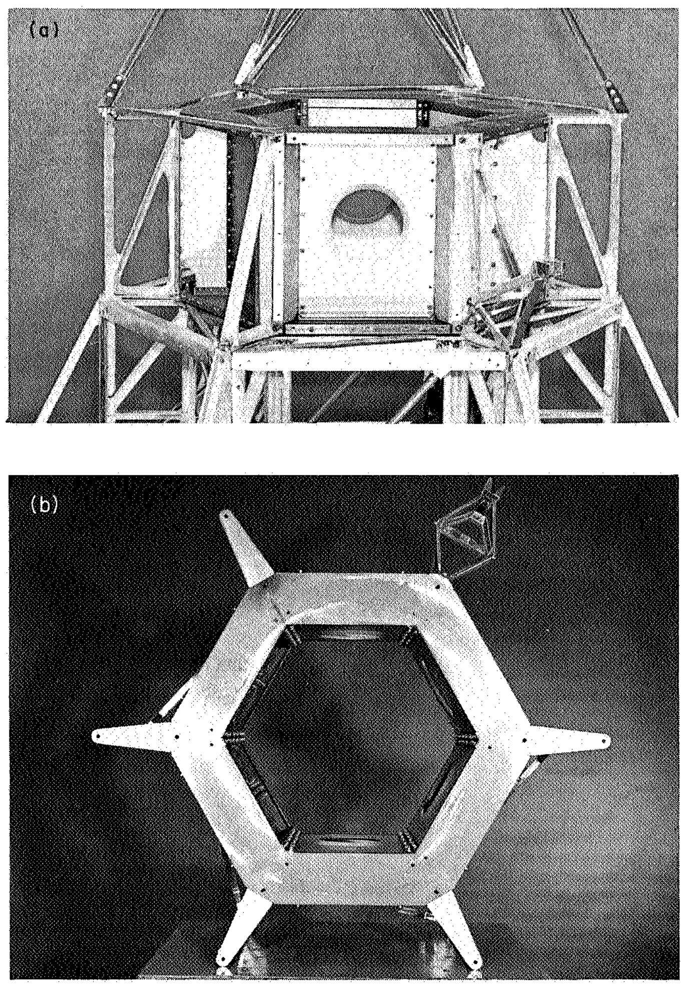

Fig. 3. *Mariner A* Venus–*Centaur* structure

became a shear web and, therefore, part of the primary spacecraft structure. This design was chosen because it offered near-minimum structural weight, yet represented a minimum departure from existing JPL technology.

The concept of not departing radically from existing JPL technology was desirable from the standpoint that it would present fewer unexpected problems to the people responsible for the job. Since the structural approach selected evolved from a broad base of JPL experience in missile and spacecraft projects, high confidence existed that it could be successfully implemented in the time the schedule allowed.

Although this structural approach was based upon past JPL experiences, analytical comparisons of the specific electronic volume (electronic volume ÷ structural weight) realized by this approach was 100% better than that achieved on the *Mariner II* spacecraft.

2. Secondary-structure design approaches. The design of the secondary structure was not initiated until the primary structure design approach was firm. Initially, the secondary structure was required to support two cold-gas nitrogen supplies, the articulating science platform structure and instruments, and a low-temperature cryostat for one of the platform instruments. Near the end of preliminary design, the instrument requiring the cryostat was replaced with a lighter one because the spacecraft was overweight. To increase the amount of payload weight available from the change, the secondary structure was redesigned to carry the lighter load. The final design of the secondary structure consisted of a rectangular framework that supported the two attitude-control-gas bottles and of the necessary structure to support the planetary science platform and instruments.

3. Superstructure design approaches. The spacecraft configuration selected required a superstructure to support (1) the fixed high-gain antenna at approximately 18 in. above the top of the octagon structure, (2) the solar panels during boost, (3) the low-gain antenna, (4) the cosmic dust detector, and (5) the upper thermal shield. The following superstructure designs were investigated to satisfy these requirements.

a. Conical support. A lightweight fiberglass or aluminum honeycomb conical truss between the top of the octagon structure and base of the antenna was the first considered. This approach was abandoned because it was found that an enclosed structure of this type limited the accessibility to the interior of the spacecraft and would be highly susceptible to damage during test and prelaunch operations.

b. Machined and riveted truss. A conventionally machined high gain antenna support using riveted truss tubes was studied, but was rejected because of the complex geometric constraints imposed by other elements, antennas and dampers, that had previously frozen their interfaces with the superstructure.

c. Welded superstructure assembly. After numerous studies, a welded truss and ring design was chosen. This design sacrificed the direct load paths from the dampers and antenna into the main truss structural members. However, it was felt that this structural inefficiency could be tolerated because of the design and schedule benefits that would be available by not modifying the existing interface requirements.

C. Design Process

Once the basic configuration of the spacecraft structure was established, detail design commenced on the primary structure, secondary structure, and superstructure. As the engineering layouts were generated, equivalent static loads for the individual structural components were estimated, based on computer analysis of the structural configuration.

The design load for the basic structure was based on the requirement for no general yielding of the structure under the maximum expected (limit) load.

The most severe of the following design conditions was used to determine the strength of each of the *Mariner* structural elements.

(1) The most severe combination of the maximum flight static accelerations (7 g axial, 1 g lateral) and unidirectional maximum flight vibratory accelerations (1 g rms axial, 0.5 g rms lateral).

(2) The unidirectional maximum flight vibratory acceleration multiplied by 1.6, in a 1-g field without additional superimposed static loads.

(3) The above limit load conditions were restricted such that the shear force at the spacecraft separation plane did not exceed 4 times the spacecraft weight and the limit load axial force would not exceed 12 times the spacecraft weight.

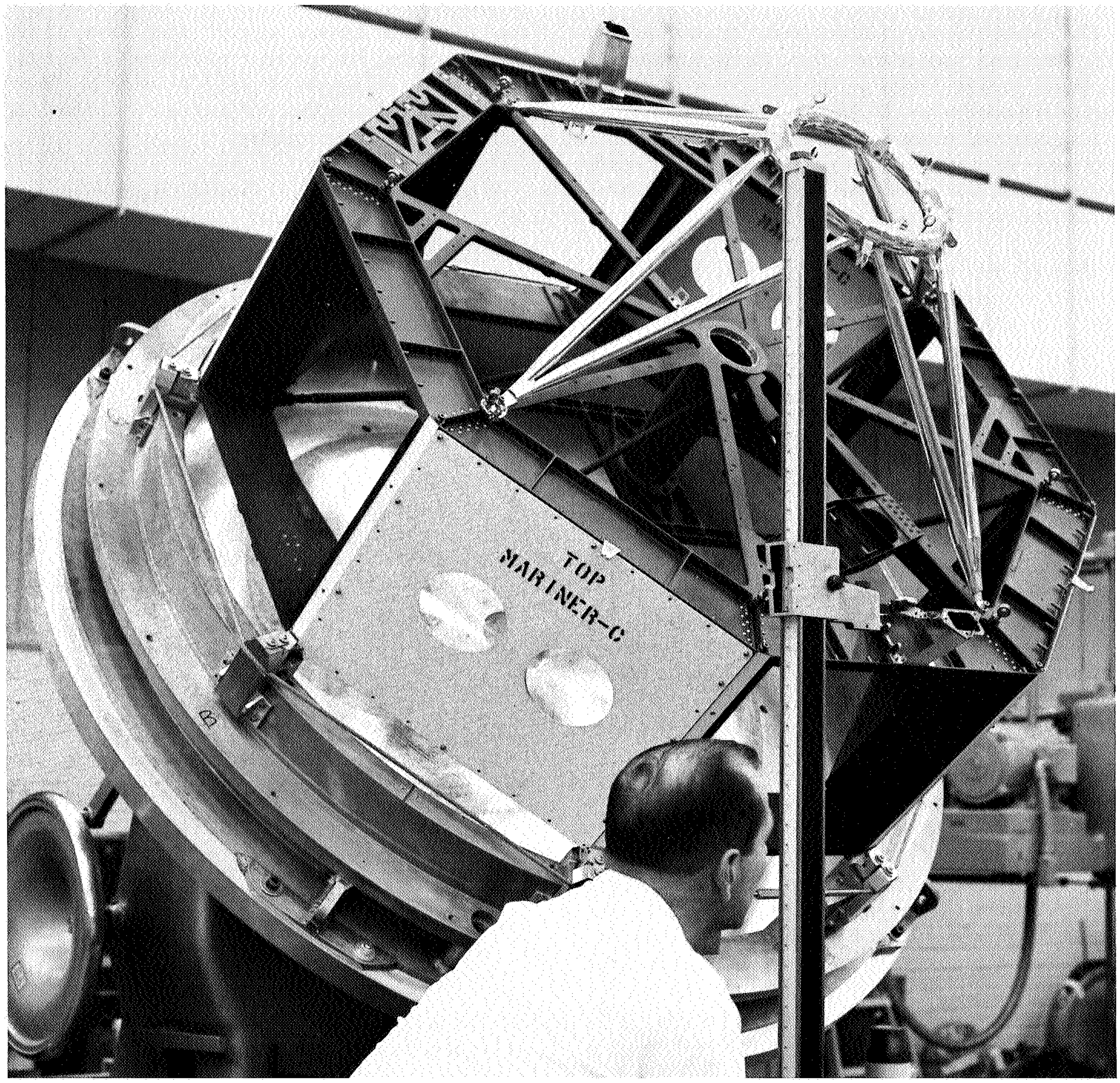

Fig. 4. Basic octagon structure (with superstructure attached)

(4) The radial preload compression force exerted on the spacecraft support points by the V-band attachment mechanism were superimposed on the above loads.

Ultimate loads were 25% higher than design loads. Qualification tests were run at both design and ultimate levels.

Structural sizing of components was established from these loads; then, the stiffness characteristics of the complete structural assembly were determined. The results of this analysis were then fed back into a composite dynamic analysis permitting redefinition of component loads. This process was repeated, in an iterative fashion, until an adequate composite vehicle structure detail design was developed that had the required stiffness and minimized vibrational interactions between components and subsystems. By the time the detail drawings were complete, sufficient structural analysis had been conducted on the individual piece parts to give high assurance that the composite structural design was adequate.

D. Detail Design Description

1. Primary structure. The primary portion of the basic octagon structure (Fig. 4), was composed of two machined octagonal rings, each approximately ¾-in. thick and 55 in. in diameter. Longerons, 16½-in. high (Fig. 5) were mounted at the octagon corners and separated the two rings. The electronic chassis (Fig. 6) containing the electronic subassemblies were mounted into the rectangular openings formed by the rings and longerons (Fig. 7). The electronic chassis in six bays, consisting of a flat, shear web outer surface and T- and box-sections for mounting of subchassis, attached to the primary structure as shown in Fig. 8.

The seventh bay housed a portion of the power-conversion equipment and battery. The power conversion equipment chassis was bolted to the outside surface of the octagon structure in the same way as the standard chassis shear web. The battery was mounted to the inner vertical faces of the longerons and to the center of the upper and lower rings in the same manner as the inboard portion of the standard chassis.

The eighth bay housed the post-injection propulsion system. The system was attached to the primary structure by fittings located in the four corners of the bay. A shear web bolted over the opening stiffened the bay, acted as a thermal shield, and protected the motor and jet vane actuators during test and prelaunch operations.

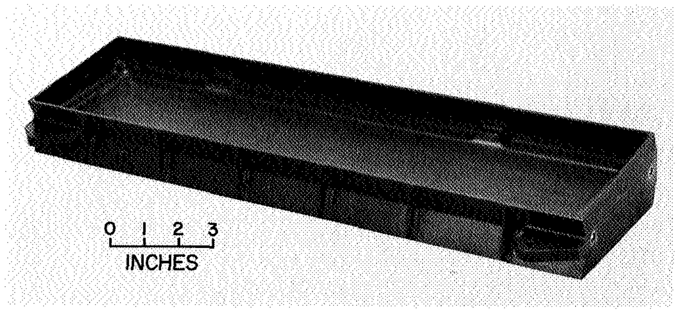

Fig. 5. Octagon longerons

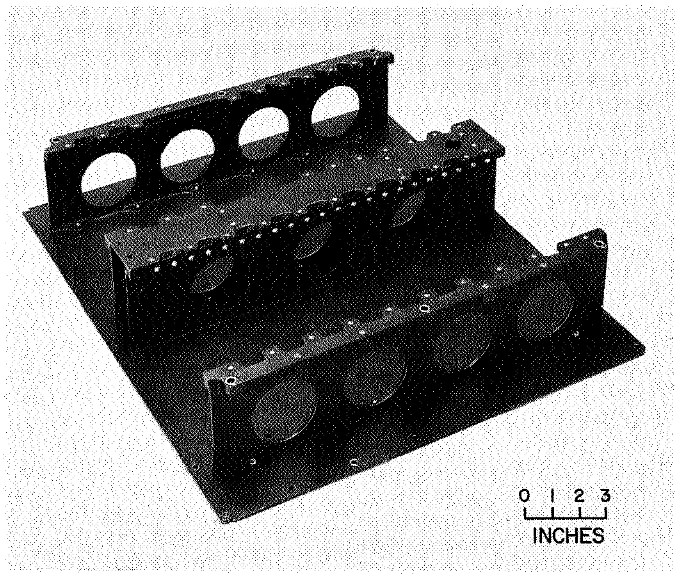

Fig. 6. Basic electronic chassis

The lower ring, Fig. 9, provided mounting surfaces for switches, adapter hardware, a sun shutter, and optical sensors. The sun sensors and Canopus tracker were bolted to mounting and indexing surfaces that were machined directly on the lower ring. The ring was composed of a channel cross section with stiffening gussets provided at the eight corners.

The spacecraft was attached to the *Agena* adapter at the eight exterior corners of the lower ring. It was held in place by a V-band that clamped eight 1-in.-wide feet (Fig. 10) to bear on a corresponding surface on the adapter. The compressive force of the tensioned V-band reacting on the upper surface of the foot, reacted the axial loads. Radial loads were reacted by a shear lip on the lower ring, which bore against a corresponding surface on the adapter. Because of the natural dimensional variation between adapter and spacecraft, the adapter

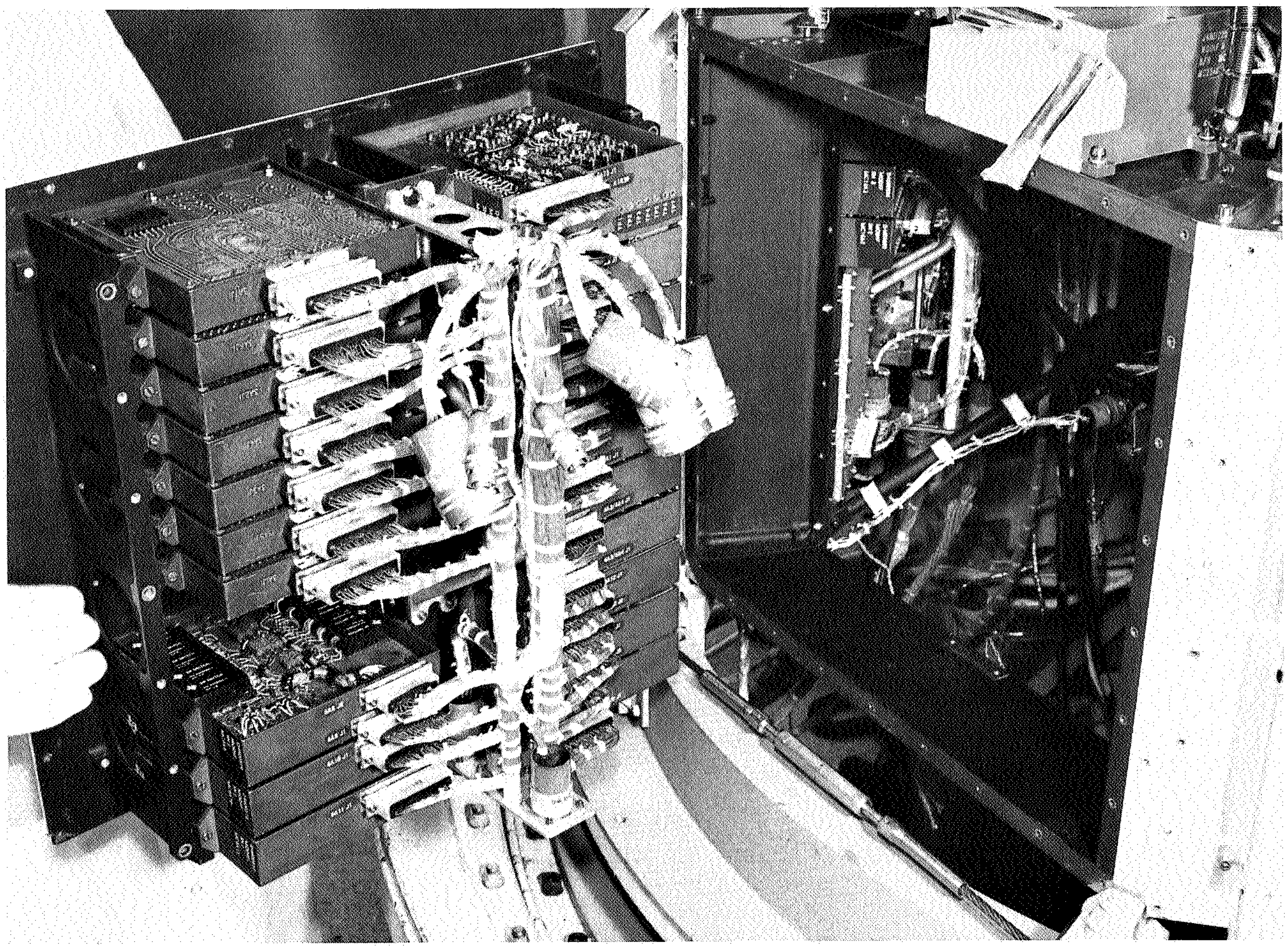

Fig. 7. Electronic assembly being installed into spacecraft structure

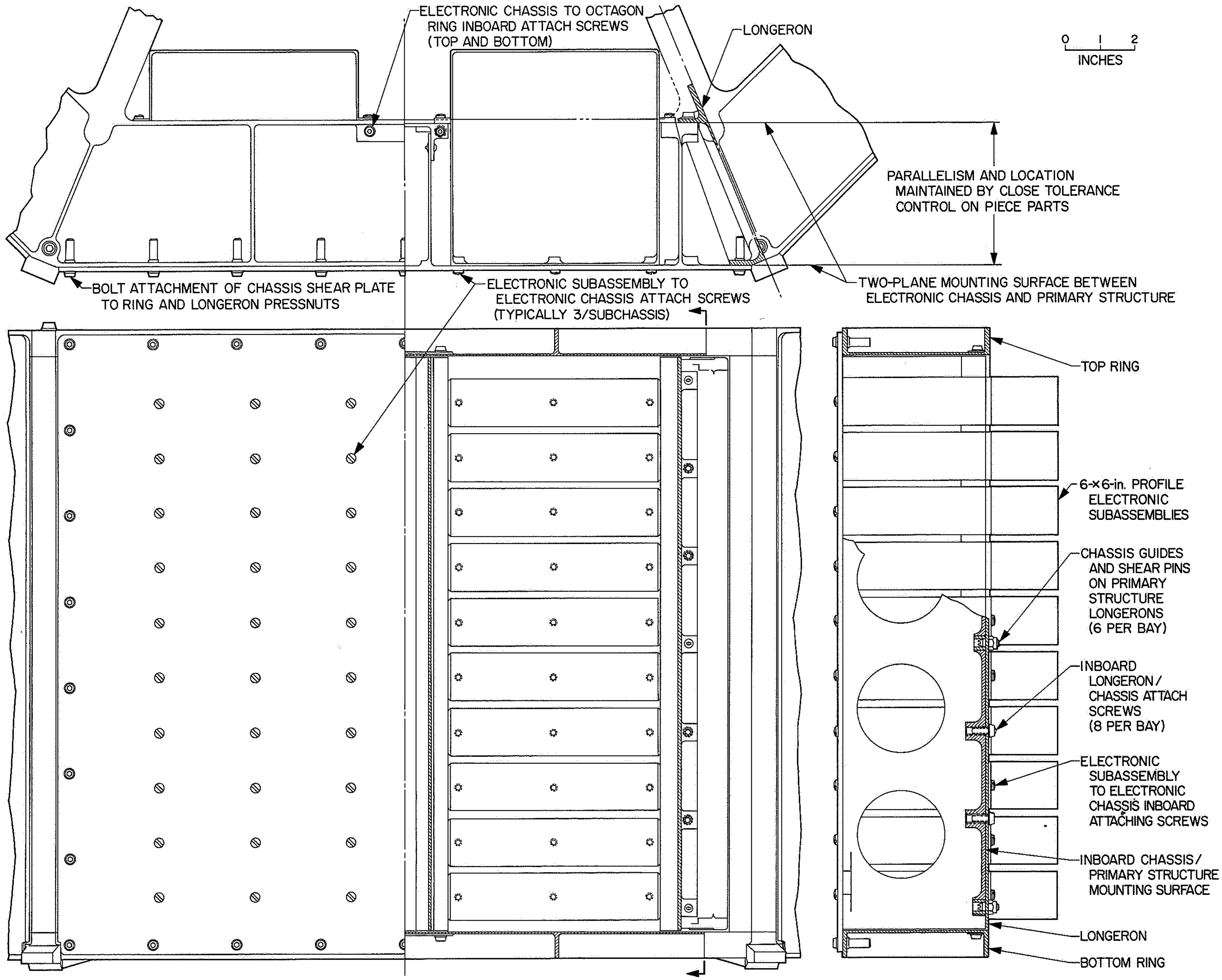

Fig. 8. Typical electronic chassis/octagon structure installation

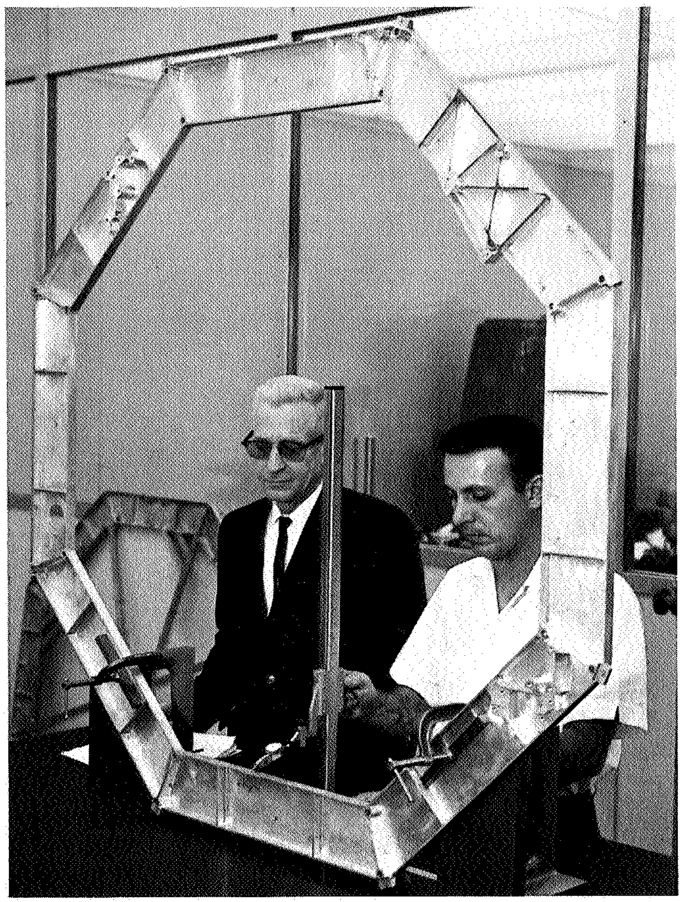

Fig. 9. Primary structure lower ring

was designed to have a radial-spring constant lower than that of the primary structure. The flexibility of the adapter allowed it to conform to the spacecraft dimensions when the two were mated.

The spacecraft adapter mounting surface was designed to be coplanar within 0.005 in. The radial tolerances between the shear lips were held to 0.010 in. across the 55-in. diameter. During the assembly, test operations, and shipment, the spacecraft was attached to a support structure called a *universal ring* (U-ring). This ring supported the spacecraft as it would be supported when on the launch vehicle, with the exception that it was much more rigid than the flight adapter.

The primary-structure top ring contained mounting and indexing surfaces for the sun sensor pedestals, and brackets for mounting miscellaneous hardware. The solar panels were mounted to rod-end monoballs at each of the inner corners of the top ring. These rod ends also provided part of the attachment of the top rings to the longerons. The superstructure was mounted at four corners of the inboard portion of the top ring. The structural cross section of the periphery of the top ring was about the same as that of the lower ring.

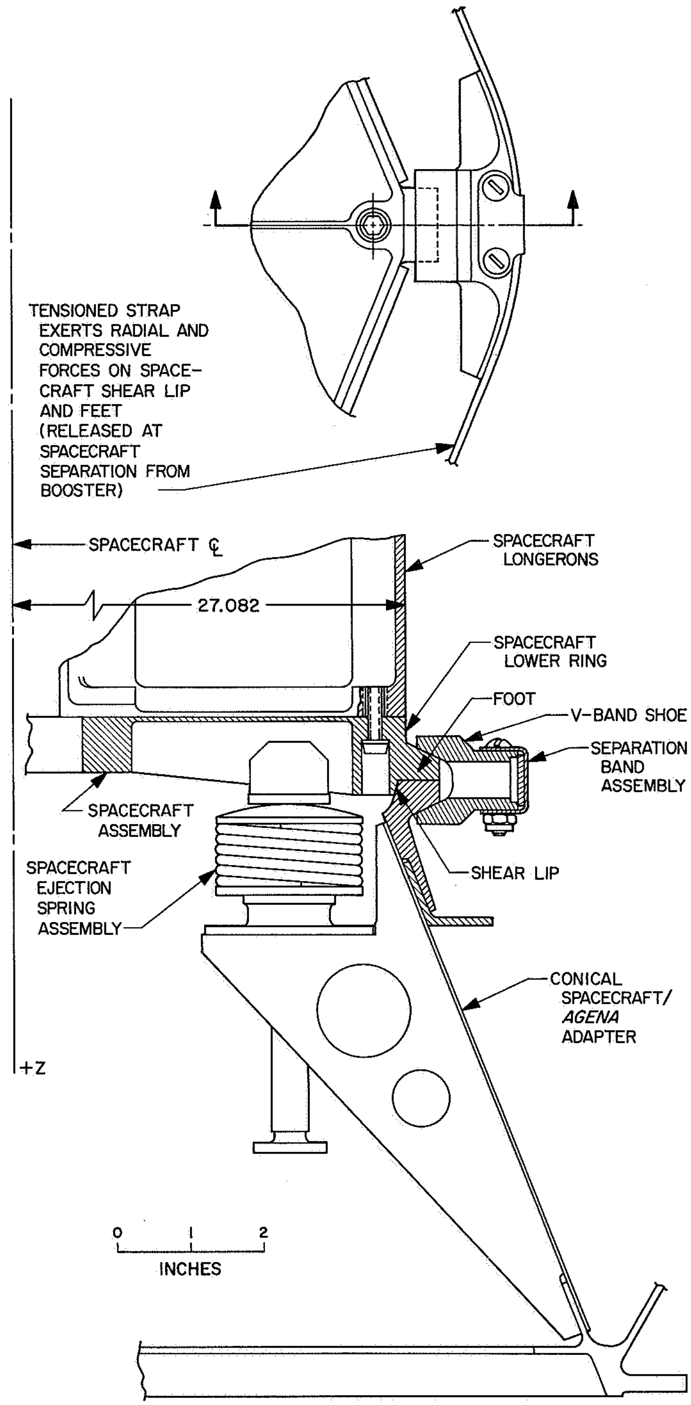

Fig. 10. Attachment detail, spacecraft to *Agena* adapter

2. *Secondary structure.* The upper ring of the primary structure contained eight T-section spokes that extended

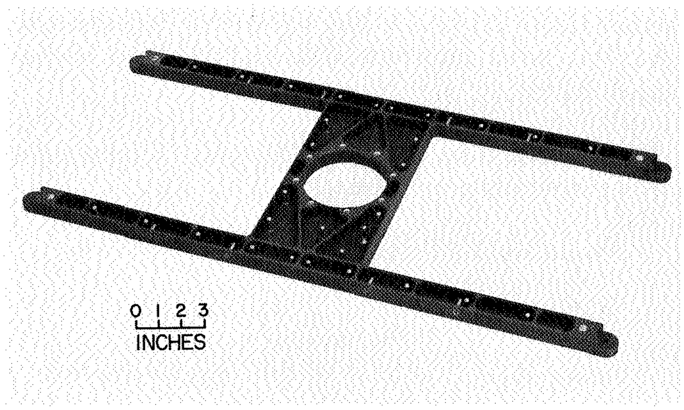

Fig. 11. H-frame structure

inboard to a central hub. This spoke arrangement was the upper part of the secondary structure. The secondary structure also included a central tube, which was mounted to the upper ring hub and the H-frame structure (Fig. 11). This tube contained bearings on each end, which supported the cantilevered science platform structure[2] and instruments. The science platform structure was rotationally restrained during boost by a pyrotechnic pinpuller, which was mounted on the H-frame structure. The two attitude-control-gas bottle support plates were mounted to the H-frame. Each plate was bolted on three sides to the H-frame. The fourth side of the plate included a flange that provided the necessary stiffness and mounted the gas regulator. With the attitude-control gas systems installed, the H-frame effectively became a rectangular shear web with a hole in the middle for the central torque tube. Twelve tubes and conventional fittings attached the corners of the H-frame structure to the top ring and to the primary structure. These tubes provided the necessary lateral and torsional stiffness to the gas-bottle mounts and to the science platform.

[2]Coyle, G., *Mariner IV Science Platform Structure and Actuator Design, Development and Flight Performance,* TR 32-832, Jet Propulsion Laboratory, Pasadena, California, November 15, 1965.

The primary and secondary structure, not including electronic chassis, weighed 30 lb.

3. Superstructure. The superstructure was an eight-membered, welded, aluminum truss, which attached at four points to the primary structure (Fig. 12) and culminated in a circular top ring which contained attach points for the high-gain antenna, one low-gain antenna

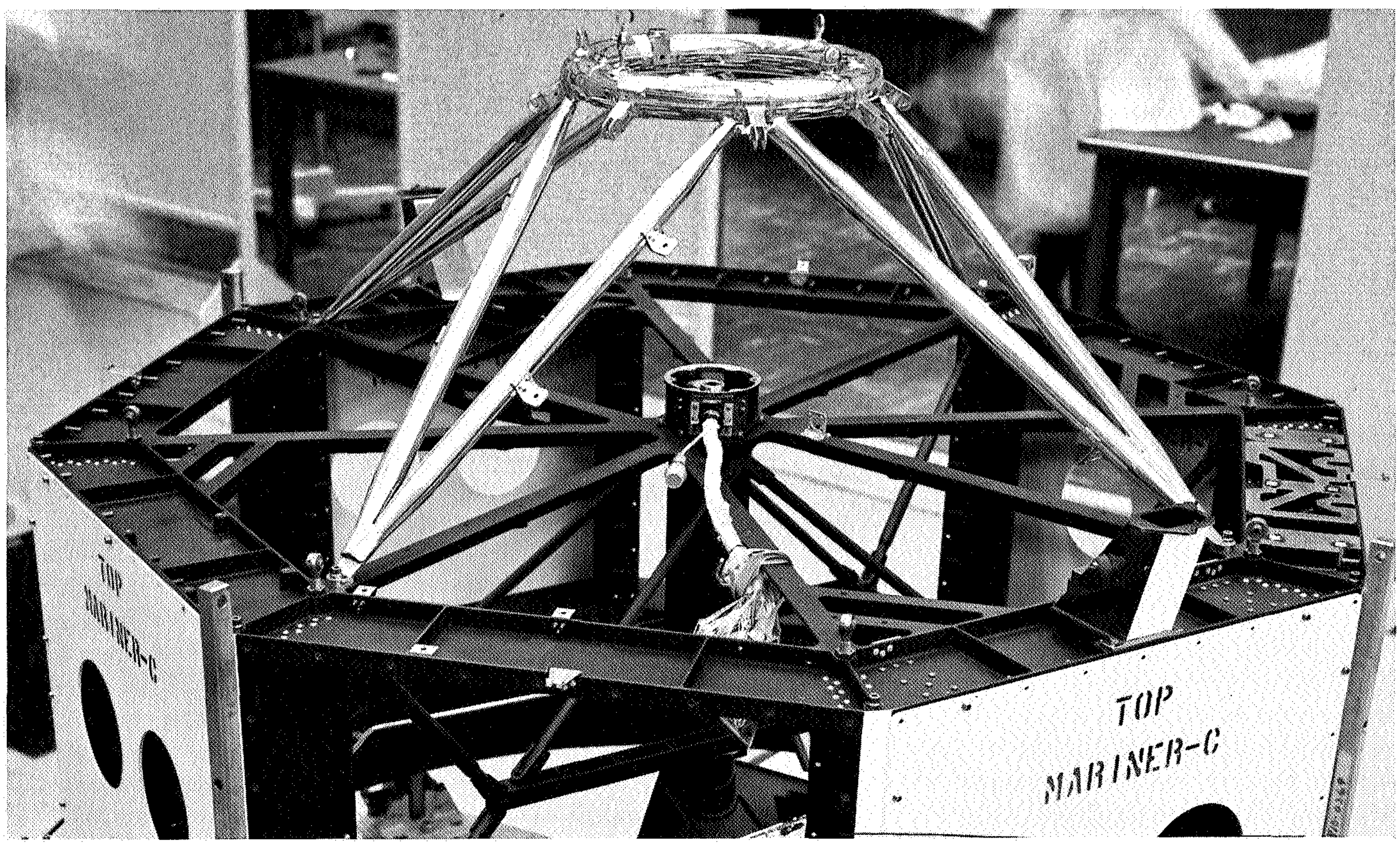

Fig. 12. Superstructure

damper, and the solar panel boost dampers. The truss was 44 in. in diameter, 17-in. high, and weighed 3.06 lb.

4. Miscellaneous hardware mounting provisions. In addition to the items previously discussed that attached to the structure, mounting and attach points for the electronic cabling and thermal shields were provided.

The main cable harness was mounted in a cable trough structure which bolted to the upper ring spokes. Mounting holes for attaching cables were required in most of the basic structural members. The cable mounting provisions, for cable clamps or string ties, were established on the cabling mockup early in the spacecraft design.

The thermal shield development lagged the rest of the spacecraft. As a result, most of the thermal shield hardware had to be *added on.* In most cases the thermal shield attach points took advantage of existing mounting provisions. The side thermal shields, overlapping the octagon corners, were attached to the two bolt-holes in the longerons provided for lifting the spacecraft. The upper and lower thermal shields were attached around their periphery to the upper and lower sets of the electronic chassis mounting screws. The upper thermal-shield blanket rested on the superstructure. New structure was required to support the lower shield. The lower thermal-shield blanket was held away from the lower ring harness by a spring-loaded tube which ran around the inside of the primary structure. The center portion of this blanket was attached to a special shield-support structure which bolted to the secondary-structure H-frame.

E. Fabrication and Assembly

To satisfy test and flight requirements, eight spacecraft structures were fabricated. Four of these units were used for structural, thermal, booster interface, and other mechanical tests; one for the proof test model (PTM), two for the flight spacecraft, *Mariner III* and *Mariner IV*; and a flight spare.

1. Primary and secondary structure.

a. Material selection. The material selected for the top and bottom rings, longerons, and H-frame was ZK60-T5 magnesium in the forged condition. Magnesium was selected because: (1) it was non-magnetic, (2) larger tolerances for a given allowable weight variation could be tolerated, (3) it permitted less weight for the same size fillet radii or for nonstructural bosses, and (4) it was easy to machine. The ZK60 alloy was chosen over other magnesium alloys because it had a relatively high-strength, ductility, good grain-flow characteristics, and dimensional stability.

The forgings were required to have the following minimum guaranteed properties:

$S_{\text{compressive yield}} = 20 \times 10^3$ psi, short transverse

$S_{\text{tensile yield}} = 26 \times 10^3$ psi, radial or longitudinal

$S_{\text{tensile yield}} = 19 \times 10^3$ psi, circumferential or long transverse

$S_{\text{tensile ultimate}} = 42 \times 10^3$ psi, radial or longitudinal

$S_{\text{tensile ultimate}} = 37 \times 10^3$ psi, circumferential or long transverse

After the cast ingots were procured, they were tested for physical properties and chemical composition before the forging process began.

The rings were machined from forgings which were 59-in. diameter by 1¼-in. thick (upper ring), and 1⅞-in. thick (lower ring). The forgings for the H-frame were 24 by 19 by 1 in., and the longeron forgings were 17 by 5 by 2½ in.

The materials were forged at temperatures of approximately 675°F, had a minimum of 10% cold working and were solution heat-treated at 300°F ±10°F for 24 hr.

After the forging operations, the billets were chemically acid-etched to detect surface cracks and to preserve the billets during ultrasonic inspection. The billets were ultrasonically inspected to MIL-STD-271, Class A, and guaranteed to Class B. After ultrasonic inspection, coupons were removed from each forging and tested by the forging contractor.

After arrival at JPL, the billets were Blanchard-ground, stress-relieved for 4 hr at 300°F and subjected again to ultrasonic and fluorescent penetrant inspections. The above quality-control testing procedures were utilized (1) to ensure that the material had the required mechanical properties and (2) to avoid scrapping of parts during subsequent machining operations due to discovery of material flaws. The machined forgings represented a significant investment in money and irrecoverable schedule time that was most important to the timely success of the project.

The other parts of the primary and secondary structures, mostly 6061-T6 aluminum, were fabricated by standard techniques and used the regular material certification processes.

b. Piece part fabrication. Because of the size and inherent complexity of the rings and longerons, there was much apprehension during the design about machining them as single pieces. Three steps were taken to assure that acceptable parts would be delivered on schedule.

(1) The materials tests minimized scrappage during machining caused by material defects.

(2) Design of the upper ring, lower ring, and H-frame structural components was such that the machining could be accomplished with standard straight plunge milling and simple lathe operations. The large upper and lower rings presented a problem of clamping and handling during machining because of their flexibility and thin web sections (the webs were nominally 0.040-in. thick). The longerons were not as straightforward in their design, but they did not present any unique fabrication problems.

(3) The fabricator's use of a complete set of tooling during the machining operations successfully eliminated most of the potential operator error. Also, stationing source inspectors at the vendor assured that proper procedures were being followed.

After the piece parts were inspected, they were given a Dow 7 conversion-coating treatment.

The rest of the piece parts for the primary and secondary structure, aluminum strut tubes and end fittings, were fabricated by conventional means.

c. Assembly sequence. A minimum of tooling was used to assemble the primary and secondary structures. Time was gained during the assembly period by building the spacecraft structure on the handling U-ring, which was designed to be a dimensionally accurate positioner for the spacecraft-adapter attach-feet. The lower ring was mounted to the U-ring and restrained by the V-band shoes. After dimensional checks were performed to verify that the lower ring was restrained properly, the eight longerons, which were machined in matched sets to obtain a consistent overall height, were bolted to the bottom ring. The top ring was then attached to the longerons. The central torque tube was riveted at the top to the center hub of the top ring and at the base to the H-frame structure. Finally, with the H-frame in place, the 12 tubes and end fittings restraining the H-frame were located, pilot holes drilled, and the parts bolted and riveted in place. The fittings were specially designed to allow the blind rivet driving tool to pull the rivets in an optimum manner.

Miscellaneous brackets and pads were bolted, riveted, or bonded in place to the top and bottom rings. Small templates were used to locate these items.

Matched drill jigs were used to drill close-tolerance holes for the electronic chassis, battery, midcourse motor shear web, and the gas bottle equipment plates. All of these items were mounted to the structure with female fasteners called Pressnuts,[3] which were successfully used in place of standard nut plates. In addition to Pressnuts being used because they are blind fasteners, there were various other advantages in their use: they are rapidly and easily installed, are nonmagnetic, are capped to prevent thread chips from falling out during the many hardware installation and removal operations, and they present steel sidewalls to the male fasteners. This latter feature prevents the bolt threads from tapping the softer magnesium material and generating chips or loading up the screw threads – problems common with nut plates mounted on the back side of a mounting surface. The Pressnuts also had the advantage of being easily replaced. They could be drilled out and speedily replaced with other Pressnuts, which could be bonded in place.

Titanium bolts were used to assemble the structure and to mount all spacecraft structural components. The use of titanium saved about 3 lb; however, it was found early in the program that such weight saving was not without problems. The softer titanium, although it had a high yield strength, galled when threaded into deformed-thread locking nuts. The standard locking nuts were deformed to provide the proper locking torque with hardened steel screws. New nuts that were deformed less, but still provided the necessary locking torque, were procured for use with the titanium screws.

The philosophy of machining critical alignments into the piece parts shortened the spacecraft-assembly period, by minimizing both the tooling requirements and the post-assembly machining operations. The use of Pressnuts also shortened the assembly period. Where it had taken

[3]Manufactured by Hi-Shear Corp., Torrance, Calif.

6 weeks to assemble previous basic spacecraft structures, this structure was fabricated and assembled in $2\frac{1}{2}$ weeks — a significant saving when schedules are tight.

After the structure was assembled, the high-emissivity temperature-control paints were applied to the interior of the spacecraft. Surfaces that could not be painted and had been scratched or damaged during the assembly were touched up with Dow 19 surface treatment.

2. Superstructure: Fabrication and assembly.

(1) *Upper truss ring.* The upper truss ring of the superstructure was formed of two halves of a torus that were machined and welded to form a complete ring assembly. Four types of ring fittings were machined and welded to the ring. These fittings were machined from 6061-T6 aluminum and annealed prior to welding. Each fitting was made with sufficient extra material to allow for machining to the required assembly dimensions after welding to the ring.

The ring weldment and fittings were tack-welded in a fixture (Fig. 13) in a heated environment (hot plate and photo floodlights). After tacking, the assembly was removed from the fixture and finish-welded in the heated environment in a sequenced manner to minimize distortion. After welding, all welds were radiographically and fluorescent-penetrant inspected. If unacceptable defects were present, they were removed by grinding, then were radiographically inspected to ensure that the defects had been removed. Once the defects were removed, the parts were rewelded and reinspected. This cycle was iterated until the parts either were free of rejectable defects (cracks, chain and sharp porosity, inclusions, non-fusion, and poor penetration) or were scrapped. After heat treating the radiographic and dye-penetrant inspections were performed again. If the parts were not acceptable, the above described process was repeated. After heat treating, the truss weldment was chemically milled to reduce the tube wall thickness from 0.058 in. to 0.040 $\pm$0.005 in. This process removed approximately 1 lb from the ring. The part was then machined, inspected, and polished prior to the next assembly operation.

(2) *Truss strut tubes.* The truss strut tubes were formed of 1-in. diameter 6061-0 aluminum tubing with 0.035-in.-thick wall. The ends of the tubes were swaged to 0.59 in. to minimize the size requirements for the end fittings. After swaging and treating to the T6 condition, the ID of the swaged ends was honed to remove inclusions and imperfections caused by the forming process. The tubes were then fluorescent-penetrant inspected and machined. After the tubes were polished and inspected, they were ready for final assembly.

(3) *Final assembly.* Final assembly welding took place in the welding fixture shown in Fig. 14. The truss

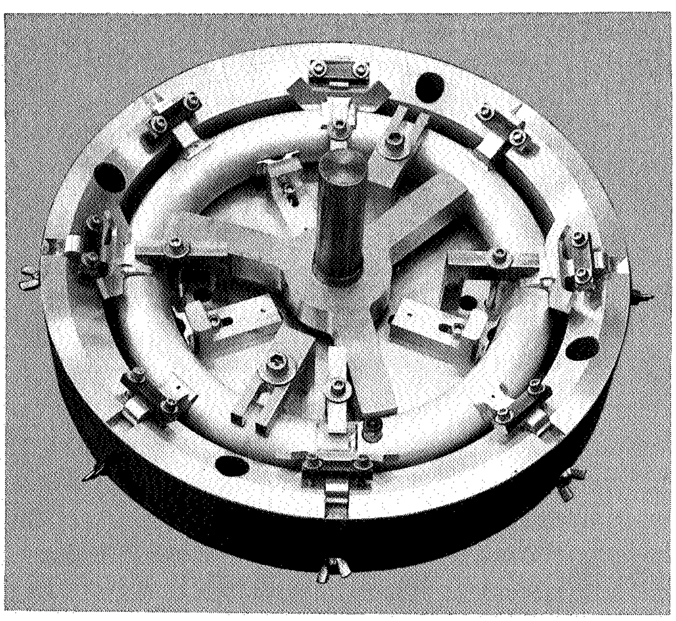

Fig. 13. Superstructure ring welding fixture

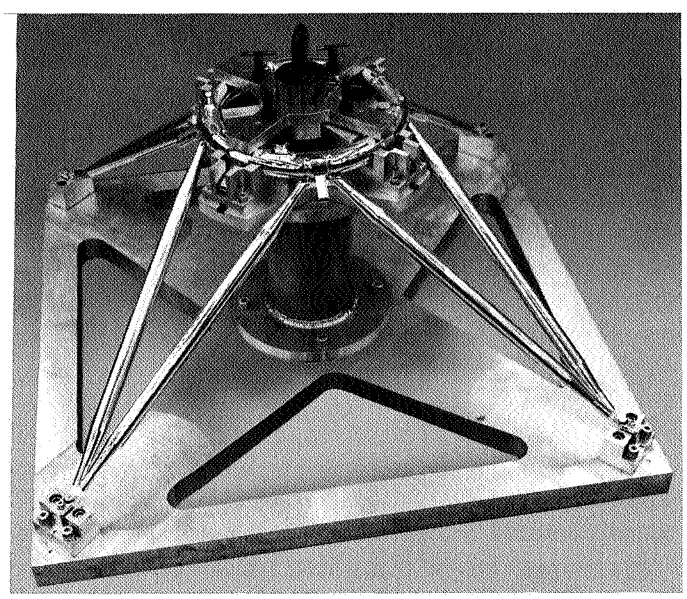

Fig. 14. Superstructure in welding fixture

ring weldment, truss strut tubes, and four feet were tack-welded together in the fixture and then finished welded in the free state. The welds were then radiographically and fluorescent-penetrant inspected. If rework were required, it was performed in the same manner as that described for the truss-ring weldment. When all welds were acceptable, the assembly was replaced in the welding fixture and stress relieved. After stress relieving, the assembly was placed in another fixture similar to that used for welding and machined to final dimensions. Following final machining, brackets were riveted on, Pressnuts installed, polished surfaces touched up and the assembly inspected.

3. Assembly and inspection of complete basic spacecraft structure. The primary-and-secondary-structure assembly and the superstructure were assembled, and a final inspection was performed. Special handling shear webs were installed over the bay openings in the primary structure to assure that its structural integrity would be maintained. In theory, rotational displacement of the rings relative to one another could cause the structure to collapse like a house of cards. However, in practice, this structure was capable of reacting to handling loads without permanent distortion. The purpose of the inspection was to verify that all of the assembled spacecraft dimensions, tolerances, and alignments which were designed into the piece parts had been met. These alignments were recorded for future use and comparison. Inspections prior to launch confirmed that the mechanical alignments originally built into the structure had been retained during the tests and operations phase of the program.

After the units had been inspected and certified, either they were delivered to the spacecraft assembly facility where flight spacecraft assembly and test operations began, or they were delivered for use in the special developmental, type approval, and interface tests.

F. Testing

The basic spacecraft structures were subjected to a complete series of structural qualification and special developmental tests during the design period. These included a static test on the basic structure while it was mounted on a booster adapter. This test verified the stiffness of the spacecraft and adapter structures and checked the adequacy of the spacecraft-booster interface. Vibration tests were conducted to type approval and ultimate levels on a complete spacecraft structure containing dynamic mockups of all spacecraft components, including cabling. Except for a few bracket failures, the basic portion of the spacecraft structure survived these tests without modification. Vibration testing to design levels was done on the PTM and to flight-acceptance levels on the flight spacecraft.

IV. Problem Areas

A. Actual Problems Solved

Although the schedule and performance criteria were met, the design and fabrication of the spacecraft structure had its share of problems. Some of the typical problems and solutions encountered during the development cycle that may be of value to future programs are listed below.

1. Inadequate weight-estimating practices. Because the total spacecraft was overweight, there was redefinition of the scientific instrument payload near the completion of the preliminary design phase, causing the secondary structure to be redesigned. Although the changes were incorporated without significantly compromising the final design, they did increase the cost and delayed the development structure deliveries for tests. A process for accurate and complete estimating of the weights of all spacecraft components early in the design process should reduce these occurrences.

2. Maintenance of machined-ring tolerances. There was some difficulty in maintaining the flatness requirement on some of the first upper rings that were machined; they had a tendency to oil-can. The first ring fabricated was in tolerance, but the next two rings were oil-canned 0.080 to 0.090 in. The rings were brought into the flatness tolerance by placing them on a large flat plate, loading them over center 0.040 in. and then stress relieving. It was later found that the oil-canning was caused during the machining operations by a combination of excessive tool pressure and dull tools. These problems were corrected and subsequent machined parts were acceptable. This was the only problem that was experienced in machining the magnesium basic structure parts.

3. Magnesium surface treatment damage. Several of the magnesium machinings were damaged during the surface treatment processing. In these instances the acid etch bath removed more material than allowable and several parts had to be scrapped. This problem was corrected by the assignment of quality assurance inspectors to the vendor's facility to ensure that the proper treating procedure was used.

4. Tube dimensional control during forming. In the superstructure fabrication, one of the major problems was the dimensional control during the tube-forming process. Problems encountered in forming upper-ring half sections included controlling the inside and outside diameters of the formed rings to the desired tolerances. Parts that were marginally acceptable were clamped in a fixture of the proper dimensions, heated to 450°F and allowed to cool to bring them into tolerance. This procedure was repeated until the parts were acceptable.

The problem in fabricating the strut tubes was swaging the tube ends without developing cracks in the internal surface of the reduced section. An acceptable two-step process was developed, which consisted of swaging the ends of the 1-in.-diam tubes to ¾ in., using 6061-0 Al material and solution heat-treating the tubes in the unstable SW condition by packing them in dry ice. The tube ends were then swaged to the final 0.59-in. diameter and aged to the T6 condition. Final honing of the tube ID removed any minor imperfections resulting from the forming process.

5. Acceptable weld quality of finished parts. Obtaining finished parts of acceptable weld quality was the most persistent superstructure problem. The welding of thin-gauge aluminum structures is primarily an art requiring considerable time to develop the proper welding techniques and procedures. Because of the limited time and small quantity of items to be produced, this problem was attacked mainly by a cut-and-try process based on fabrication and engineering experience. Once a procedure that produced nearly acceptable welds was evolved, it was followed religiously and reiterated until the part was acceptable or rendered useless by producing more defects. The following listed techniques were developed:

(1) Using the same welding personnel to fabricate all parts

(2) Using a shielded gas with a low dew point and checking the dew point regularly at the torch

(3) Preheating the parts in a controlled and repeatable fashion

(4) Cleaning and maintaining clean parts, particularly in the areas to be welded

(5) Rigidly controlling the welding sequence

This technique, although seldom producing an acceptable part on the first iteration, did produce remarkably good parts, considering the complexity of the joints involved. Generally, the parts had to be reworked four or five times before they were acceptable.

6. Use of titanium bolts. Although the change from A286 bolts to titanium bolts appeared to be straightforward, a number of problems arose. The softer titanium screws galled when they were threaded into standard MS deformed-thread locking nuts. These nuts had been deformed to provide the proper locking torque with hardened steel screws. New nuts, built to MS standards but with less deformation, were used. This modified fastener assembly provided the required locking torque.

7. Incomplete dimensional inspection. After one of the basic structures was assembled, it was discovered that the angle on the spacecraft-booster adapter shear lip was out of tolerance. This error had been missed during the inspection of the lower ring. Rather than scrap the part or remove the bottom ring from the structure assembly, the complete structure was installed on the milling machine, and the shear lip was machined to tolerance. Although this was a relatively hazardous operation to the assembled structure, no alternative existed. This was a painful lesson about the importance of complete dimensional inspections. Had an error been made during the corrective machining, the structure could have been lost.

8. Bracket failure. During the spacecraft qualification vibration testing the low-gain antenna-support bracket failed. This bracket spanned two spokes of the upper ring and reacted the low-gain antenna axial loads. During the axial vibration testing, the bracket flanges failed in a local highly stressed area. The geometry restrictions meant that gross changes to the bracket design could not be made; therefore, special flanged doublers were used.

B. Other Possible Problem Areas

The foregoing discussion records some of the actual development problems with the basic structure. The following discussion concerns some problems that were anticipated but never developed.

1. Electronic chassis/spacecraft attachment. During the design phase there was concern about the problems that may be involved in physically attaching the electronic chassis to the basic spacecraft structure. Specifically, the problem was being able to get the chassis to mate to the structure at two parallel planes, one the outer shear web plane and the other the inboard longeron mounting surface. Also, the chassis had to bolt to the upper and lower rings. An associated concern was whether or not the electronic chassis could engage the six shear pins mounted on the inboard web of the longerons.

Prior to the release of the detailed drawings, dimensional checks were done to verify that the tolerance buildups and dimensioning techniques would ensure that the electronic chassis would mate with the spacecraft. Special, matched tooling plates were fabricated to drill both the electronic-chassis mounting holes and the attach-holes in the primary structure. Ground-handling frames were built and used to position the electronic chassis shear pins whenever subassemblies were installed in the chassis prior to installation in the spacecraft. All the preassembly handling of the electronic chassis was done in this handling frame.

The electronic chassis were designed to allow for a certain amount of mismatch, if the inboard and outboard faces of the structure were not parallel. The exterior chassis shear-web structure-attach area could deflect slightly out of plane and, thus, provide a tight fit against the outside surface of the octagon structure. The attachments between the top and bottom rings and the chassis center support were designed to allow the web of the top and bottom rings to slightly deflect to compensate for any non-nominal tolerances between the center chassis support and the top and bottom rings.

The result of these efforts was the fortunate absence of problems in mating the electronic chassis to the structure during the test phase of the operations.

2. Machining mechanical alignments into structure. The philosophy of machining the mechanical alignments into the structure, rather than aligning items at final assembly, raised questions as to whether it could be done without significantly increasing the fabrication costs or jeopardizing the delivery schedule. Also, it was questioned if the mechanical alignment, once achieved, could be maintained.

Measurements of critical alignments made during the lives of several test and flight structures verified that the built-in alignments were stable during their ground lives. The successful completion of the midcourse maneuver and the mission tends to verify that the alignments remained stable during flight. The fabrication costs were not significantly higher and the fabrication and assembly time were decreased by the avoidance of shimming and aligning periods during the spacecraft final assembly.

V. Recommendations

Valuable experience was gained during the design and fabrication of the *Mariner* Mars 1964 spacecraft. The basic structural design of the spacecraft satisfactorily met the requirements that were imposed on it. These included: ease of manufacture, satisfaction of test and final assembly considerations, minimum weight, structural efficiency, meeting alignment accuracies, ease of ground handling, and adaptability to changes.

Below are listed some of the more general conclusions and recommendations that may be useful to future spacecraft structural designs; they have been categorized by project phases of (1) design, and (2) fabrication and assembly.

A. Design

1. Design responsibility. The best total design will come about if the responsibility for the design of the structure rests with the project group that is knowledgeable of the total spacecraft; spacecraft mechanical designs are so integrated with the spacecraft system and subsystem functions that a thorough understanding of the considerations influencing the design is a big job in itself. The structure serves to tie all these elements together. However, the project group should be supported by a group of structural specialists who, through detailed analysis and test, ensure that the structure is adequate and that its efficiency has not been unduly compromised.

2. Integrated structures. Integrated spacecraft mechanical designs are workable if potential integration problems are studied thoroughly during the preliminary design phase. The electronic packaging design can be integrated into the spacecraft structural design if early in the program a standard interface is developed between packaging and the structures. The integrated electronic-chassis/primary-structure and the attitude-control pressure-vessel brackets to the H-frame used on *Mariner* Mars 1964 are examples of spacecraft elements being used as structural members.

However, carrying this philosophy too far will cause problems. Accessibility to components during test and assembly will be sacrificed if there is too much integration. Also, the total structure should be divided into structural elements so they can be independently designed and fabricated. This approach will aid in meeting the typically short schedules for design and fabrication. The separation of the superstructure from the octagon-structure is an example of the structural design being broken into discrete areas to meet this criteria.

3. Mechanical alignments. Realistic mechanical alignment criteria should be established early in the design. Other error sources can preclude the need for small tolerances on mechanical alignments such that standard machining and assembly mechanical tolerances can be applied to the structural design without degrading the flight accuracies required. The use of these realistic tolerances will also minimize the need for any adjustments after the spacecraft is assembled.

Checks to verify the adequacy of the structural mechanical alignments should be done after assembly and during the test and operations. If realistic mechanical alignments have been allotted, speedy mechanical checks can be made with standard instruments, avoiding sophisticated optical alignment techniques.

4. New developments. Typical spacecraft development schedules are so short that the solution of special problems should not involve the development of new items or techniques during the program. The advantages promised by a concept that departs from existing technology should be weighed seriously against potential future surprises. If new developments are required, alternate methods of accomplishing the same task should be studied, as a precaution, should unanticipated problems develop. Although the manufacturing technique of the octagonal upper and lower rings was an outgrowth of that used on earlier *Mariner* spacecraft, it approached being a new development. If the large rings could not be manufactured, an alternate method of using smaller piece parts in riveting the ring assemblies could have been used.

5. Adaptability to changes. Since during the typical spacecraft design, there are late changes and additions, it is imperative that flexibility be maintained in the design and fabrication of components to accommodate these changes. In *Mariner*, the change in scientific payload caused extensive revision of the secondary structure; however, the basic structure was readily adapted to the change by the simple addition of brackets and fittings. Also, belated changes and additions in the adapter interface area were easily accommodated by the addition of brackets.

6. Over optimization on a weight basis. Optimization of a basic spacecraft structural design solely to minimize weight can cause problems during a spacecraft development program. Because the structure provides the support for all other spacecraft components from the beginning of spacecraft buildup through the flight, any mishandling damage or failures of basic structural components require that the *complete* structural assembly be replaced. Typically, these assemblies are large and require much time to replace if structural failure occurs. Therefore, a conservative design approach should be used on the basic structure to minimize failures and, thus, reduce repair time. Also, if there are margins allowed, design changes, additions, and errors typically occur in the course of a program and are easily handled. For instance, when dimensions of the spacecraft booster adapter feet were found to be wrong, the shear-lip angles were machined to tolerance with minimum concern about the structural implications. Had these feet been designed with minimum allowable safety margins, the flight structure would have had to be scrapped or schedule time lost in an extensive repair.

7. Attention to detail. No detail of the structural design should be overlooked by the designer. Typically, in spacecraft structures, failures do not occur in the basic structural members. They occur in the piece parts, brackets, etc. The detailed attention paid to the attachment of the electronic assemblies to the basic structure, although a seemingly insignificant potential problem, resulted in a direct attachment method. It was this same attention to detail that resulted in the choice of spacecraft fasteners; the use of female Pressnut fasteners minimized the basic structure assembly time and had the additional benefit of being easily replaceable if a fastener were damaged.

B. Fabrication and Assembly

1. Large machinings. During the design phase there was much concern about the desirability of machining the upper and lower rings out of a magnesium forging. Although some warping was found in the first parts, the rings were satisfactorily machined to tolerances. The successful ring machining is attributed to the extensive use of tooling. Adequate tooling is mandatory in machining shapes of such complexity. Also, the attention directed in the design to simple plunge-milling machining practices simplified the machining operations. Large shapes can be machined to the spacecraft tolerances if the machining is straightforward and adequate tooling is available.

2. Fabrication time. Spacecraft structures must be capable of being fabricated and assembled quickly. Because they interface with practically every subsystem, they are the last items to be completely defined. Because the structure must be available when the initial assembly begins, it is one of the first items required by the spacecraft system.

3. Quality control. The materials used in the large machinings of the spacecraft structure must be constantly inspected prior to, and during, the fabrication cycle to guarantee material properties and quality. The many inspection steps involved in forging and machining the upper and lower rings resulted in all of the parts that were received from the machining vendor being acceptable. Had this attention not been paid to the material quality, large amounts of time and money could have been lost on parts that were found to be unacceptable late in the fabrication sequence.

4. Handling of structures. Spacecraft structures can be susceptible to damage in handling. Precautions must be taken at all times to ensure that the structures are not overloaded or mishandled – this is especially true during the periods when the structure is not completely assembled. Handling damage was avoided on *Mariner* by putting shear webs on the outside of the bays when the electronics chassis were not installed and by stipulating that the basic structure must at all times be supported on the rigid ground-handling support ring.

The requirements of the ground handling and support operations on the spacecraft should be included during the preliminary design of the spacecraft structure. Provisions for lifting the spacecraft, partially or completely assembled, should be considered. Also, the ground-handling considerations imposed by the systems test operations, environmental tests, weight and *cg* determination, shipping, hoisting, and special tests should be factored into the design. In *Mariner*, the spacecraft was designed to be permanently mounted to the universal support ring in the same way as it mated to the booster adapter. All the ground-handling provisions were provided on the U-ring, with a single exception. Hoisting points were provided on the spacecraft to allow it to be moved from the U-ring to the launch vehicle adapter.

5. Welded structures. Although a welded superstructure design was successfully built for the *Mariner*, it is recommended that other approaches be used when a small quantity of parts is needed or when time is short. The fabrication problems encountered required a high expenditure of manpower to ensure that an adequate, qualified structure was delivered. The weight savings provided by a welded superstructure design did not justify the problems inherent in the fabrication of such a structure. In addition, structural failures are difficult to repair. If damage occurred, the complete superstructure assembly would have had to be scrapped and replaced with another. The use of machined and riveted structures is recommended in applications where a small number of assemblies is required.

6. Capped fasteners. The use of capped, nonlocking Pressnuts in female fastener applications is recommended. These fasteners reduce assembly time and are easily replaced if damaged. They also contain any chips or debris that may result from fastener installations. With *Mariner*, they saved time, not only during the assembly of the structure, but during the spacecraft assembly and test operations, because of their long service life.

VI. Conclusion

The structural design and development of the *Mariner* Mars 1964 basic structure was truly an engineering challenge. The requirements placed upon the structure strained the state of the art in a variety of disciplines; they demanded the best in analysis and design techniques, as well as excellent methods of fabrication and quality control. The integration of packaging and structure merged disciplines and innovated a system-approach to structural design. The pressures of limited weight and development time forced concern for detail and equipment safety that is noteworthy. The result was a well designed spacecraft that may serve as model for future planetary projects. The job was challenging and fraught with problems.

Happily, the challenge was met, the problems were solved and the mission was eminently successful.