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S-IVB SATURN HIGH ENERGY UPPER STAGE AND ITS DEVELOPMENT

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ABSTRACT

The development of carrier rockets for manned space missions has been one of the major activities in the aerospace field during the past decade. The early space efforts were made possible by the existence of large ballistic missiles. It soon became obvious that the delivery of weapons and the launch of large spacecraft could not be combined into one operational system in an efficient way; therefore, a family of spacecraft boosters had to be created.

The Saturn System is a significant development of such a carrier rocket for space operations, and will develop, in its final version, the Saturn V, the most powerful carrier at the present time.

The LH_2/LOX technology developed by Douglas for NASA represented by the Saturn S-IV and S-IVB stages could also be applied to smaller LOX/LH_2 vehicles. Larger rockets generally can achieve higher mass fractions due to structural and equipment weight efficiencies. Smaller stages require increased sophistication to achieve similar design mass fraction objectives. The techniques used on S-IV and S-IVB are applicable specifically in the areas of aluminum tankage, ground equipment, pressurization techniques, propellant management, propellant utilization, and instrumentation disciplines. The Saturn S-IV/S-IVB technology will significantly contribute to space flights in the future.

S-IVB SATURN HIGH ENERGY UPPER STAGE AND ITS DEVELOPMENT

INTRODUCTION

The development of carrier rockets for manned space missions has been one of the major activities in the aerospace field during the past decade. The early space efforts were made possible by the existence of large ballistic missiles. It soon became obvious that the delivery of weapons and the launch of large spacecraft could not be combined into one operational system in an efficient way; therefore, a family of spacecraft boosters had to be created.

The United States Saturn system is a significant development of such a carrier rocket for space operations, and will develop, in its final version, the Saturn V, the most powerful carrier of the present time. (Figure 1)

The performance of these systems, in carrying large weights to the moon and planets, is largely determined by the use of rocket engines consuming the high energy propellants, oxygen and hydrogen. These engines have been developed to a high degree of reliability during the last ten years and gradually incorporated into the large miltistage carriers.

Staging of rocket carriers is the most efficient way to transport payloads to the necessary altitudes and then accelerate them to high velocities. In these systems, performance requirements are most critical for the end-stage because of the required accuracy of delivery, and the ratio of burn-out weight to payload. The overall performance of the end-stage has greater influence than the primary stages. The Saturn V launch vehicle for the lunar mission requires 50 pounds of booster weight at liftoff for each pound of payload injected into a translunar trajectory. Without high energy upper stages this factor would be significantly greater.

The Saturn program, so far, has created three distinct vehicle systems: the Saturn I, the Saturn IB, and the Saturn V. (Figure 2)

The Saturn I, a two-stage carrier, consists of the S-I first-stage and the S-IV second-stage. The ten test flights conducted with this version, all of them successful, provided information and experience in the handling and performance of this type of rocket booster. This system, with an orbital payload of 20,000 pounds, fulfilled all expectations and has opened the way for the development of the next generation of large carriers.

The Saturn IB, another two-stage carrier consists of a modified S-I first-stage and the S-IVB, a larger upper stage, (figure 3). The first 3 test flights were successfully conducted in February, July and August 1966, (figure 4). The Saturn IB will not only test the performance and the reliability of the new second-stage the S-IVB, (figure 5), but also test the different modules of Apollo and establish the first orbital flight with the operational Apollo hardware. This carrier will then continue to be used for many scientific and operational experiments and will become one of the workhorses for space exploration.

The Saturn V, a three-stage carrier, will be used as the standard booster for manned lunar and planetary flights as well as for establishing large space stations in their desired orbits. The Saturn V consists of the first-stage, S-IC, with a propulsion system creating 7.5 million pounds of thrust and lifting in the Apollo mission 6,400,000 pounds of weight. The second-stage is the S-II, a multi-engined booster with one-million pounds of thrust which accelerates 1,440,000 pounds, using the same high energy hydrogen fuel as the third and final stage, the S-IVB. (Figure 6)

The development of carrie rockets in the USA has been historically a joint venture of government and the aircraft industry. There seems to be a logical connection between the technologies of aircraft and spacecraft development; but this is only partially true. Aircraft development is probably closer to spacecraft development than any other type, but it is sufficiently different to require special rules and approaches and has, therefore, established a philosophy of its own. Before describing the S-IVB stage in detail it may be desirable to give some information about the development of space systems in general.

SPACE BOOSTER DEVELOPMENT

Space systems usually require a significant research and development (R&D) effort to bring new hardware concepts to the operational phase. The magnitude of this effort, in both resources and schedule, generally is directly proportional to the degree of advancement in the state-of-the-art of the critical technical disciplines. New technologies and new management and control principles have to be developed. For large boost vehicles, major portions of the program resources must be devoted to producing the extensive ground support equipment and facilities, in addition to flight hardware design and development. Mission requirements establish the flight hardware configuration. The facilities and ground support equipment (GSE), in turn, must accommodate the flight hardware configuration without compromise of the basic mission requirements. The hardware configuration is defined only in gross concepts in the initial phase of the program. Since the development of GSE and facilities demand a similar time span as the vehicle itself, requirements for this equipment must be determined with adequate flexibility and capability to satisfy the vehicle through its complete development cycle. Since current carrier rockets are expendable, a comprehensive ground testing program of components and complete systems is required to achieve early flight success. This extensive testing has a major impact in the type and quality of GSE and facilities.

Management techniques are undergoing evolutionary changes to more efficiently accomplish this system development phase. Emphasis today is shifting to finalize the development program planning and documentation to firmly establish the system requirements prior to hardware fabrication. Major program phases are:

Phase A - Conceptual/Feasibility

Phase B - Preliminary Definition

Phase C - Development

Phase D - Operations.

Formal review by both the Government and contractor managers is required at the completion of each phase. Before final commitment to production hardware, it is then possible to define the development program with the good assurance of producing the required operational system within the available resources and on time.

Development programs can be divided into several functional areas: design, manufacturing, test, and flight. The United States aerospace industry characteristically performs these functions in an overlapped fashion to minimize the total development schedule. The design phase requires technical analysis and applied research to resolve significant technological unknowns for detailed design of components and subsystems. This phase results in the release of drawings and specifications to vendors to initiate manufacturing. Production planning, tooling design and fabrication lead to manufacturing of ground support and flight equipment.

Ground testing simulates the flight design requirements. The anticipated level of actual flight environments are usually exceeded, leaving the margin as small as possible to assure high performance without degrading reliability. Acoustical, structural, static and dynamic, and thermal environments are among those imposed upon test articles. Only after development and qualification testing, are the components and subsystems committed to final system testing. Saturn stages use structural, dynamic, battleship and facility loading vehicles for ground testing. These test vehicles provide preflight system operating data. For large carrier rockets, static firing tests are conducted on test stands to provide system qualification and verification testing prior to flight. Extensive instrumentation is generally used during this ground test phase to provide maximum technical data retrieval, (figures 7 and 8).

It should be noted that energy release rates for space rocket propulsion systems are several magnitudes higher than in aircraft propulsion systems. Combustible fuels and oxidizers require stringent safety measures to preclude test failures and possible stage destruction. Another factor requiring comprehensive instrumentation is that remote control and observation are mandatory during the development phase. (Figure 9)

The static acceptance firing of a flight vehicle (figure 10) is not the final step for flight commitment. A successful checkout, just prior to launch, certifies the booster ready for flight. The value of acceptance firing does not have universal acceptance. One faction considers that static firing expends operational life and degrades ultimate flight reliability. Static Firing, proponents, on the other hand, argue that early in a development program this degration is more than outweighed by system operating experience obtained. The eventual result is deletion of acceptance firing at an appropriate point in the stage hardware development program.

Large carrier rockets must be brought to operational status with a minimum of developmental flights. Although early missile test programs required as many as 50 developmental shots, current space booster ground and flight test philosophy and the introduction of multi-channel flight telemetry has permitted the number to be reduced by a factor of ten, conserving cost and time. This has been accomplished by rigorous, successful ground testing, followed by a flight program with early success. This has been demonstrated in the Saturn program where the Saturn I rocket booster achieved success on all ten of its flights and the Saturn IB rocket was successful on the first attempt.

Early liquid rockets used liquid oxygen (LOX) with conventional noncryogenic hydrocarbon fuels such as ethyl alcohol. The V-2 ballistic missile represented the first major operational development of this propellant combination. Subsequent liquid booster systems evolved using LOX/kerosene and storable propellants such as nitrogen tetroxide (N_2O_4) and unsymmetrical dimethyl hydrazine (UDMH) and hydrazine (N_2H_4) combinations. Although storable systems generally have somewhat lower specific impulse than LOX-hydrocarbon systems, they possess operational advantages such as simplified ground equipment, storability, reduced safety hazards (excluding toxicity), and excellent flight readiness. Another family of rockets use solid propellants because of improved operational characteristics, such as propulsion system simplicity and fast launch reaction time.

For chemical propellants, LOX/LH_2 is one of the highest performing liquid propellant combinations known and used today. It produces approximately 430 seconds impulse for large area ratio nozzles. Liquid fluorine/liquid hydrogen (LF_2/LH_2) produces approximately 7 percent improvement in specific impulse but is still in its early developmental phase. Because of the great development experience with LOX, fluorine may never be used for flight systems.

LH_2 TECHNOLOGY

Work in the United States on LH_2 rockets dates from 1952. An initial problem was efficient production of LH_2 in large quantities. Significant pioneering research work was conducted by the National Bureau of Standards in the conversion of ortho-hydrogen to para-hydrogen. Because of the extremely low saturation temperature at one atmosphere (21° Kelvin) hydrogen poses serious ground handling problems. Much of the technology developed for LOX was applicable to LH_2 both in ground and flight systems which helped speed development. Detailed technical areas of LH_2 systems requiring special consideration were: 1) valve design, 2) propellant transfer techniques, 3) purging techniques and fill procedures, 4) venting techniques, 5) safety, and 6) instrumentation.

In the design of LH_2 valves the general packing designs adequate for LOX are also applicable to LH_2 service, (figure 11). The major difference is assuring that the packing glands are located far enough from LH_2 to keep the shaft and seal temperature equivalent to that of LOX service. Because the hydrogen molecule is significantly smaller, leakage poses a much more severe problem. Significantly tighter shaft and seal tolerances are required.

Large storage vessels are located remotely from the test stand area for safety. This requires long propellant transfer lines. LH_2 lines must be vacuum jacketed to minimize boiloff and to assure good quality liquid propellant into the flight tanks. Bolted flange connections are avoided wherever possible by welding lines together. At geometric discontinuities, such as valve bodies, foam insulation is used to minimize heat leaks, since a vacuum jacket around a valve body could be excessively complex.

Large LH_2 tanks require specialized purge techniques to avoid the accumulation of oxygen, moisture, liquid or frozen air, frozen nitrogen, and other contaminants. A LH_2 tank is filled with gaseous nitrogen for leak checks and standby. The nitrogen is then expelled by a series of helium purges until the nitrogen content is reduced below 1 percent by volume. It is important to keep helium pressurant free of moisture.

The maximum commercial purity attainable is used. Filling procedures use low flow rates until the hydrogen tank pressurant (helium) is chilled to near working temperatures. If this is not done, geysering of LH_2 can cause rapid cooling reducing tank pressure below atmospheric pressure, resulting in inversion of the tank wall. After LH_2 operating temperatures have been achieved at low levels, the tank is rapidly loaded at the maximum fill rate to the 99 percent level. Topping to the 100 percent level is conducted at low flow rates. The LOX loading is similar to this procedure.

The S-IVB common bulkhead undergoes severe thermal stresses in the loading procedure. To minimize these, the LOX tank is filled, putting the upper face (LH_2 Tank side) in compression. When LH_2 is introduced, the thermally induced stresses are relieved by the cold hydrogen.

Hydrogen has a very wide flammability range in air (from 4 percent to 75 percent concentration by volume). Whenever hydrogen is handled in large systems, leaks are possible. In addition, when large quantities of hydrogen are vented or dumped, static discharge may provide an ignition source. To preclude serious fires, hydrogen is vented through a closed system to a remote location from the test stand. Douglas, at its static test site, uses a burn pond where hydrogen is consumed under controlled conditions in air.

As previously stated, hydrogen systems inevitably produce leaks because of the extremely small molecule size and low handling temperatures. Fire detection and hydrogen sensing are both important. Low pressure hydrogen in large quantities has been spilled without causing a fire.

However, when higher pressure hydrogen leaks are encountered (200-300 psi) fires almost always result. The hydrogen/air flame is of very low luminosity, generally invisible. Douglas uses burn wires, strung at critical locations in the engine compartment and near the tank fill and engine feed lines and valves. Discontinuity in the wire, caused by burning, will automatically provide a signal. The system has worked with high reliability.

It is routine procedure for test personnel to visually examine the condition of the vehicle for leaks after propellants are loaded prior to the acceptance firing. A simple effective sensor detects hydrogen concentration as low as 1 percent. After locating the source, corrective measures can be taken.

INSTRUMENTATION

Temperature and pressure transducer instrumentation techniques evolved for LOX are directly applicable to LH_2. Temperature transducers, either as probes to measure fluid temperature or as patches to measure surface temperatures are constructed from fine platinum wire. (On S-IVB, 0.4 to 1.0 mill diameter wire [0.0004 to 0.001 inches] is wound around a platinum or ceramic mandrel. Temperature change varies the resistance of a Wheatstone bridge. The resistance variations may be accurately predicted by calibration at various temperatures.) Temperatures measured on S-IVB vary from -440° to +1800°F.

Square temperature patches (from 0.1 inches to 1.0 inches) are bonded to vehicle structure, black boxes, and propellant lines. The major difference between the surface patches and the aforementioned probes is in the physical configuration and installation modes.

Pressure transducers used on the S-IV and S-IVB are comprised of strain gage and potentiometer types. (The strain gage consists of a stainless steel diaphragm with the gage circuit attached thereto. Pressure deflects the diaphragm, elastically stretching the wire, and unbalancing the Wheatstone bridge.) Hydrogen pressures are measured from -425°F to +600°F, maintaining high accuracy and frequency response where necessary.

Potentiometer transducers are used to handle low frequency pressures in moderate environments. (A Bourdon tube, or bellows, actuates a movable contact, again changing the resistance of a Wheatstone bridge.)

Point level sensors indicate liquid level measuring the change in capacitance of four stainless steel concentric rings. The change in capacitance is caused by the difference in dielectric constant between liquid and gas.

Conventional turbine flow meters are used to measure flow rates; and piezo electric crystal elements are used for vibration and acoustics.

MATERIALS

Material selection is critical for cryogenic propellants. The AISI 300 series (18 percent chromium and 8 percent nickel alloy group) stainless steel is extensively used for ground installation due to its excellent low temperature characteristics and good weldability. Specific alloys used are 304L (low carbon), 321 (titanium stabilized), and 347 (columbium stabilized), depending upon the specific application. Usually 304L or 321 are preferable for most welding applications.

Saturn propellant tanks are fabricated from aluminum alloy for both LH_2 and LOX tanks. Douglas uses 2014T6 aluminum alloy which is mechanically formed, chemically and mechanically milled to form integral waffle patterns in areas encountering critical buckling loads. The tank sections are welded using metal-inert-gas (MLG) and tungsten-insert-gas (TIG) processes. Extensive tooling is required to achieve acceptable tolerances both in detail fabrication and assembly. This type of aluminum structure has the advantage of being free-standing on the launch pad with the payload emplaced without internal pressure required.

Titanium 6Al4V alloy is used for the S-IVB stage high pressure (3,000 psi) bottles; 22-inch diameter spheres are immersed in LH_2 for minimum system weight. Ambient temperature gas storage is accomplished with 24-inch diameter spheres. The excellent strength properties at LH_2 temperatures, and the high strength/weight ratios exhibited at ambient temperatures are the major advantages provided by titanium. Solid roller billets are forged, machined, and diffusion welded to form the spheres.

Potential titanium applications to large volume tankage with LOX and LH_2 are of interest; however, titanium is not considered compatible with LOX by some agencies. By suitable treatment or passivation, this problem can probably be overcome.

Teflon (fluorinated hydrocarbon) is used in conjunction with raised-face, serrated seals for bolted flange connections on the ground. Other line joints use ring face seals with oval or octagonal cross section copper rings. This joint is used where disassembly is absolutely necessary to remove components for maintenance. As previously stated, welded connections are used whenever possible to preclude leakage.

Although LH_2 does not have the incompatibility problems characteristic of LOX, the same level of system cleanliness is used. In addition, liquid hydrogen encounters contamination from N_2, H_2O, and liquid air. These frozen constituents can cause valve and heat exchanger malfunctions. Tanks must be periodically emptied and brought to ambient temperature.

ENGINE DEVELOPMENT

In aircraft, as well as in spacecraft development, propulsion system development has preced vehicle development by several years. Pratt and Whitney started development of the first LH_2 engine in 1957. The RL-10 is a 15,000 pound thrust, 300 psi chamber pressure engine using LOX/LH_2 at a mixture ratio of 5 to 1 (oxidizer to fuel). No gas generator was used, turbine power being provided by GH_2 bled from the thrust chamber regenerative cooling jacket. Hydrogen is bypassed around the turbine pump and fed directly into the injector to control the chamber pressure and thrust. The first flight was used on the Centaur stage whose development was initiated in 1958. Development of the Saturn S-IV stage was initiated in 1960. After overcoming the development problems characteristic of the new LH_2 engine technology, the P&W RL-10 engine emerged as an extremely reliable engine. This was important since two of the engines were used on each Centaur and six engines were clustered to provide 90,000 pounds of thrust for the S-IV.

The next major advance in engine development brought the Rocketdyne J-2 200,000 pound LOX/LH_2 engine to the operational phase. This program was initiated in 1960. The engine cycle was conventional, using the gas generator for turbine power supply, driving separate oxidizer and fuel turbines and pumps. Turbine exhaust is introduced into the thrust chamber divergent nozzle. Like the RL-10, this engine is also used for two flight applications. The S-II stage clusters five of these engines to produce 1,000,000 pounds of thrust for the second stage of the Saturn V launch vehicle. The S-IVB uses a single engine for both the Saturn IB/S-IVB and the Saturn V/S-IVB, the latter in a restartable configuration. The S-II program was initiated in September 1961; the S-IVB program was initiated in December 1961 - lagging engine development by a year.

Engine development tests for the RL-10 were conducted for approximately four years prior to installation on the stage. J-2 development tests were conducted for two years prior to stage testing. The stage development programs for the S-IV and S-IVB initially used heavy-walled, stainless steel Battleship nonflight stages with internal tank geometry identical to the flight tank volumes and shapes. After a series of system verification tests confirming the basic stage propulsion system design concepts, prototype flight configuration system testing was initiated. Using this technique, the S-IV and S-IVB stages were sufficiently developed through ground testing to achieve successful first flights. Even after static firing, engine development is continued to further improve engine reliability and performance. With each successful flight, therefore, stage reliability improves.

S-IVB MISSIONS

As stated before, the basic function of an upper stage is to provide sufficient kinetic energy in a predetermined direction for the spacecraft to achieve its primary mission, which is earth orbital velocity in the case of the Saturn I/S-IV and the Saturn IB/S-IVB applications. The S-IV stage provided approximately 85 percent of the payload kinetic energy injecting approximately 22,000 pounds of payload into 100 nautical mile earth orbit. The Saturn IB/S-IVB correspondingly provides 90 percent of the payload total kinetic energy injecting approximately 36,000 pounds of payload into 100 nautical mile orbit. For the Saturn V three-stage vehicle, the S-IVB provides 60 percent of the payload kinetic energy, 50 percent of that in the second burn after restart from earth orbit into the translunar trajectory. This is a measure of the criticality in performance of the upper stage. In performing its primary function, the S-IVB must interface with the J-2, the Instrument Unit, and the ground support equipment at the static test and launch facilities. Other mission requirements are to provide vehicle flight control by gimballing the J-2 engine and by actuating the Auxiliary Power System (APS), telemeter flight performance data to ground stations, and separate from the lower stages.

In performance of the dual burn mission for the Saturn V vehicle application, approximately 6-1/2 hours of coasting flight are required.

During this period, the S-IVB APS must provide vehicle/spacecraft attitude control. During the 4-1/2 hours of earth orbital coasting flight, an additional propellant management requirement is imposed under zero g or near-zero g (10^{-5} g's). Under this extremely low acceleration, the liquid hydrogen tank is continuously vented at a pressure level of approximately 20 psia. The hydrogen boils due to aerodynamic heating during boost, and solar and earth albedo fluxes in orbit. It is essential that only gaseous hydrogen be discharged to retain the liquid propellant for maximum propulsive efficiency. Baffles are required to preclude liquid entrainment in the hydrogen vent line. The feasibility of this technique has been proven in an experiment with a specially modified Saturn S-IVB stage, (figures 12 and 13). Observation of the hydrogen tank interior was telemetered to ground stations from on-board flight television, (figure 4). Special instrumentation to determine hydrogen propellant liquid/gas phase interface position was also provided.

After restart, when the remaining two-thirds of the S-IVB propellant is consumed, the stage provides two hours of translunar coast attitude control during which the Apollo Spacecraft Command/Service Module (CSM) is transposed and docked to the Lunar Excursion Module (LEM). Only after this operation is the Apollo spacecraft complex separated from the S-IVB.

SYSTEMS DESCRIPTION

The S-IV and S-IVB flight stages are comprised of the following major subsystems: structure, propulsion, control, and electronics.

As previously indicated, the propellant tank assembly is an all-welded 2014-T6 aluminum structure forming a cylindrical hydrogen tank wall, capped by hemispherical domes on each end. Integrally connected to the aft dome, is an aluminum-faced fiberglass honeycomb sandwich common bulkheat (figure 14) (spherical sector in shape) separating the oxidizer tank from the fuel tank. This design was originally developed for S-IV and directly applied to the S-IVB. Other major structural subassemblies are the forward and aft cylindrical skirts and aft interstage, built with conventional skin and stringer aluminum construction. Thrust and gimbal loads from the J-2 engine are passed into the vehicle by an aluminum casting to a conical skin and stringer frustum mechanically attached to the aft dome. The S-IVB stage separation plane is between the aft skirt and interstage. Separation is accomplished by exploding a confined detonating fuse, parting a cylindrical tension member initiated by exploding bridgewire.

An interesting aerodynamic problem required provisions of vents in the aft interstages for controlled dissemination of atmospheric pressure during boost flight. Careful sizing of the venting orifices was required to prevent excessive internal pressure which could collapse lower stage structure and yet provide adequate pressure differential to counteract aerodynamic loads (especially critical for the conical Saturn V interstage).

To minimize propellant boiloff of the liquid hydrogen during orbital coast, it is found necessary to provide internal insulation. The thermal conductivity is approximately 0.03 BTU/ft^2 hr°F. This insulation was developed by Douglas specifically for the Saturn upper stage applications. A polyurethane foam (0.1 specific gravity) is reinforced in three dimensions by orthogonally oriented fiberglass strands. The insulation blocks, approximately 1 foot square by 1 inch thick, are nested and bonded in the internal waffle pattern of the hydrogen tank. The forward dome is covered with 1/2 inch thick insulation. The internal surface is lined by fiberglass cloth impregnated with epoxy resin. Although this construction does not prevent hydrogen diffusion, convective currents are so restricted that appropriate heat transfer rates are achieved.

The major functions of the propulsion system are: propellant containment, propellant positioning, pressurization, engine feed, venting, and propellant conditioning. Because mass fractions must be maximized in the upper stage, high propellant utilization efficiencies are required. During the S-IVB terminal countdown, just prior to launch, propellants are topped off to ±1/4 percent nominal mission propellant load. The maximum propellant load leaves only 3 to 4 percent ullage for initial pressurizing gas volume, imposing stringent starting requirements on the pressurization system design.

Different design concepts are used for liquid hydrogen and liquid oxygen tank pressurization. The design pressures in the hydrogen and oxygen tank are 39 psia and 44 psia, respectively. Prior to engine start both tanks are pressurized with helium from a ground source. During the starting transient initial pressurization is provided by on-board helium supplied from high pressure bottles through appropriate regulating devices. During steady state operation, the liquid hydrogen tank is pressurized from warm hydrogen gas bled from the J-2 at approximately 110°K. The liquid oxygen pressurant during S-IVB boost is supplied by cold helium bottles immersed in the liquid hydrogen tank, heated by the J-2 turbine pump exhaust gases in a heat exchanger and subsequently injected at the forward end of the oxidizer tank. This sophisticated technique is used to minimize storage bottle weights, again to maximize payload.

Repressurization for Saturn V restart is accomplished using helium from the cold helium bottles, heated in a LOX/LH_2 burner to increase the tank pressures from approximately 20 psi in the coasting mode, to 39 and 42 psia in the hydrogen and oxygen tanks, respectively, prior to restart. (Earlier vehicles used an ambient helium bottle repressurization system contained in eight 3-cubic-foot spheres mounted on the thrust structure.)

Propellant feed is accomplished through prevalves, vacuum jacketed lines, supplying propellants through 8-inch ducts directly to the oxidizer and fuel pump inlets. Just prior to engine start, propellant is recirculated through the feedlines to assure proper liquid vapor quality during the critical transient engine acceleration phase to preclude pump cavitation. This process is repeated for the second start of the Saturn V/S-IVB after 4-1/2 hours of earth orbital coast.

Propellant sequencing is accomplished by both the stage valves and engine valves actuated by the stage sequencer. Full thrust is achieved approximately 3 seconds after engine start signal. Shutdown is initiated by the stage sequencer or by propellant level sensors contained in the feedlines sensing propellant depletion. After the booster propulsive function is complete, oxygen and hydrogen tank venting is sequenced to avoid perturbations to the spacecraft/vehicle attitude during critical coasting operations. An auxiliary propulsion system (APS), comprised of two diametrically opposed modules, use storable hypergolic propellants, nitrogen tetroxide (N_2O_4) and mono-methyl hydrazine. The S-IVB attitude is controlled in pitch and yaw by gimbaling the J-2 engine and by the auxiliary propulsion system for roll control during powered boost. During coasting flight, the APS provides pitch, yaw and roll control. Signals are initiated by the guidance platform in the Instrument Unit, passed through the stage sequencer and actuating the appropriate APS valves and/or gimbal servo actuators.

Another control function, propellant utilization, is used to assure propellant depletion to within 1/4 of 1 percent of the total usable propellants. Propellant level sensing is accomplished by capacitance probes sensing conductance differences between the liquid and gaseous phase propellant. This signal controls a diverter valve in the J-2, changing engine mixture ratio during flight.

Physical separation of the S-IVB is aided by solid rockets (ullage rockets) of 3,400 pounds of normal thrust operating for 4 seconds. They are mounted in three places equidistant on the aft skirt of Saturn IB and diametrically opposed in two places in Saturn V. In addition to physical stage separation, these rockets also provide acceleration (10^{-3}g) for propellant settling prior to initial engine start.

Although the S-IVB is primarily a propulsive stage, its size, importance, and complexity require 400 channels of instrumentation for R&D flights. These data are transmitted over five telemetry sets, PAM/FM/FM, single sideband (SSB/FM), and digital data acquisition system (PCM/FM). Signal conditioning equipment is mounted in the forward and aft skirt periphery. Control, power distribution, and instrumentation networks are comprised of conventional wire harnesses, connecting the forward and aft equipment areas through a tunnel. The stage functions are accomplished in proper time-phasing controlled by the stage sequencer.

Operational power is supplied from four 28 and 56 volt dc batteries with a total capacity of 380 ampere hours for Saturn IB and 635 ampere hours for Saturn V.

RF antennas are mounted in four places at the forward end of the stage.

GROUND SUPPORT EQUIPMENT

Large carrier rockets are so complex that extensive ground support equipment is necessary. The mechanical equipment is straightforward in design, although unusual in size. Functions that must be performed are handling, access, transportation, and storage, (figure 15).

The electrical support equipment developed for S-IVB represents a significant advance in the state-of-the-art. Primary functions required are factory checkout, pre-static acceptance firing checkout, static firing control, and poststatic checkout (delivery verification) tests. Major system elements, in addition to the computer, are stimuli and signal conditioning units, telemetry ground stations, safety item monitor, computer interface unit, telemetry, pneumatic, electric propulsion, and system test operator consoles. Automatic control is accomplished by checkout procedures programmed into magnetic tape operating a Control Data Corporation CDC 924A digital computer with peripheral equipment. Checkout and static firing functions are performed in a completely automatic mode; however, manual intercession capability for shutdown and "safeing" is provided in case a testing malfunction or anomaly should occur. Similar NASA/MSFC developed equipment will be used at KSC for prelaunch checkout and launch control.

FLIGHT PERFORMANCE

The S-IV flight test program is summarized in table 1. This stage, the forerunner of the S-IVB, provided verification of the Douglas design concepts for liquid hydrogen-oxygen upper stages.

Major performance characteristics of the S-IVB flight tests are shown in table 2. The Saturn V version of the S-IVB is scheduled to fly in 1967. Because of the similarity between the Saturn IB/S-IVB and the Saturn V version, a successful flight test program is anticipated.

FUTURE APPLICATIONS

Current development of the Saturn vehicles is being conducted directly for application to the Apollo Lunar Landing Program. After initial lunar exploration, additional planetary and space exploration missions will be pursued. Because of the large orbital and escape payload capability provided by the Saturn vehicles and because of the significant national investment they represent, these rockets (or modifications thereto) will be used as the basic launch vehicles for the next decade. Several future mission applications have been examined such as planetary probes, comet probes, out-of-the-ecliptic probes, solar probes, and extra-solar applications. These missions are discussed in detail in reference h.

For orbital payloads in the range of about 30,000 to 40,000 pounds the Saturn IB will be used for the next several years. A third stage is required to provide this vehicle with significant escape velocity payload capability. The Saturn V three-stage vehicle can inject 95,000 pounds payload into a translunar trajectory or 240,000 pounds payload into earth orbit.

Planetary exploration is receiving considerable attention in the scientific community. Additional Mars and Venus probes are being considered for both manned and unmanned payloads. More demanding missions investigating Mercury and Jupiter are also of interest. The Saturn V vehicle can boost 24,000 pounds to Jupiter on a 750-day Hohmann transfer mission (figure 16).

Cometology can be significantly advanced by unmanned probes used for comet intercept. The three-stage Saturn V can boost 22,000 pounds on an intercept trajectory with the comet Encke. The trip time would require approximately 100 days. Asteroid probes can also be accomplished with Saturn V, placing 7,000 pounds past Ceres.

For very high velocity missions, a four-stage Saturn V vehicle may evolve. This will permit out-of-the-ecliptic probes, solar probes to within 0.2 astronomical units, as well as shortening transfer time to Jupiter and the nearer planets, Mars and Venus. Probes out of the solar system could also be achieved with the four-stage Saturn V. However, the flight times to reach beyond Pluto are quite large, approximately 11 years for a 13,000 pound extra-solar system payload.

In summary, Saturn V, in a three or four-stage version, can satisfy many complex requirements for solar system exploration in the foreseeable future.

As man ventures into orbit and space operations become routine, the capability must be developed to retrieve the astronauts in case of equipment malfunctions or personnel illness. Depending upon the criteria for reaction time, launch windows, orbital inclinations and altitudes, a high energy rescue device appears mandatory. The rescue mission must ultimately require higher payload velocity capability than the payload with which it will rendezvous. This problem is receiving serious consideration as the tempo of manned operations increases (reference i). The Saturn V vehicle can provide maximum launch capability of this type in the immediate future.

The advent of manned space flight inevitably commits consideration and ultimate development of permanent or semi-permanent orbital operation complexes. The ratio of takeoff weight to payload weight of approximately 25 to 1 for earth orbit, and approximately 50 to 1 for lunar trajectory for the Saturn V vehicle precludes direct payload injection for planetary missions. It is probable that manned exploration of Venus and Mars will be conducted in the not too distant future. To amass sufficient payload capability for the near planet exploration of Mars and Venus, orbital assembly will ultimately be used. Round trips to these two planets will take from two to three years. Orbital payloads in the 500,000 to million-pound range are being discussed. Space propulsion systems and spacecraft can be launched into coplanar orbits assembled checkout prior to interplanetary injection. Since these operations may take several weeks, a long-term operational base must be established.

In addition to manned missions, large scientific payloads will ultimately be developed. Visual and radio astronomical telescopes will be assembled and operated in orbit.

The role of the S-IVB in these future operations is of great importance. In addition, LH_2 technology will be applied, either in chemical or nuclear propulsion, required for future high energy missions.

The LH_2/LOX technology developed by Douglas for NASA represented by the S-IV and S-IVB stages could also be applied to smaller LOX/LH_2 vehicles. Larger rockets generally can achieve higher mass fractions due to structural and equipment weight efficiencies. Smaller stages require increased sophistication to achieve similar design mass fraction objectives. The techniques used on S-IV and S-IVB are applicable specifically in the areas of aluminum tankage, insulation, ground equipment, pressurization techniques, propellant management, propellant utilization, and instrumentation disciplines. The Saturn S-IV/S-IVB technology will significantly contribute to space flights in the future.

SATURN V LAUNCH VEHICLE

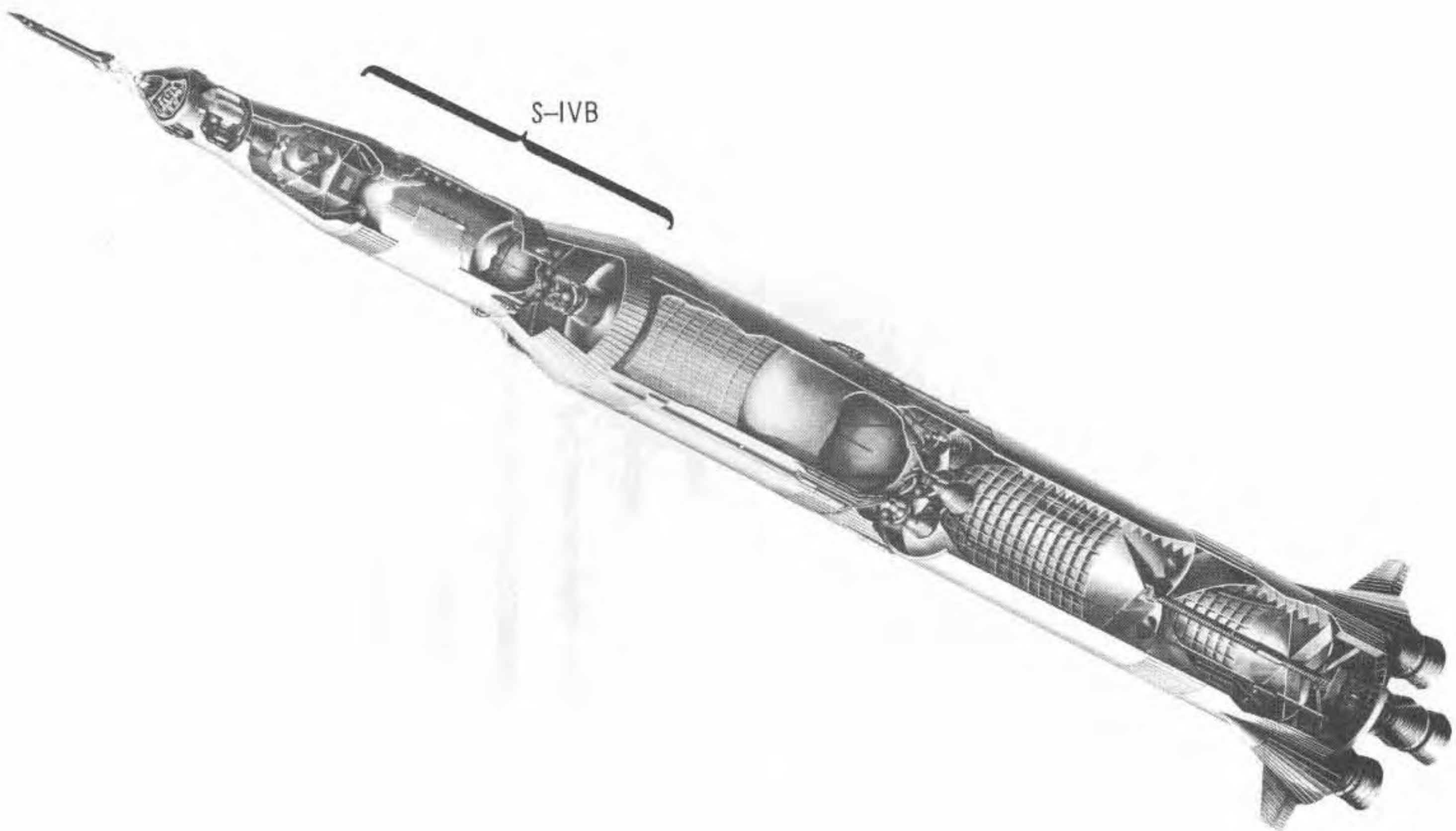


FIGURE 1

SATURN COMPARISON

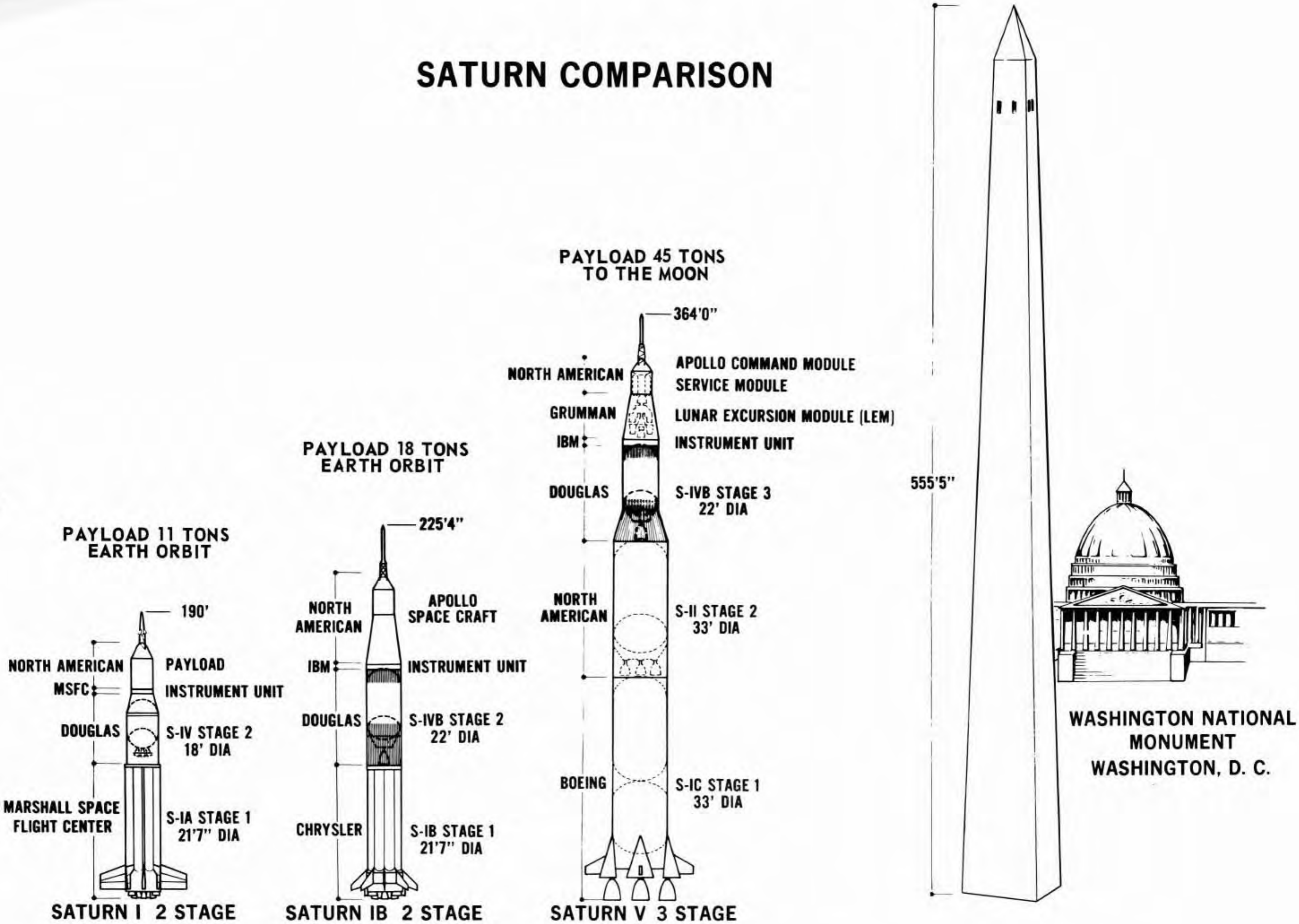


FIGURE 2

SATURN IB LAUNCH VEHICLE

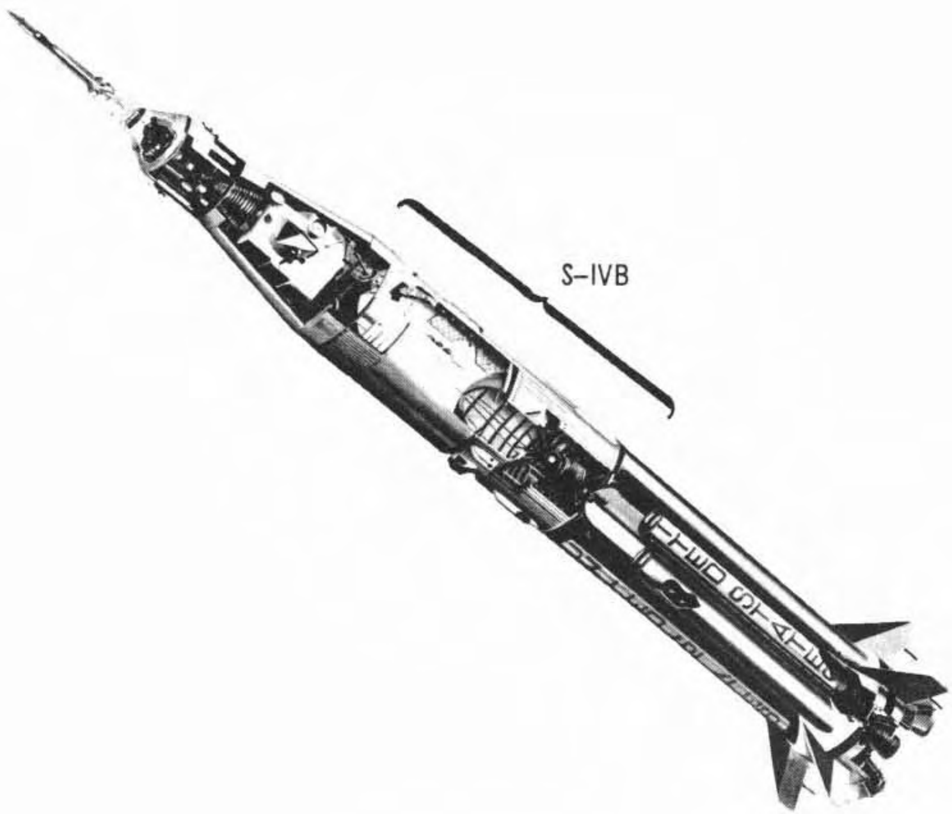


FIGURE 3

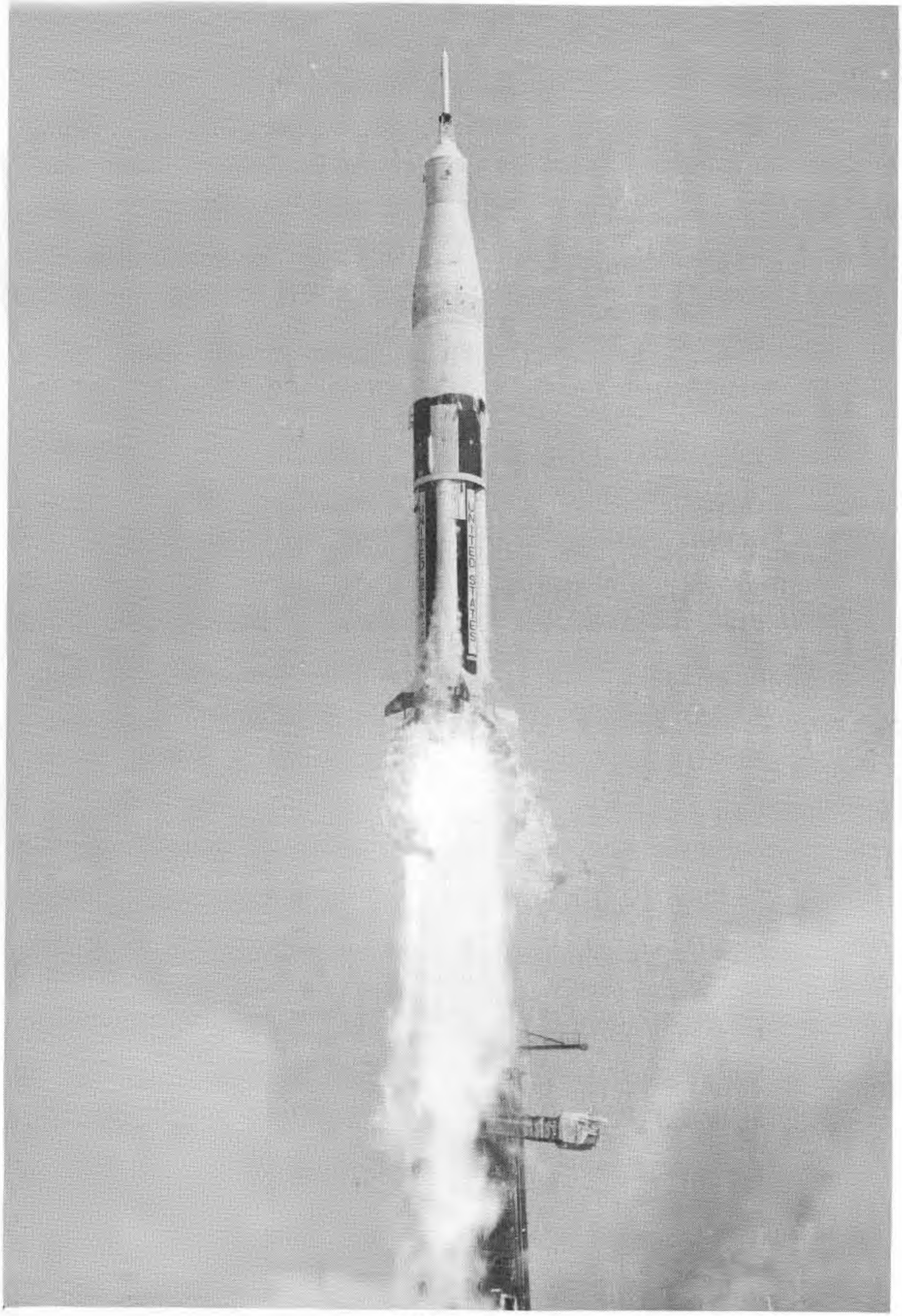
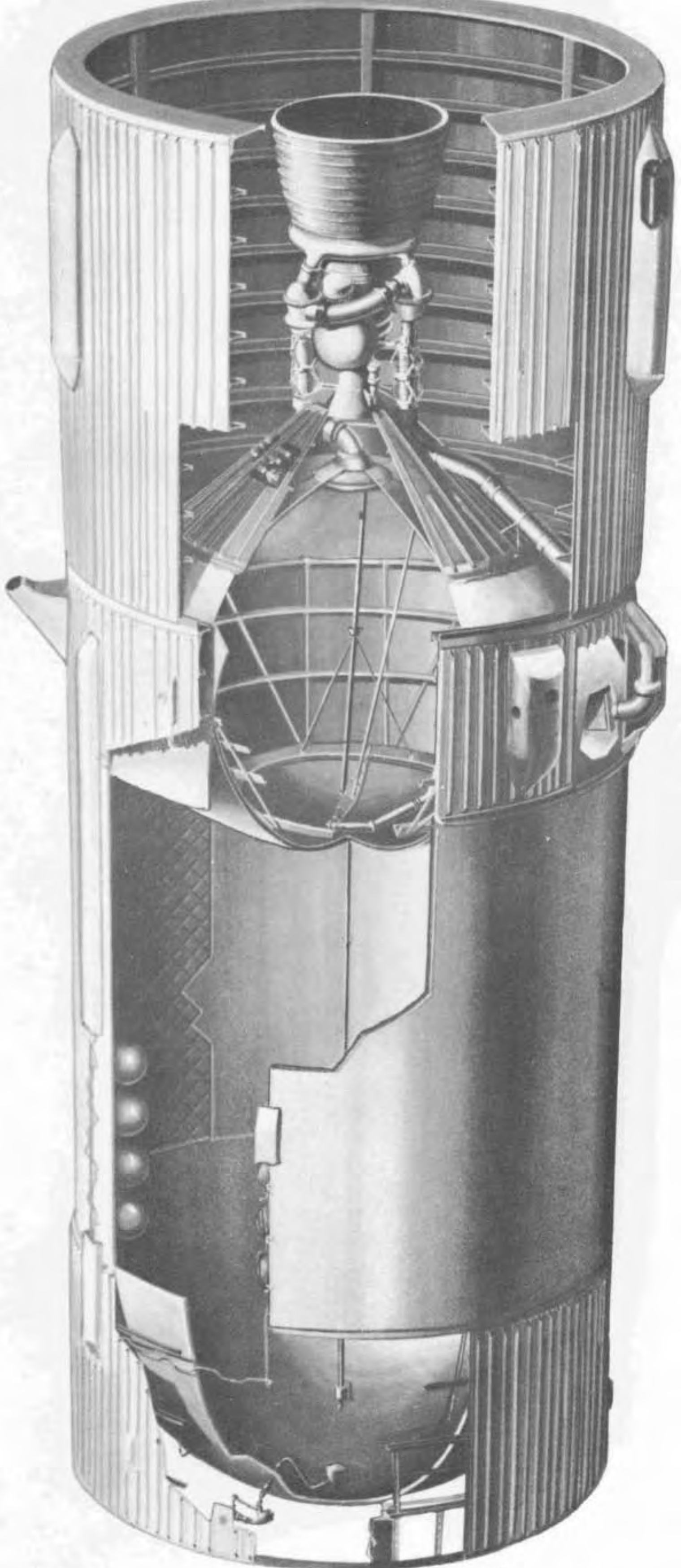


FIGURE 4

SATURN IB SECOND STAGE



SATURN V THIRD STAGE

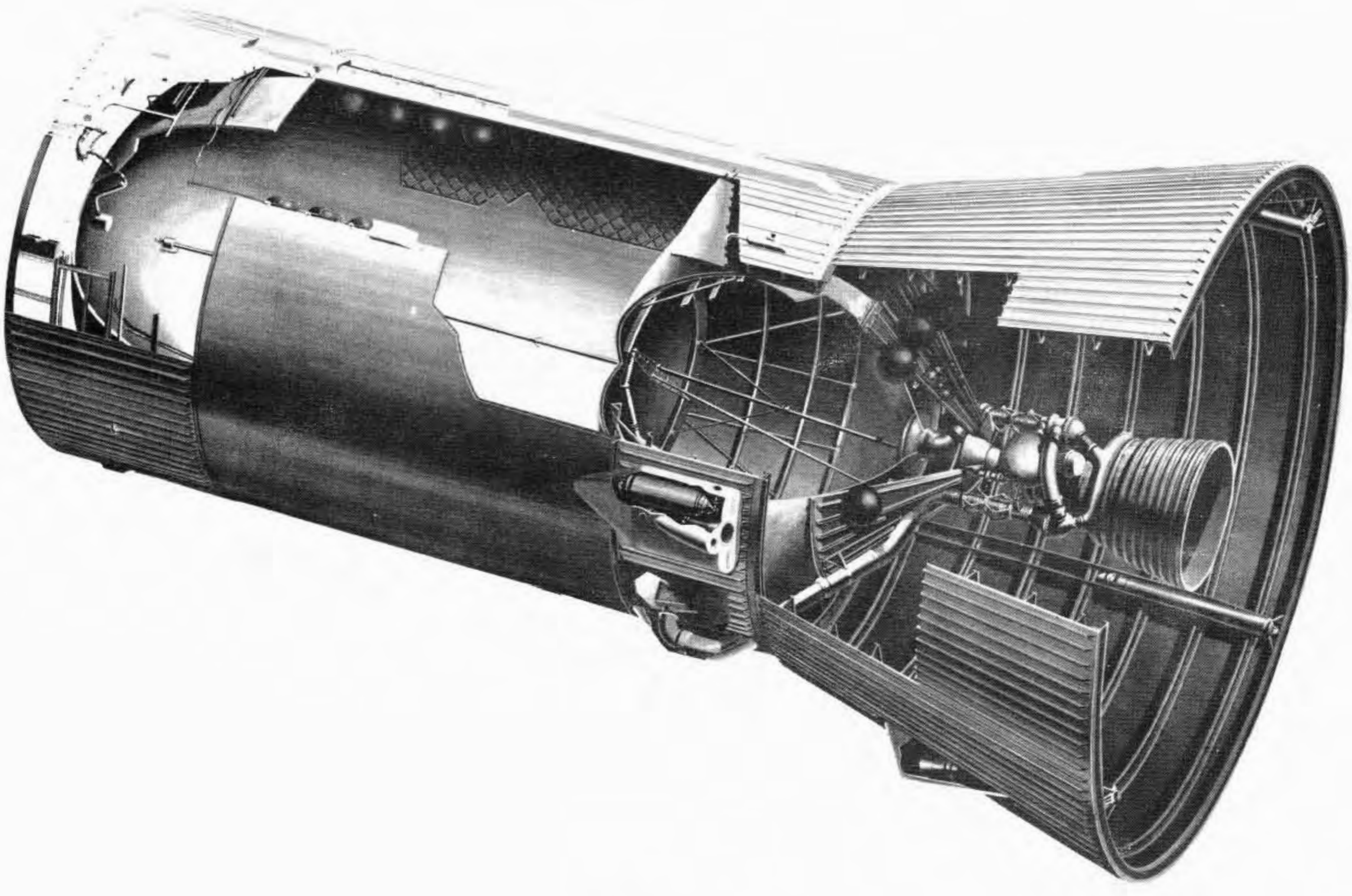


FIGURE 6

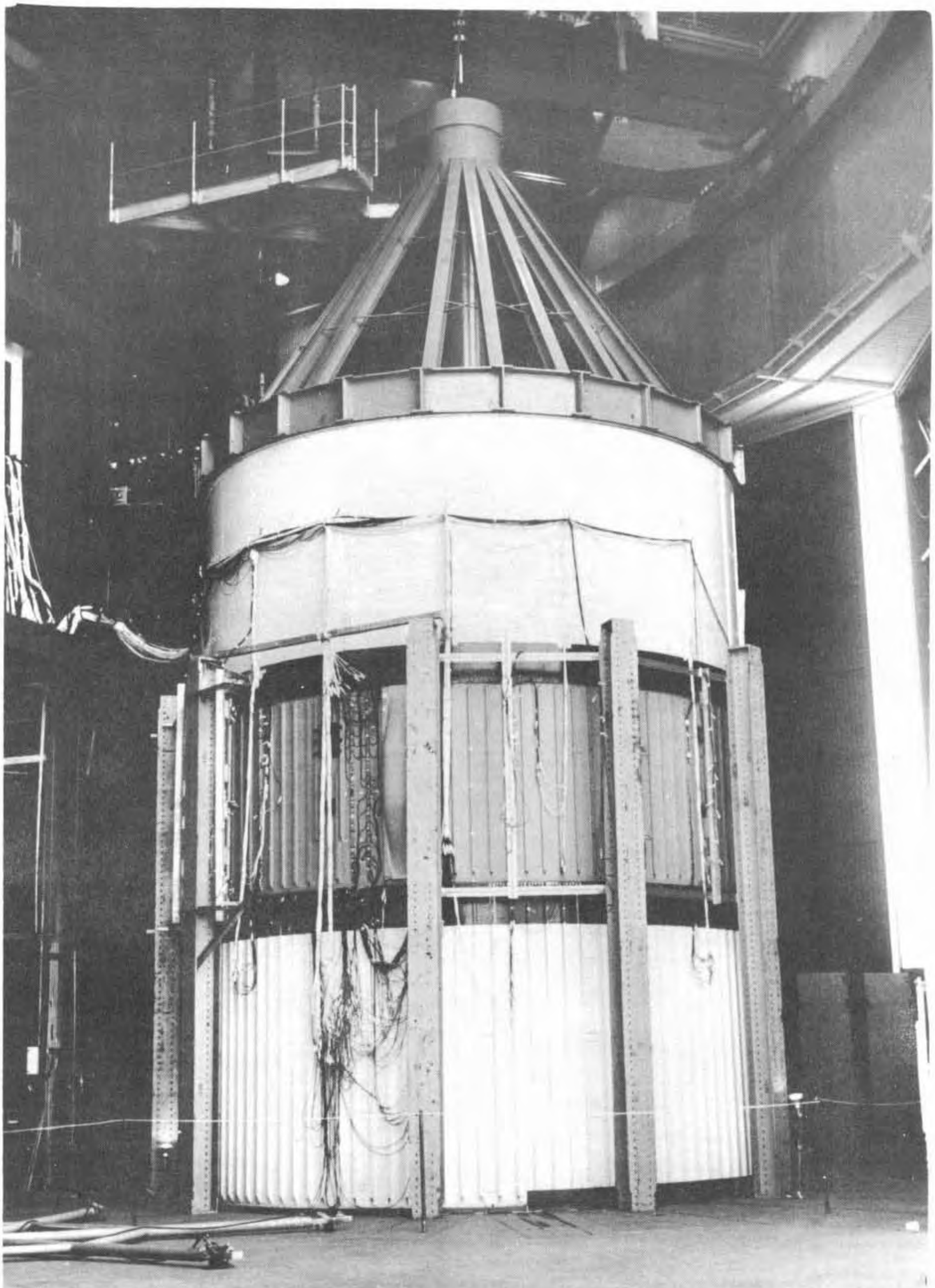


FIGURE 7

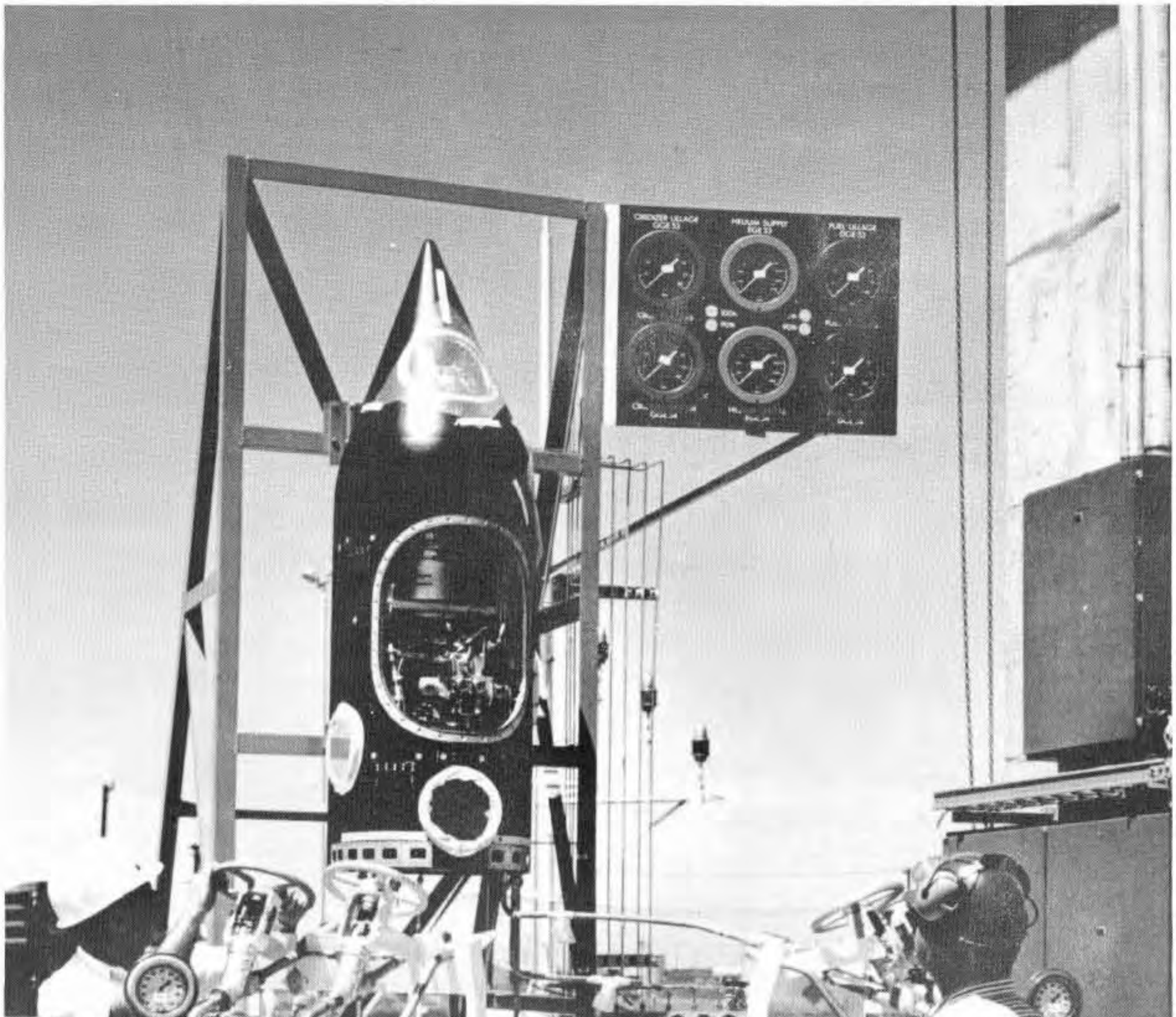


FIGURE 8

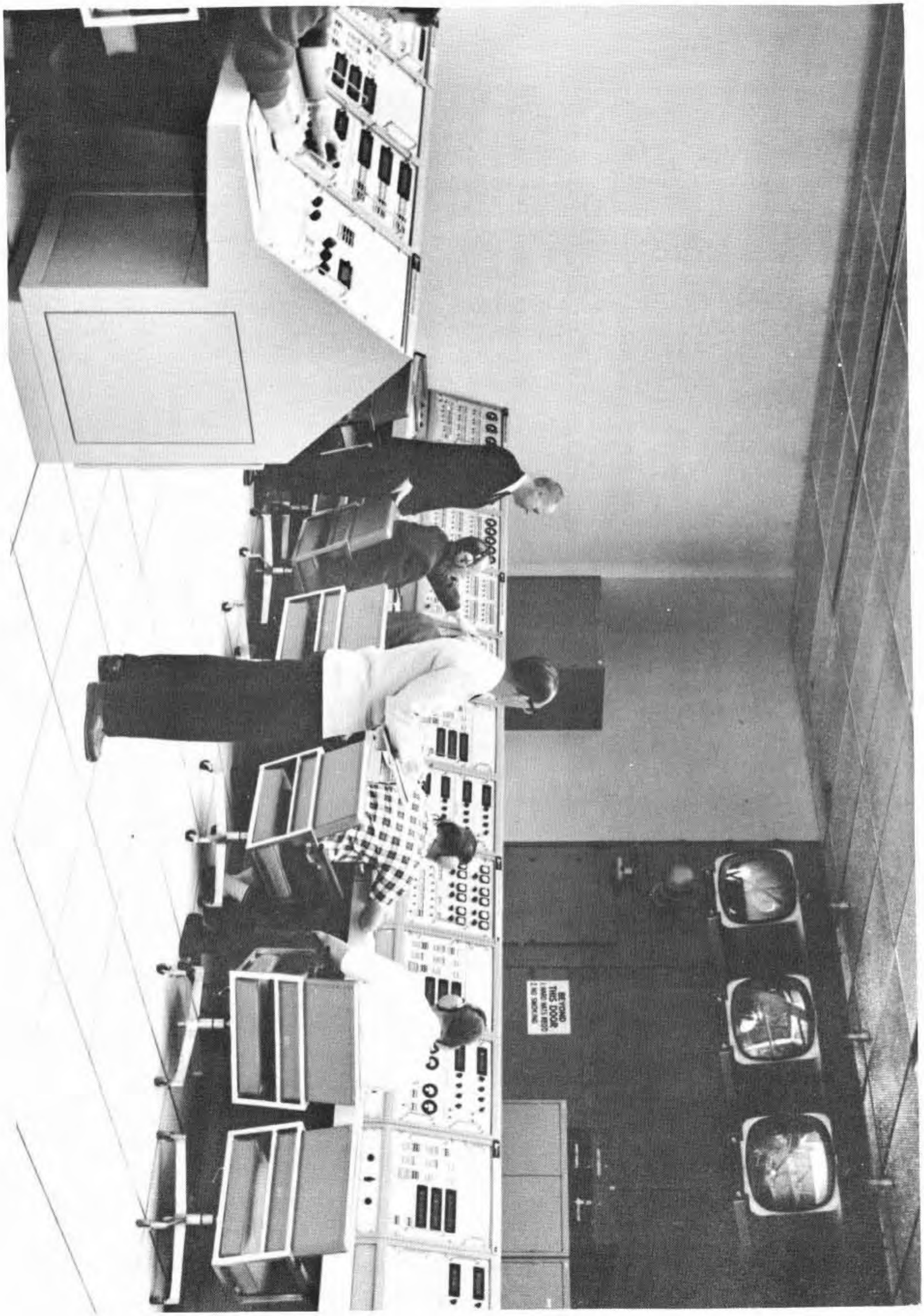


FIGURE 9

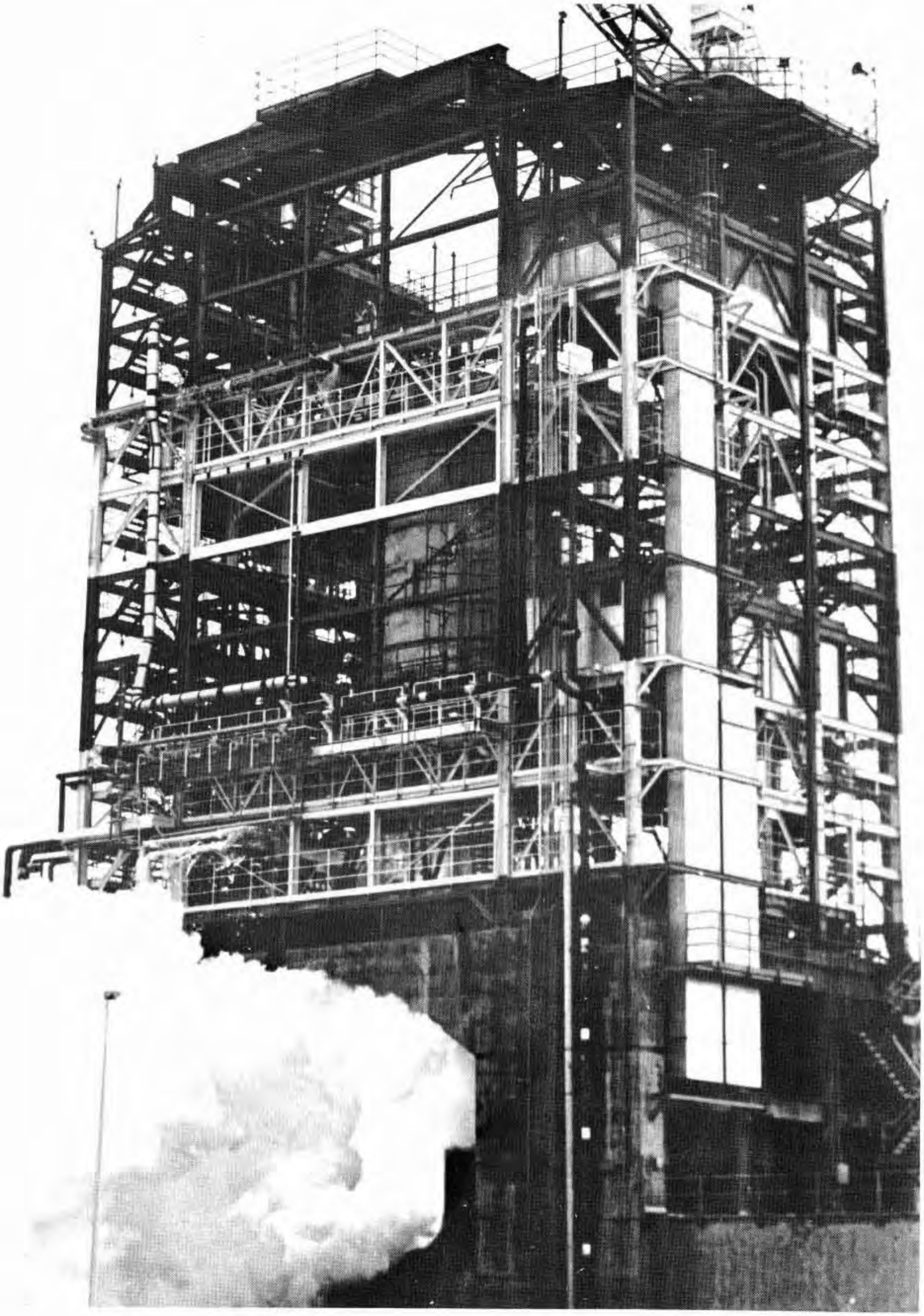


FIGURE 10

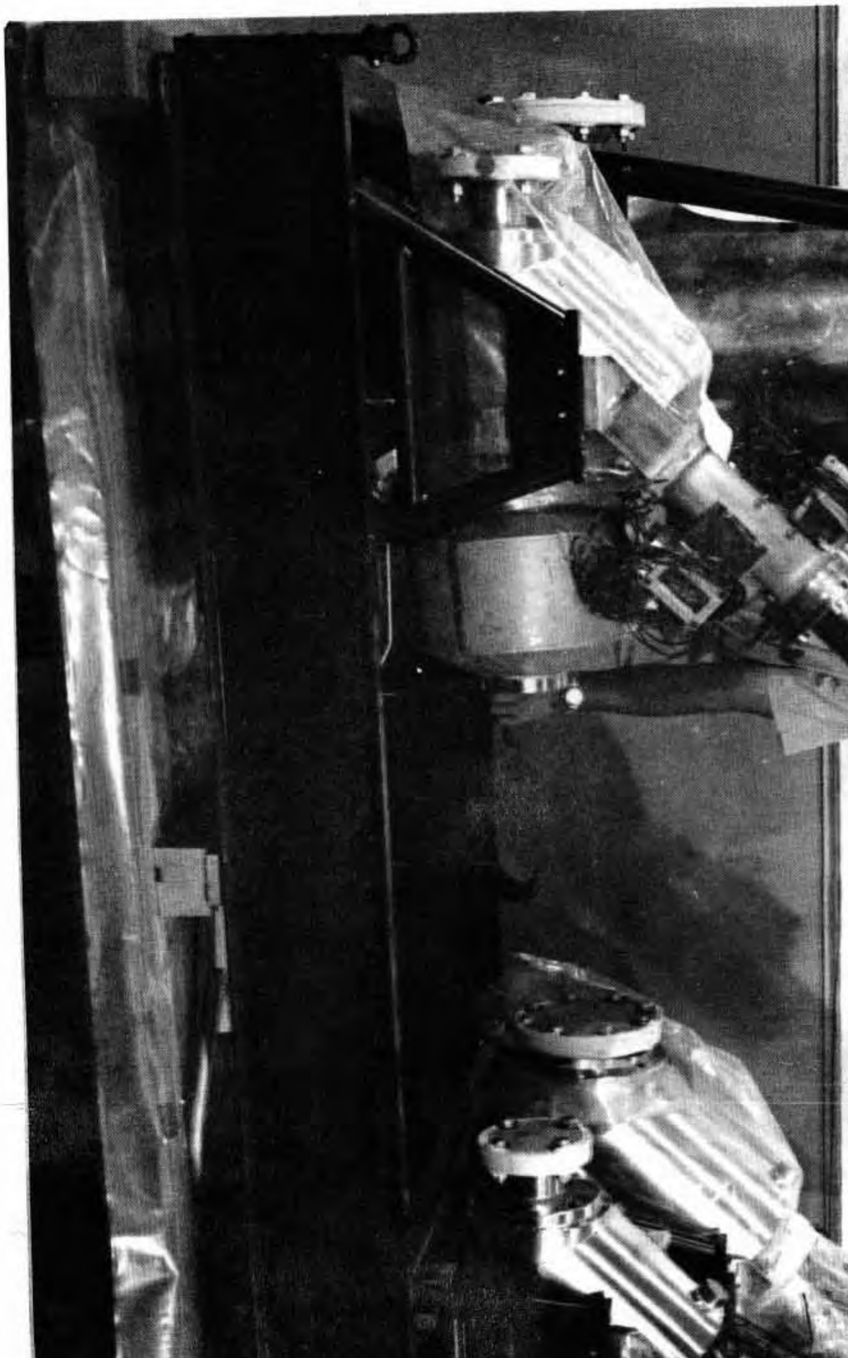


FIGURE 11

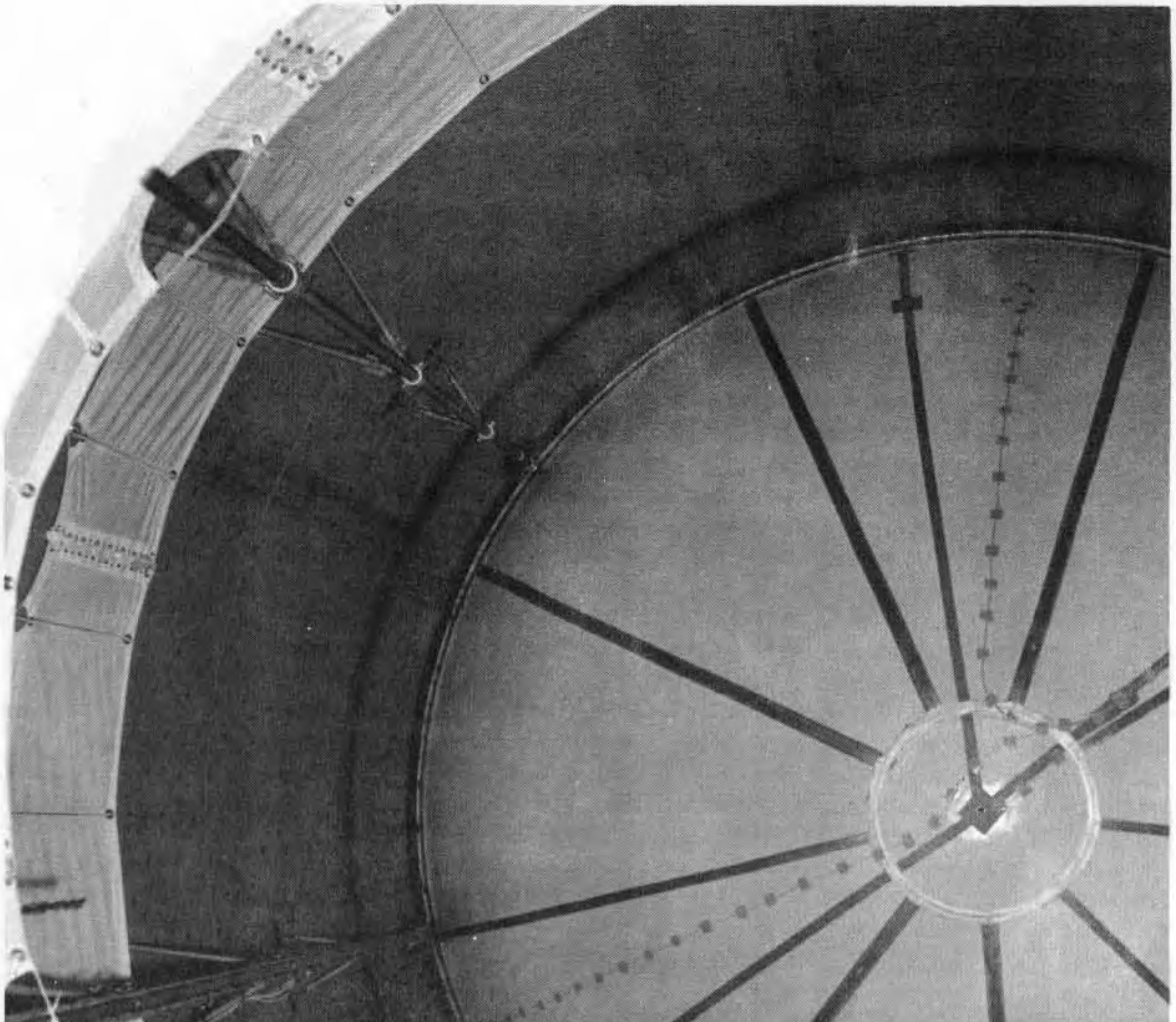


FIGURE 12

SATURN IB S-IVB/203LH$_2$ EXPERIMENT

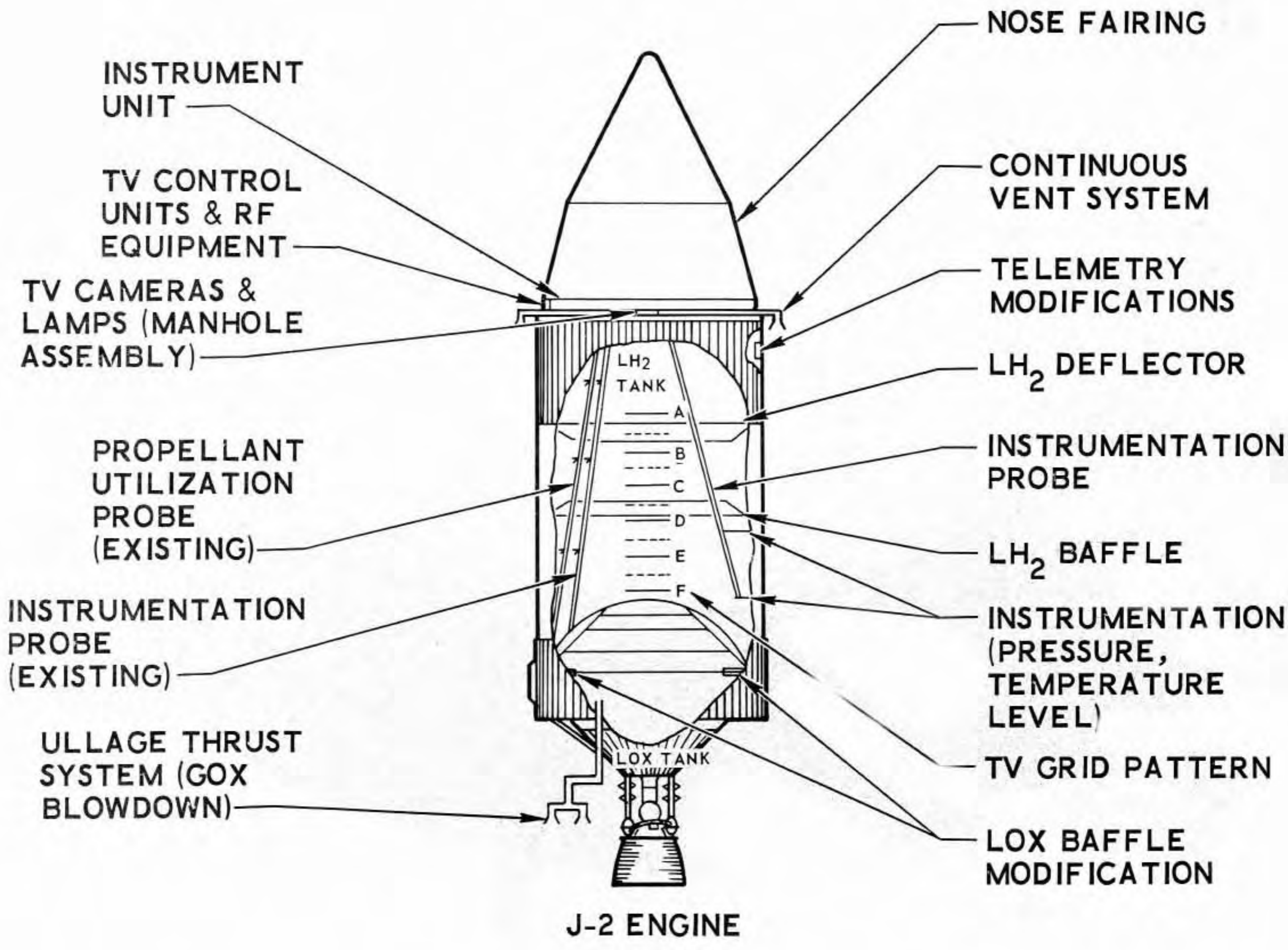


FIGURE 13

SATURN S-IVB
COMMON BULKHEAD JOINT CROSS SECTION

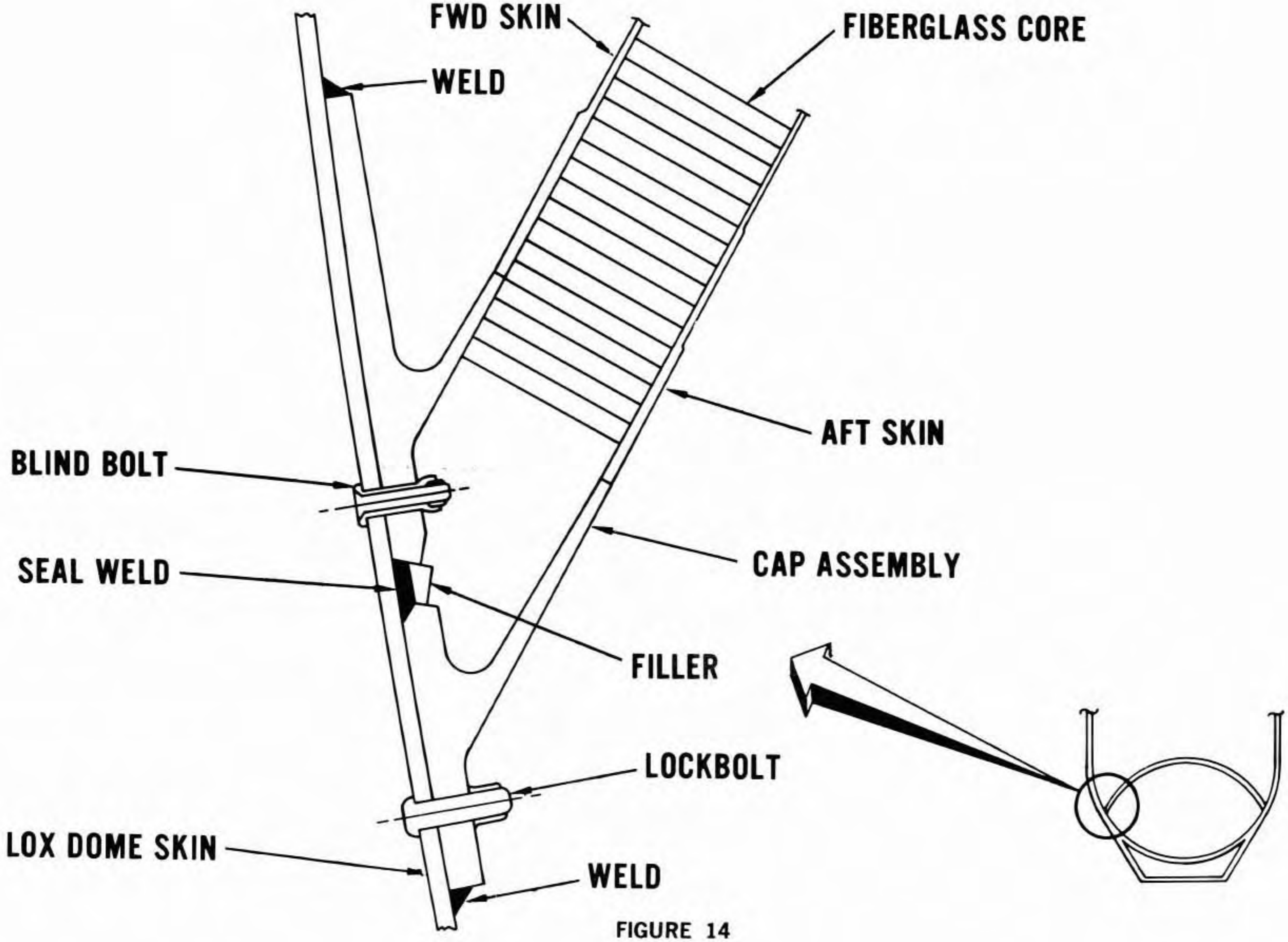


FIGURE 14

FIGURE 15

TRIP TIME VS PAYLOAD CAPABILITY

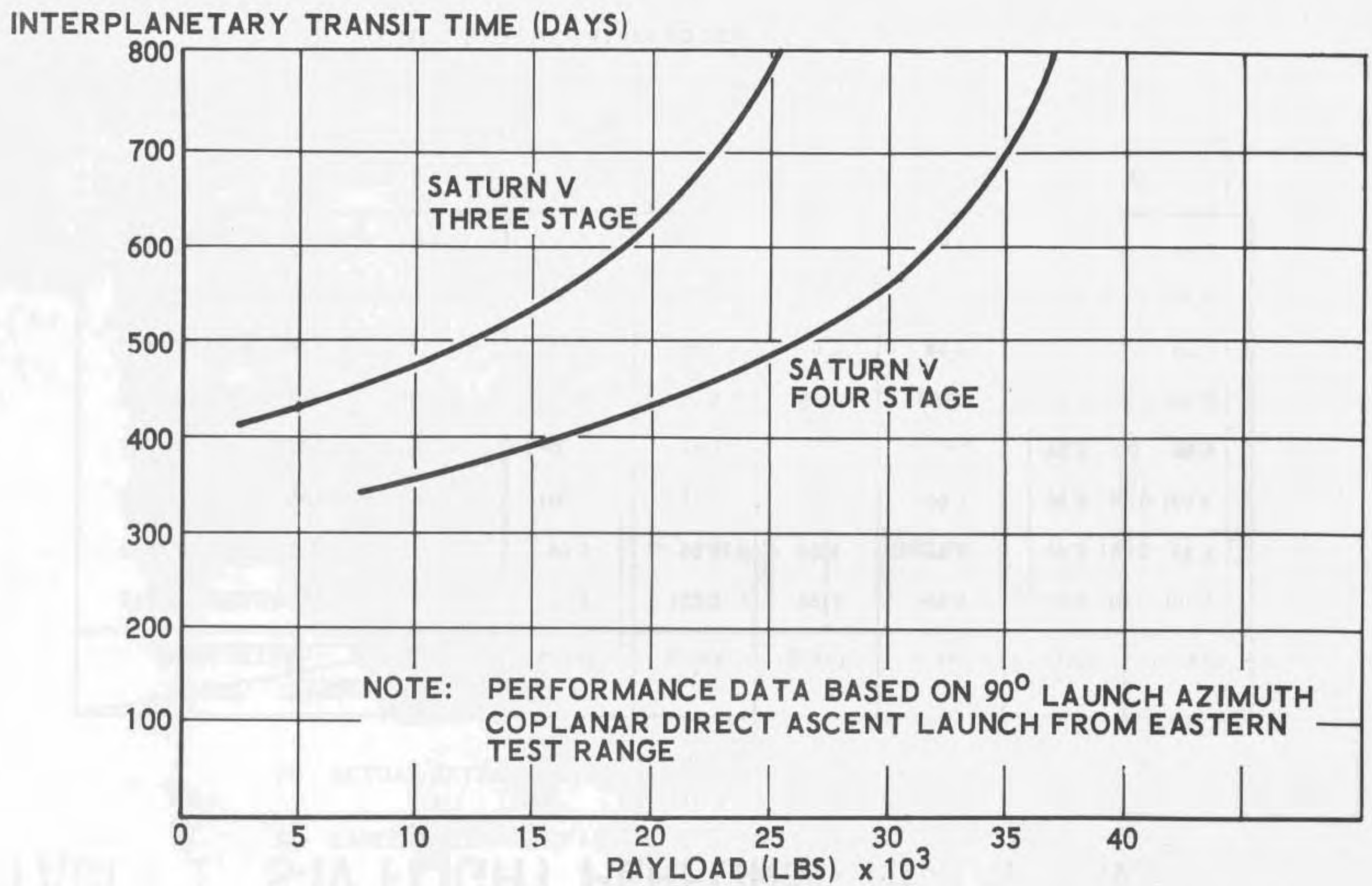


FIGURE 16

TABLE 1. S-IV FLIGHT PERFORMANCE SUMMARY (1)

STAGE / PARAMETER	S–IV5	S–IV6	S–IV7	S–IV8	S–IV9	S–IV10
STAGE THRUST (AVE)	99.9	102.1	99.4	99.9	100.0	100.3
STAGE SPECIFIC IMPULSE	99.9	99.8	99.8	99.8	99.5	99.9
STAGE TOTAL IMPULSE	100.2	101.2	99.6	99.3	98.5	101.9
ENGINE BURNTIME	100.2	99.1	100.3	99.4	98.5	99.9
PROPELLANT UTILIZATION (3)	100.0	99.9	99.9	100.0	99.9	100.0
T/M DATA RETRIEVAL	98.9	98.0	98.7	98.8	99.0	97.3
INJECTION VELOCITY	100.2	100.1	100.0	100.0	100.0	100.0
CUTOFF ALTITUDE	99.0	97.9	98.9	100.0	99.9	99.4

NOTES: 1) $\% = \frac{\text{ACTUAL}}{\text{PREDICTED}} \times 100$

2) ACTUAL EXTRAPOLATION TO DEPLETION

3) BASED ON CHANNELS ACTIVE AT LAUNCH

TABLE 2. S-IVB/SATURN IB FLIGHT TEST DATA

PARAMETER (1)	S-IVB-201	S-IVB-202	S-IVB-203
STAGE THRUST (AVERAGE)	97.5	102.1	99.3
STAGE SPECIFIC IMPULSE	100.0	100.0	100.2
STAGE TOTAL IMPULSE	99.9	98.6	100.2
ENGINE BURNTIME	102.9	97.4	101.0
PROPELLANT UTILIZATION (2)	99.9	99.9	(4)
T/M DATA RETRIEVAL (3)	97.2	99.8	98.0
INJECTION VELOCITY	100.0	100.0	100.0
CUTOFF ALTITUDE	100.5	99.8	100.0

NOTES: 1) $\% = \frac{\text{ACTUAL}}{\text{PREDICTED}} \times 100$

2) ACTUAL EXTRAPOLATION TO DEPLETION

3) BASED ON CHANNELS ACTIVE AT LAUNCH

4) NOT APPLICABLE, SPECIAL EXPERIMENT