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HEADQUARTERS
U.S. ARMY ORDNANCE MISSILE COMMAND
REDSTONE ARSENAL, ALABAMA

IN REPLY
REFER TO ORDXM-CRR

25 FEB 1960

SUBJECT: Semiannual Reports on ARPA Orders 14-59 and 47-59, 39-59 and 74-59 (U)

TO: Director
Advanced Research Projects Agency
Washington 25, D. C.

1. (U) Inclosed are copies of this Command's semiannual technical progress reports on ARPA Orders 14-59 and 47-59; 39-59, Task 4; and 74-59. These reports cover progress from 1 July 1959 through 31 December 1959.

2. (U) AOMC comments, observations, and recommendations on contractor efforts in these tasks are given in AOMC summary reports attached to individual contractor prepared technical reports. The inclosed report on SATURN is based upon the NASA-recommended C-1 configuration for the vehicle.

FOR THE COMMANDER:

THOMAS W. COOKE
Colonel GS
Chief of Staff

5 Incl
1. Semiannual Rept, ARPA Orders 14-59 and 47-59, dtd 15 Feb 60 (S) (6 cys)
2. Semiannual Tech. Summary Rept. Nr 1, ARPA Order 39-59, Task 4 (C) (6 cys)
3. High-Altitude Platform Study, Semiannual Rept., BR-804 (S) (6 cys)
4. Semiannual Tech. Summary Rept. Nr 1, ARPA Order 74-59 (S) (4cys)
5. Research in Coherent Digital Phased Arrays (S) (4 cys)

Copies Furnished:
OCO, ORDPM, w/incls 1,2 and 4
OCRD, w/incls
NASA, Maj. Gen. Ostrander, w/incl 1
NASA, AOMC Rep., w/incl 1

ARPA Cont. No. 1064

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15 February 1960 (5) 80 890 Report No. DIR-TR-1-60
RCS ORDXM-C-1004

(9) SEMIANNUAL TECHNICAL SUMMARY REPORT ON no. 2,
~~ARPA ORDERS 14-59 and 47-59 (U)~~

~~From~~ 1 July 1959 ~~Through~~ 31 December 1959,

Prepared by

ARMY BALLISTIC MISSILE AGENCY

U. S. ARMY ORDNANCE MISSILE COMMAND

REDSTONE ARSENAL, ALABAMA

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SEMIANNUAL TECHNICAL SUMMARY REPORT ON
ARPA ORDERS 14-59 AND 47-59
From 1 July 1959 Through 31 December 1959

FOREWORD

This is the second semiannual technical summary report for the SATURN Project, which is being accomplished for the Advanced Research Projects Agency by the Army Ordnance Missile Command. The report summarizes the Army Ballistic Missile Agency research and development plans, findings, and accomplishments in carrying out ARPA Orders 14-59 and 47-59. It covers the period from 1 July 1959 through 31 December 1959.

SECURITY NOTE

This document contains information affecting the national defense of the United States within the meaning of the Espionage Law, Title 18, U.S.C., Section 793 and 794 as amended. The transmission or revelation of its contents in any manner to an unauthorized person is prohibited by law.

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GLOSSARY

ABMA	Army Ballistic Missile Agency
AOMC	Army Ordnance Missile Command
AMR	Atlantic Missile Range
NAA	North American Aviation, Inc.
Rocketdyne	Rocketdyne Division of North American Aviation H-1 Engine Contractor
H-01 - H-09	These symbols designate R&D H-1 engines which will be used in development testing at Rocketdyne.
H-10 - H-?	These series of H-1 engines will be R&D flight-prototype engines used for testing at Rocketdyne.
H-1001 - H-1010	Production H-1 engines to be used for R&D testing at ABMA and on the static test booster SA-T.
H-1011 - H-?	Production H-1 engines for flight boosters.
SA-T	Designation of the SATURN Static Test Booster to be built and tested at ABMA.
SA-1 - SA-4	SATURN flight vehicles.
RP-1	Kerosene-type fuel used by the H-1 engine as well as by the engines of most of the U.S. large ballistic missiles.
Lox	Liquid oxygen, used as an oxidizer by the H-1 engine and most large ballistic missile engines.
LN_2	Liquid nitrogen, used for instrument compartment cooling, etc.
GN_2	Gaseous nitrogen.
PAM-FM-FM	Standard JUPITER-type pulse amplitude modulated telemetry system.
SATURN C-1	SATURN vehicle with lox-LH_2 second and third stages.

SECTION I. GENERAL

A. (S) Background and Technical Origins of Project

The Army Ballistic Missile Agency was among the early groups who considered a payload capability of 20,000 to 40,000 pounds for orbital missions and 6,000 to 12,000 pounds for escape missions as urgent requirements for space missions of the near future. Development of high thrust engines at that time was also considered essential for future army transport programs.

Studies were initiated by ABMA in April 1957 to establish possible booster configurations to satisfy such programs. The initial studies aiming at boosters in the 1.5 million-pound thrust class placed special emphasis on arriving at an acceptable propulsion system. Clustering of four 380,000-lb thrust NAA E-1 engines which were in early stages of development was first considered. This booster, which in the beginning was designated SUPER-JUPITER, and several stages were investigated by ABMA with the assistance of North American Aviation.

In July 1958, representatives of the Advanced Research Projects Agency (ARPA) showed interest in a clustered booster with 1.5 million pounds thrust achieved with available engine hardware. After considering ABMA's experience in the field as well as the availability of its facilities and manpower, ARPA formally initiated the project by issuing ARPA Order 14-59 on 15 August 1958. The immediate goal of the program was to demonstrate the feasibility of the engine clustering concept with a full-scale captive dynamic firing. In September 1958 the program was extended to include four flight tests of the booster. A separate ARPA Order No. 47-59 dated 11 December 1958 requested that AOMC, on behalf of ARPA, (1) perform the design, construction, and modifications of the ABMA Captive Test Tower and associated facilities required in the SATURN booster development; and (2) determine design criteria for SATURN launch facilities.

In November 1958 a new objective, the development of a reliable, high performance booster to serve as the first stage for a multi-stage carrier vehicle capable of performing advanced space missions, was approved for the program.

The booster project was unofficially known for some time as JUNO V. An ARPA memorandum on 3 February 1959 made the name SATURN official for the project.

In March 1959, the SATURN system study, outlining various upper stage configurations and indicating the use of either an ATLAS or TITAN second stage would be acceptable, was submitted to ARPA. ABMA was informed by ARPA in May 1959 that a decision had been reached to use modified TITAN hardware for the SATURN second stage, and that the third stage should utilize the CENTAUR vehicle.

However, in July 1959, ARPA informed AOMC that all in-house and contract effort and other expenditures relating to the modified TITAN second stage be suspended unless the work was not directly connected with the stage diameter. This suspension was necessitated by a new requirement to relate SATURN second stage planning to other Department of Defense programs.

On 24 September 1959, ARPA submitted guidelines and requested a study be commenced immediately to determine the two best configurations for increasing SATURN capabilities for both military and NASA payloads.

As a result of the Presidential Order proposing transfer of ABMA from DOD to NASA, an interim agreement was made on 25 November 1959, between ARPA-NASA and the Department of Defense, which placed the technical direction of the SATURN program under NASA. ARPA was to retain administrative direction and committees, composed of representatives from ARPA, NASA, ABMA and the Air Force, were established to provide technical assistance to NASA.

On 7 December 1959 ARPA-NASA requested that AOMC prepare an engineering study for a SATURN configuration consisting of the clustered booster with a second stage of 4-20K lb thrust hydrogen/oxygen engines, and a third stage of 2-20K lb thrust hydrogen/oxygen engines. Results of this study were formally transmitted on 28 December 1959.

B. (S) New Objectives

A decision on the over-all SATURN upper stage configuration was finally achieved with the report from the SATURN Vehicle Team (Silverstein Committee) on 15 December 1959. This report recommended a long range development plan for the SATURN with use of all hydrogen/oxygen upper stages, the use of the C-1 configuration for the initial vehicles, and the development of a high thrust hydrogen/oxygen rocket engine. (See Appendix for the complete SATURN Vehicle Team Report)

C. (U) Major Subsystem Contractors

The Army Ordnance Missile Command has the responsibility of designing and developing the SATURN booster and system management for the complete vehicle. Consequently, most of the booster development effort has been reflected as an in-house activity.

Component and subassembly hardware for the ABMA designed guidance and control system, adapted from JUPITER, is being obtained from a variety of commercial concerns. Individual engines are being supplied by Rocketdyne, a division of North American Aviation, while the design and development of the booster recovery system is being accomplished by Cook Research Laboratories. Hayes Aircraft Corporation has a major part in the design and production of support equipment. The U. S. Corps of Engineers is in charge of all facilities construction for the SATURN project while the Army Transportation Corps is providing for all booster transportation requirements.

SATURN second stage development, formerly under contract with the Martin Company and Aerojet General Corporation, has been suspended in lieu of the program realignment to the C-1 configuration (See Appendix). The adoption of the C-1 configuration requires that the second stage contractor be redetermined. Negotiations for this purpose have been undertaken. It is known, however, that the second stage will employ Pratt and Whitney engines.

The SATURN third stage, an adaption from the standard CENTAUR, will be provided by Convair Astronautics, using engines provided by Pratt and Whitney Division of United Aircraft Corporation.

SECTION II. VEHICLE DEVELOPMENT

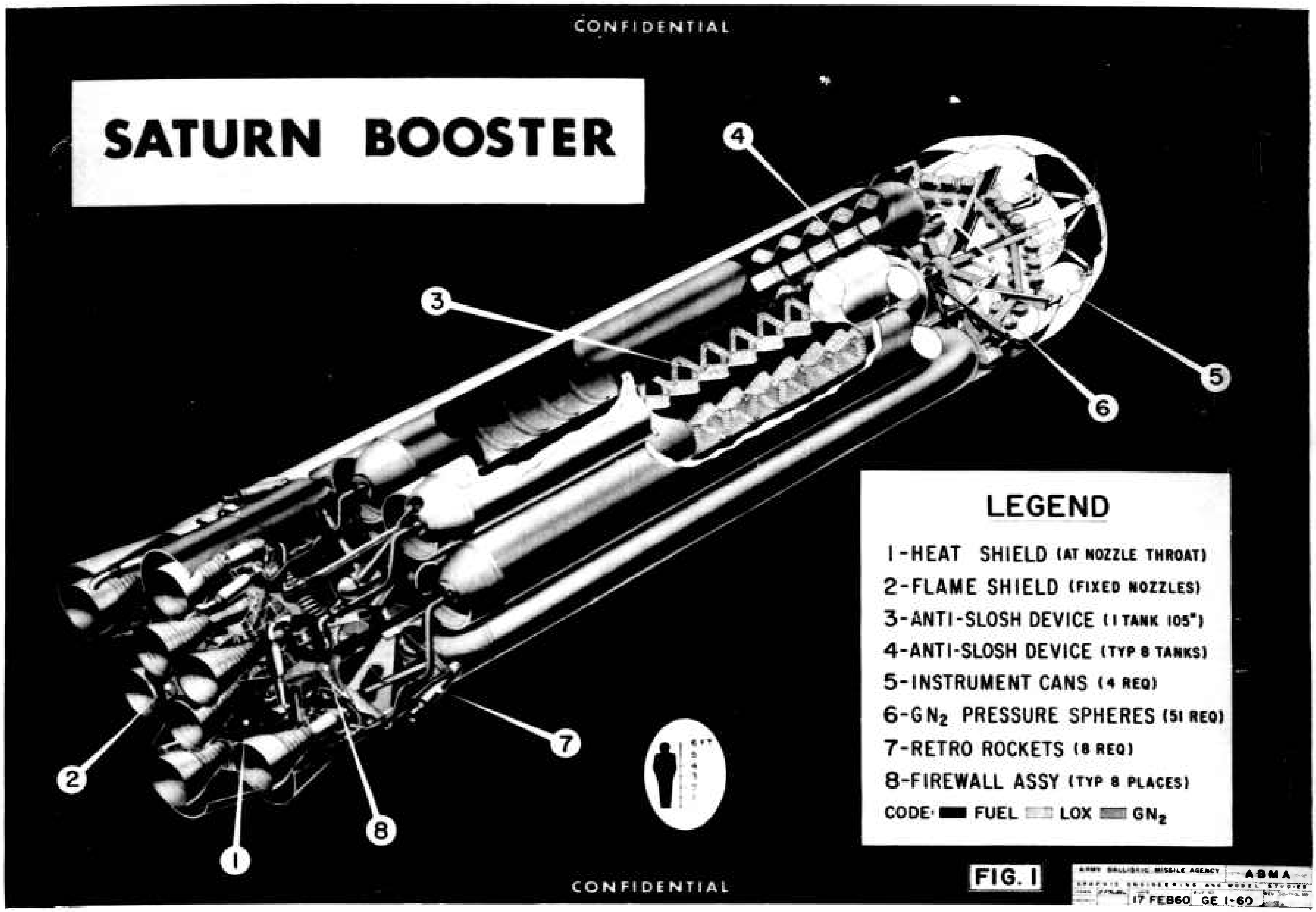
A. (U) General

In December 1959, the Silverstein Committee issued firm recommendations for the SATURN Program which called for continuing development of the SATURN with the "C" series of vehicles (see Appendix). ABMA immediately initiated studies on the newly proposed configuration under the assumption that final approval of the recommendations will be received from ARPA in the near future. Also, search for a qualified upper-stage contractor for the new configuration has started, and selection should be achieved by April 1960.

Prior to December, changing considerations for various upper-stage configurations resulted in reduction, and later, virtual elimination of engineering services provided by the Martin Company and Aerojet General Corporation in establishing the TITAN type upper stage. The C-1 approach also rendered the results of a large part of ABMA study efforts on a vehicle with TITAN type staging obsolete. Areas especially affected were the dynamics, control parameters, load and trajectory studies. However, a brief discussion is included for these areas to reflect the scope of the investigations.

Booster design continued without interruption throughout the report period, and the finalization of propulsion, control, and instrumentation system designs is being achieved in accordance with schedules determined by the funding limitations placed on the program. See Figure 1 for booster layout. Structural design, with exception of the upper-stage adapter, was not affected by upper-stage considerations and is nearing completion. Details of the adapter could not be defined until an upper-stage configuration was definitely determined; however, this will not delay the SA-1 schedules as the adapter portion can be attached to the main structure during final assembly.

Initial verification of the soundness of the clustering concept will be achieved starting in early April 1960 when the static test booster, SA-T, will commence hot firing build-up.


CONFIDENTIAL
SATURN BOOSTER
1
2
3
4
5
6
7
8
LEGEND
1-HEAT SHIELD (AT NOZZLE THROAT)
2-FLAME SHIELD (FIXED NOZZLES)
3-ANTI-SLOSH DEVICE (1 TANK 105")
4-ANTI-SLOSH DEVICE (TYP 8 TANKS)
5-INSTRUMENT CANS (4 REQ)
6-GN_2 PRESSURE SPHERES (51 REQ)
7-RETRO ROCKETS (8 REQ)
8-FIREWALL ASSY (TYP 8 PLACES)
CODE: FUEL LOX GN_2
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ABMA
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FIG. I

B. Dynamics and Control

1. (U) Introduction

During the course of the SATURN booster program, consideration has been given to various vehicles to serve as the upper stages. Each combination of booster and upper stages that has been proposed has required evaluation to determine its merit. The evaluation, in its early stages, involves the performance characteristics of the proposed configuration, payload capability, present or future availability, use of current vehicles to reduce design and tooling costs, the cost and time of producing a new vehicle versus modifying a design already in production, manufacturers' capabilities, reliability, time needed for testing, etc. Each new proposed configuration has required mathematical investigation of the dynamics and control stability and performance characteristics as a part of the evaluation. It is apparent that a large amount of mathematical work has been expended on proposals which for one reason or another were not accepted. Firm decisions, at the close of this report period, on upper stages not heretofore analyzed means that results from old studies are now obsolete and new studies must now be initiated on the complete new configurations.

2. (C) Aerodynamic Studies

a. Estimated Aerodynamic Data for Two- and Three-Stage SATURN Vehicles

Various missile configurations have been studied aerodynamically since inception of the SATURN program.

As the SATURN program progressed it reached the point where two basic configurations were being considered, namely, an ATLAS prototype and a TITAN prototype, each having a two- and three-stage configuration. Both missile configurations were analyzed by theoretical methods to provide data on the gradient of normal force coefficient, center of pressure, local gradient of normal force coefficient by linearized theory, and local grsdient of axial force coefficient by an empirical method.

b. Static Longitudinal Stability Investigations

An investigation has continued from the last report period on the static longitudinal stability characteristics of the SATURN vehicle since this is one of the many parameters which influence selection of an upper stage for the multistage configuration. The study established effects of upper stage length and upper stage shroud configuration on the static longitudinal stability characteristics of a complete SATURN. A 1.480-inch diameter model, comparable to a full scale diameter of 257 inches, was tested in the ABMA 14x14-inch trisonic wind tunnel over a Mach number range of 0.70 to 4.96 and angle of attack range of -4 to +12 degrees.

In order to conduct the over-all SATURN operational space vehicle program within the economic limitations, booster recovery and reuse of hardware has been considered mandatory. Valuable information can also be gained from study of recovered boosters. To use parachutes, as discussed in the section on recovery, it is necessary to predict booster motion prior to actuation of the recovery gear. Static longitudinal stability characteristics of the booster were necessary parameters for such prediction. Wind tunnel tests were conducted at the ABMA 14x14-inch trisonic wind tunnel to determine stability and drag characteristics at angles of attack ranging from 0 to 180 degrees and Mach numbers ranging from 0.69 to 4.96. Results of this investigation were provided to the Cook Research Laboratories for use in designing the booster recovery system.

c. Tail Section Cold Flow Investigation

The tail section cold flow studies mentioned in the report for the last period have been partially carried out. Base pressure measurements were conducted in the NASA, Lewis Research Center 8'x6' supersonic wind tunnel and the ABMA 14x14-inch trisonic wind tunnel, employing SATURN model configurations with cold supersonic air jets. Test Mach numbers extended from 0.09 to 1.97 in the NASA tunnel and 1.50 to 3.45 in the ABMA tunnel. Motor gimbal angles for these tests were fixed while model angles of attack remained at zero degrees. Jet static pressures ratios were varied at each test Mach number such that the model jet nozzles were both over-expanded ($P_e/P_s < 1$) and under-expanded ($P_e/P_s > 1$).

Several patterns of flow, particularly in and near the center region, can be predicted from the results of the tests, and the trends in these flow patterns will be fairly certain to exist on the full scale SATURN and will require attention to base heating protection. The data have indicated these flow patterns to be primarily a function of jet static pressure ratio (P_e/P_s). Two distinct flow patterns exist with transition from one to the other occurring at or near values of jet static pressure ratios of one. For jet static pressure ratios less than one the flow direction on the base is laterally toward the center, then between the inner engines and out downstream from the "center star" region; for jet static pressure ratios greater than one the process reverses, with resulting flow from downstream into the "center star" region with splashing on the base and subsequently moving outboard between the inner engines and toward the base rim. These flow phenomena are indicative of "center star" heat protection requirements which have been considered for SATURN design. The pressure within the center core region formed by jet impingement in close proximity to the inner four nozzle exits remains nearly constant at the higher jet static pressure ratios. These results indicate the values to be from 18 to 27 percent of computed nozzle exit static pressure (the lower value conforms to open base configuration whereas the higher value is associated with the "center star" region sealed).

Base pressure as well as temperature and heat flux results covering a wider range of test variables will be available after extended ABMA cold jet tests and NASA Lewis Research Center hot jet tests that were initiated in December 1959.

3. (C) Propellant and Structures Dynamics Studies

a. Propellant Sloshing Analysis

Past experience has shown that oscillations generated by sloshing of large fluid masses of propellants can have serious dynamic effects on a missile in flight. In a vehicle of the SATURN configuration, which is long and slender and has fluid propellants located far from the vehicle's center of gravity, the propellant sloshing effects can obviously not be ignored. The low structural bending mode frequencies may fall within the range of lower mode sloshing frequencies, making excitation in this range possible and dangerous. An investigation was undertaken to obtain close approximations to the total forces and moments which could result from propellant sloshing. Since sufficient data concerning the powered flight after separation of the upper stages from the booster were not available, the investigation took into consideration only those sloshing forces which might occur during first stage burning.

From the tests it was found that the first mode bending frequency is in the range of the second mode sloshing frequencies of all tanks of the cluster. The tests also disclosed that the forces and moments resulting from oscillations in the 120-inch diameter tanks (the upper stage configuration assumed for the study) are considerably greater than those resulting from the cluster booster. These large forces and moments of the 120-inch tanks in resonance areas are due mainly to the large amplitude of translation of the upper tanks and the location of these tanks above instead of below the entire missile's c.g. (center of gravity). The translation of the propellant tanks located above the missile's c.g. will be in opposite direction to the tanks translation below the missile's c.g. However, the tank rotation will be in the same direction whether the tank is above or below the missile's c.g. Thus it was seen that the rotational sloshing force will tend to reduce the force due to translation in tanks located below the missile's c.g., while for tanks above the missile's c.g. the translational and rotational forces will be additive, giving a larger total force.

Since oscillations in the 120-inch diameter upper stage tanks will produce the predominant sloshing forces and moments during first stage burning, introduction of considerable damping in the tanks was indicated as necessary. (With new upper stage versions a review of this will be necessary). Damping of oscillations in the 70-inch and 105-inch diameter cluster tanks during the early part of first stage burning is not so important as damping in the upper stages. However, damping in the cluster tanks becomes very important after about 40 seconds.

b. Damped Oscillations in a Connected Fluid System

A study was made of damped fluid oscillations in a system consisting of two tanks with different diameters connected with a pipeline. Certain energy losses were neglected in deriving the equations for the oscillation of the fluid system from the energy equation for instationary flow and the continuity equation for incompressible fluid. Solving the nonlinear differential equations by the perturbation method showed that the damped natural frequency differed little from the undamped natural frequency and that the logarithmic decrement is very small and decreases with flight time. This indicates that at later flight times the oscillation, if excited, is less damped which indicates in turn that an aperiodic motion at the beginning can change during flight into a damped oscillation. Bent pipelines and inlet energy losses will introduce additional damping beyond what the study shows. Since the flow in the pipeline is turbulent rather than laminar, higher damping can be expected from that condition also.

c. Effect of Propellant Oscillation on SATURN Roll Moment of Inertia

Propellant oscillations in tanks whose centers are not located on the roll axis of the vehicle may result in a considerable increase in the vehicle's roll moment of inertia. The magnitude of the propellant sloshing effects depends largely on the distance the tanks are located from the vehicle axis and on the roll control frequency. Since the clustered tanks of SATURN are in this category, an investigation was made to determine the increase in roll moment of inertia which might result from propellant oscillations in the outer tanks of the SATURN booster due to roll control. The propellant in the center tank, 105-inch diameter, was neglected since the only oscillations in this tank during roll are due to friction between the fluid and the tank wall.

It was found that the increase in the roll moment of inertia in the SATURN booster due to oscillations of the propellant during roll control will be small if a control frequency no greater than 0.6 cps is used. For a roll control frequency of 0.6 cps the maximum increase over the roll moment of inertia of a rigid propellant mass would amount to only 5% or 6%. However, if the roll control frequency should fall within the range of 0.8 cps to 1.2 cps during flight, a roll moment of inertia of up to 3.5 times that of a rigid fluid mass might occur. The indicated increase in the roll moment of inertia ratio during flight can be attributed to the fact that as the fluid height decreases during flight a greater percentage of the propellant in the tank participates in oscillations. The effective moment of inertia in roll decreases with a decrease in fluid mass or an increase in damping, but the greater percentage of propellant participating in the fluid oscillations makes this decrease in the effective roll moment of inertia less pronounced at later flight time.

4. (S) Control and Control Feedback Stability Analysis

a. First Stage Control Analysis

Early in the SATURN program a preliminary control study was made, based on a configuration in which the upper stage would be an ATLAS booster (without sustainer engine) or a TITAN (first stage with one engine). The study of controlling the missile with the four outboard engines and also for the case of failure of one of these outboard engines developed an approach and equations which will be useful in similar studies in the future.

b. Control Feed-back Stability Analysis

Control feed-back stability characteristics of the SATURN missile family, using various types of control systems, has been the subject of an extended investigation. The control frequency and the first four bending mode roots were investigated in detail throughout first stage powered flights with an ATLAS second stage. An Edcliff angle-of-attack meter was used for alpha control. The effect of arbitrary, artificial phase and attenuation upon stability were studied. A report indicated the problem consisted of two major parts: (1) computation of the frequencies and mode shape curves of the missile, performing a free bending oscillation without control system and exterior forces, and (2) investigation of the behavior of the missile under actual flight conditions. From the second part it was found that, with the gyros located in front of the second stage ATLAS, the first two bending mode roots have undesirable operating characteristics. If the gyros are placed on top of the first stage, the oscillation characteristic is within the range of phase and attenuation conditions that can be met with proper control system design.

A second report dealt with control flutter characteristics using accelerometers for alpha control. The major results were that, with the gyros and accelerometers located at the geometric missile center, the first two modes would have operating characteristics that are desirable whereas the third and fourth modes could be made desirable by the proper introduction of phase and attenuation into the control system. The inclusion of the elastic effects of the hydraulic servo system had no adverse effects upon missile stability.

A third report considered loading on the missile due to propellant sloshing, a factor not considered in the two earlier reports. Studied were the effects of:

(1) Sloshing propellant upon bending mode stability by considering the case of the propellant solidified and the case of the propellant free to oscillate.

(2) Different damping devices in the propellant tanks.

(3) Accelerometer location upon sloshing root stability. The conclusions were (1) that the SATURN missile, two stage version, will be free from control feed-back instability in all sloshing modes by using perforated accordion baffles in the booster and one or two 15-inch perforated single ring baffles at the surface of the second stage propellant tanks, (2) bending mode root stability is only slightly affected by the inclusion of sloshing forces, and (3) the location of the accelerometers has a negligible effect upon the stability of the sloshing roots.

5. (S) Trajectory and Performance Studies

a. Flight Performance for Ignition Failure of One Second Stage Engine

In considering upper stages for the SATURN, one proposal by The Martin Company was to use a single 200,000-lb thrust engine. Convair proposed an ATLAS configuration with two 200,000-lb thrust engines. Since there was an apparent difference of about 10% in payload capability, a comparison study was made of the reliability of the one- and two-engine second stages. Knowledge of the extent to which a given mission could be carried out in the event of failure of one of the pair of second stage engines of the SATURN-ATLAS version was required in order to make the comparison study. A mathematical analysis was made of the alternate tilt program which could be employed when failure of an engine was sensed, the effect on payload and orbit altitude, and the feasibility of flight termination.

b. Liftoff Motion Hazards of SATURN Vehicles

Problems which would exist at time of launch of the SATURN with modified ATLAS second stage were investigated:

(1) Possible contact with fuel lines immediately after liftoff.

(2) Available clearance for the missile to pass by the fixed support arms (there are four of these fixed support arms located 90° apart at the inboard engine outriggers).

(3) Possibility of missile contact with the umbilical tower, especially in case of motor failure at early flight time. The tower stands 213 feet above the base of the launch pad which is 27 feet above ground level. The long axis of the missile on the pad would be 645 inches away from the center line of the tower.

The disturbances assumed for this investigation were winds, thrust misalignments, and control motor failure. For control motor failure immediately after launch (assuming no other disturbances) it was determined that the missile would crash into the umbilical tower provided that the control motor that failed at launch was located nearest to the tower and was one of the two control motors lying on a straight line passing through the centerline of the tower.

For the case of the control motor failure at two seconds of flight time with assumptions of a 10 m/sec constant wind and a 1-degree thrust misalignment in unfavorable direction for each of the three remaining control motors, it was found that there would be sufficient clearance for the missile to pass the umbilical tower. With a 20 m/sec constant wind however, the missile would hit the tower at about 165 feet above the pad, but would pass the tower if motor failure was not until the 5th second of flight time.

Preventive measures were conceived to reduce the danger of missile contact with the tower. The first measure would be to arrange the missile launch position for all firings such that the cross of the four outboard motors is under 45° to the line between missile center and tower center. This would call for a rotation of the missile some seconds after liftoff into the desired azimuth, since tilting is scheduled to occur about a missile-fixed axis that would be unchanged from flight to flight. Another measure to avoid tower contact could be the introduction of a "clearance-cycle" into the gyro tilt program, to be arranged so the missile, at an early time, is forced to tilt into a direction away from the tower, independently of the tilt program desired for the mission of the flight. For the first three vehicles the problem will be solved by not using a tower.

c. <u>Trajectories for ATLAS and TITAN Versions of SATURN Vehicles 3 and 8</u>

Calculations were made on trajectories for SATURN missiles of two- and three-stage configuration. Since the information was needed before determination of the second and third stage configurations had been made, the analysis was based on the most probable configuration at the time. Two sets of trajectories were determined for flying the SATURN with a modified ATLAS 400K second stage and with a TITAN 200K second stage to an altitude of 200 km, circular velocity and a horizontal path. Two sets of trajectories were also developed for the same combinations with the addition of a Centaur 30K third stage to provide the velocity necessary for transfer from the 200 km perigee to a 24-hour orbit altitude (Restart of the CENTAUR third stage and the right amount of propellants left at apogee for the "kick" to reach circular velocity in the 24-hour orbit were assumed).

d. <u>Optimum Cutoff Angle for Lunar and Escape Missions</u>

In computing ballistic trajectories to the moon, one influential parameter to be considered is the angle the velocity vector makes with a radius vector from the center of the earth at injection. This parameter, which when referred to the multistaged propelled phase of a lunar trajectory becomes the path angle at cutoff, also influences the amount of payload which can be boosted to injection conditions. Another parameter influencing the ballistic lunar trajectory is the injection azimuth, and this also has its effect on the payload through the earth rotation velocity contribution. The relations of these parameters to payload and other characteristics of the trajectory for a typical lunar shot by an assumed SATURN missile with eight canted 188K engines in the first stage, a two-200K engine second stage and a two-15K engine third stage were reported.

C. Booster

1. Structure

a. (U) Introduction

Structural design progress of the SATURN booster was not significantly affected by the several changes considered for the upper stage configuration. The 220-inch diameter across the upper stage support pads was maintained, and the over-all booster structure proved sufficiently versatile to accept the resulting changes in loading and dynamic conditions. All structural drawings required for the completion of SA-T were released, and drawings for the SA-1 vehicle are progressing satisfactorily. Design concepts and general layouts of flight vehicle components such as shielding, shrouding, and the upper stage adapter structure were completed. Investigations were initiated to establish acceptable heat protective coatings required for application in the base areas where high heat fluxes are anticipated. Also, booster loading studies were continued for the changing configurations. A detailed description of the booster structure and general discussion of the studies and test programs accomplished are given in the following pages.

b. (U) Tail Section

The major structural part of the SATURN booster from the standpoint of size and weight is the tail section assembly (Fig. 2). It is located below the propellant tanks and consists of the following major components: tail section barrel, outrigger supports, tension and tie rods, actuator beams, thrust pads, retrorocket support brackets, heat shields, flame shield, fire wall, shrouding, and access chute.

The tail section assembly supports and structurally interconnects the tanks; it provides mountings for and transmits the thrust from the eight engines and supports and holds down the whole booster plus upper stages on the launch pad. Most of the tail section is fabricated from aluminum alloy 7075, heat treated to the T6 condition; however, some parts are made from 6061-T6 and 7079-T651 to facilitate fabrication. The 7075-T6 aluminum alloy was selected because of its high strength to weight ratio.

The barrel assembly is composed of a lower thrust ring assembly, upper thrust ring assembly, top ring segments with spacer, corrugated skin segments, longeron panels, shear plates, and plate assemblies.

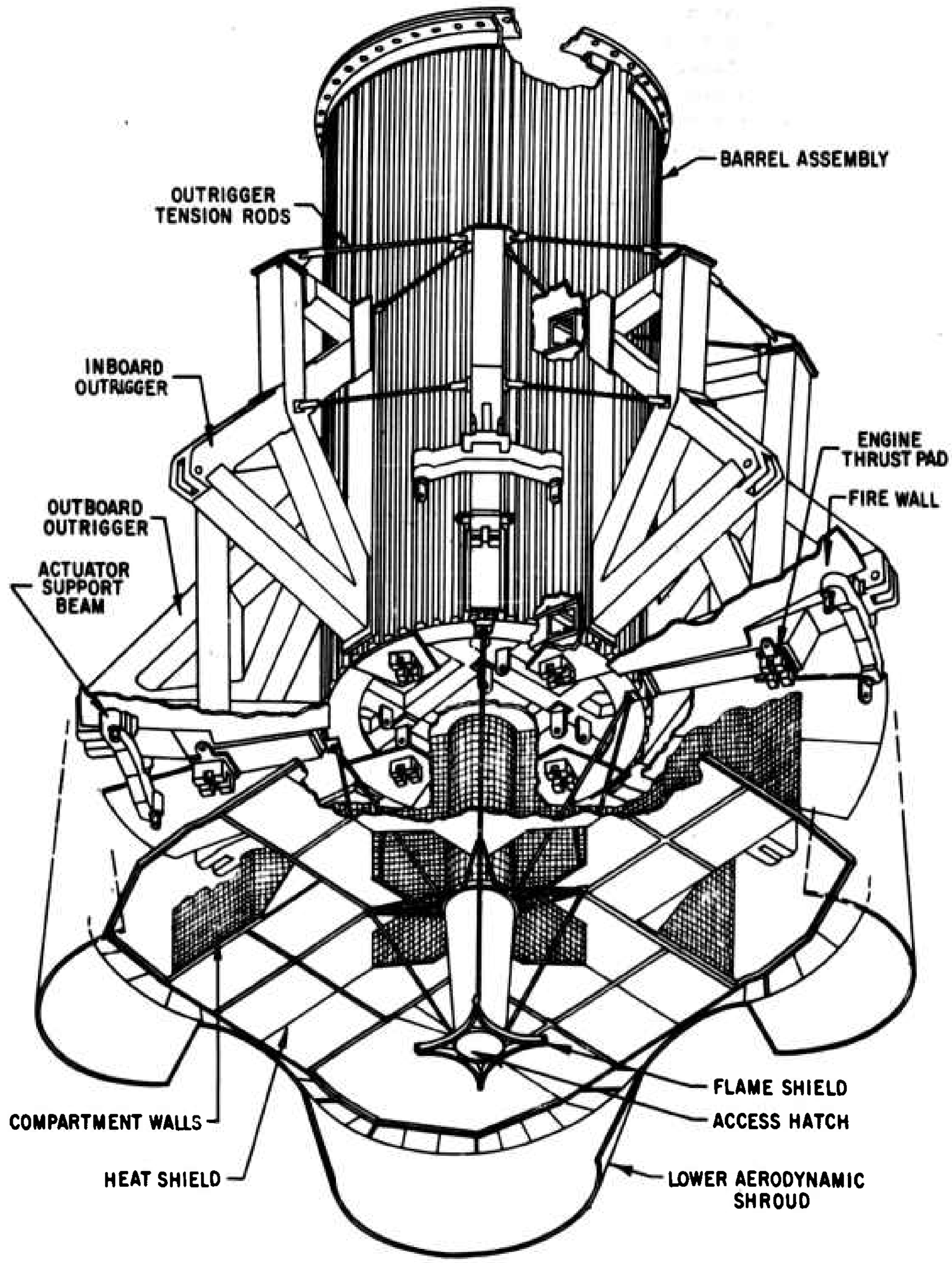


FIG. 2
TAIL SECTION ASSEMBLY
SATURN BOOSTER

Both the upper and lower thrust rings contain connection points for attaching the outriggers to the barrel assembly. Both rings are built-up box rings of 7075-T6 aluminum alloy. The lower ring is made from extruded channels and plate; the upper ring, from extruded angles and plate. The box ring is used in order to obtain the required section modulus for the loads imposed by the outrigger assemblies. The top ring also consists of 7075-T6 aluminum alloy segments and a spacer plate which provides both rigidity to the top of the barrel section and a method of connecting the tail section assembly to the 105-inch diameter lox container assembly. Neither angles nor spacer plate are bridged, but rather, are positioned at assembly so that a break in any part, including the angle segments on the center section, is 30° from any break in another part.

Twelve corrugated skin segments are used in making up the barrel assembly. The principle of corrugating the skin was first applied to the JUPITER and is used here because of its weight to strength ratio and the section modulus required for this particular area. The configuration of the corrugation was changed from the JUPITER to the SATURN to obtain even a higher section modulus and to accommodate the 7075-T6 aluminum alloy used. Large cutouts were made in the corrugated skin to permit routing of lox and fuel cross-over lines; however, they have been reinforced with plate assemblies to distribute load paths around the cutouts.

Longeron panels and shear plates are attached to the skin and to the upper and lower thrust rings at each outrigger location. Their purpose is to aid in stabilizing the thrust rings and to transfer the thrust loads into the corrugated skin. The longeron panels are attached inside the barrel section between the upper and lower thrust rings and the shear plates are mounted outside the barrel section between the outriggers and barrel skin. Material used for the longerons is 7075-T6 aluminum alloy. Alloy 7079-T651 is used for shear plates to eliminate warpage due to machining.

c. (U) Outriggers

The outriggers are the components of the tail section on which the outboard engines are mounted. They also support the 70-inch diameter tanks, transmit the thrust from the outboard engines to the barrel assembly, and provide the holddown and support points on the missile during test firings and launching. The outriggers are spaced 45° apart radially with the outrigger thrust assembly being located on fin positions.

A truss is considered the best design for the outriggers because of the multitank configuration of the booster, disassembly requirement imposed by transportation limitations, the better accessibility into the tail area, and economical fabrication. Material used in fabricating the outrigger is 7075-T6 alloy. It is used in both the outriggers and the connecting blocks located at the load transfer points between the outriggers and the barrel. The outriggers are stabilized with tension and tie rod assemblies. The tie rod assemblies are made from 4140 steel heat treated to 160,000-180,000 psi minimum tensile strength. This strength is needed because of the high loads imposed laterally on the outriggers by engine gimballing.

The outboard actuator attach-points are part of a welded beam structure attached perpendicularly to the outrigger assembly. The dummy actuator attach-points for the inboard engines are part of a cross-beam structure attached to the lower thrust ring of the barrel assembly. The thrust pads which mate to the gimbal blocks of the engines are machined from 7079-T651 alloy to reduce weight. The inboard thrust pads are mounted on the lower thrust ring; the outboard pads are mounted on the outrigger thrust assembly. The pads are positioned to permit inboard engine mounting on a 32-inch radius and the outboard engine mounting on a 95-inch radius.

Sockets on top of each outrigger, allowing quick assembly or disassembly of tanks, receive ball joints attached at two points on the periphery of each propellant tank. Bracketry for supporting retrorockets used in the booster recovery program is also attached to each outrigger. These brackets are fabricated from corrosion resistant steel.

d. (U) Shielding

The heat shield assembly (Fig. 2) is located at station 54 and is supported longitudinally by the lower ring frame in the barrel assembly and the aerodynamic shroud. The heat shield has a two-fold purpose: (1) it serves as a lateral support for the shroud; (2) it protects the structure, engines, and all existing equipment forward of station 54 from exhaust flame feed-back and expected high heat flux.

The support structure and removable panels of the heat shield are fabricated from 410 stainless steel sheet. The doors have angles attached for stiffeners. The side facing the heat source is a smooth surface to eliminate the possibility of protruding objects from burning off and to allow application of 1/4-inch thickness of PR-1910, a silicone rubber compound, or approved equal for heat protection.

The firewall, a horizontal partition located beneath the barrel assembly and between outriggers at the lower channel assembly, acts as a thermal barrier to protect the main structure against possible engine fires and heat radiated through the heat shield. It also supports the engine compartment curtains at their upper connections.

The firewall is made of spacemetal sandwich material because of its high strength to weight ratio and thermal characteristics. The spacemetal sandwich consists of facing sheets .008 inches thick and a corrugated core .002 inches thick spotwelded to the facing sheets.

The sides of the corrugations contain stiffening beads to provide an increase in strength. The material is type 301 stainless steel, full hard. The firewall between any two outriggers consists of four individual flat panels mounted to the forward side of their beam support members, which in turn, are connected to the outriggers and along the periphery of the barrel skin. A 3-inch clearance exists between the firewall and inside mold line of the periphery of the shroud to provide for differential movement between structure and shroud. The supporting structure consists of channels formed from type 17-7 stainless steel sheet.

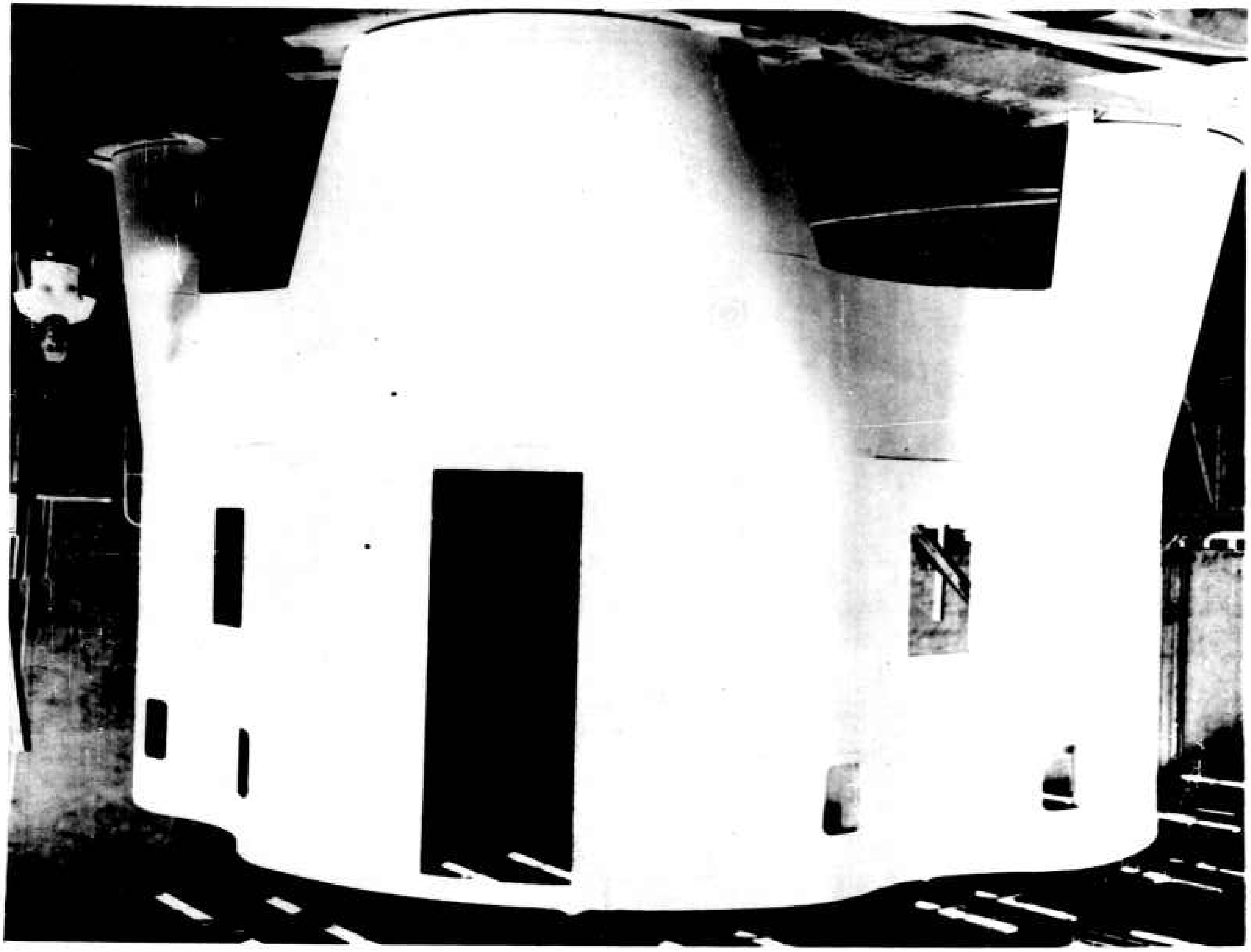
The spacemetal is provided with inserts at locations where fasteners are required. Reinforcements are provided where fuel, lox, and other lines extend through the firewall. Openings through and at the periphery of the firewall are closed with asbestos insulation, or fiberglass cloth, and areas on which personnel stand or walk during assembly are surfaced with nonskid material. The firewall will support an ultimate load of 500 pounds in any octant and, at 300°F, a uniform differential pressure of ±1.0 psi.

A substantial flow back of hot gases is expected in the area between the four inboard engines. To prevent this flow back and consequent high heating of the base area, a star-shaped flame shield is placed between these engines at the chamber exit plane. The heated side of the flame shield consists of a 1.3-inch thick CT-301 slab backed on the cold side by a .125 thick 410 stainless steel plate. This assembly is supported by stainless steel tubing connected to the heat shield. To allow engine movement during operation, a small gap is maintained between the flame shield and engine chambers. This gap is sealed with several layers of flexible asbestos cloth reinforced with inconel wire. A circular portion of the flame shield is removable to allow accessability of personnel. A conical frustrum chute leads upward from this opening into the engine turbopump area. It allows last minute access into the tail area and emergency exit if required.

e. (U) Shrouding

The shroud configuration (Fig. 3) was designed to maintain the smallest possible cross sectional area of the missile tail and yet enclose and provide clearance for all parts requiring protection. Special provisions were incorporated to avoid flat surfaces, reducing the possibility of flutter due to vibration. Efforts were also made to avoid any warped surfaces or surfaces containing compound bends to eliminate the necessity for performing the skins. Consequently, the shroud contains only cylindrical and conical shapes which afford the simplest means of fabricating to the prescribed contour.

The shrouds are supported vertically by the aft edges of the 70-inch diameter tank skirts. Support by the outriggers was specifically avoided because of anticipated outrigger deflections. Intercostal continuous ring frames are maintained about the circumference of the



missile to resist radial forces caused by air loads. Each shroud panel is composed of a skin supported by a framework of stringers and ring frames.

The shroud panels above station 95.5 are made of .032-inch thick aluminum alloy. This metal was selected because it approached an optimum strength to weight ratio at 260°F, which is the expected temperature in this area.

The aft ring is made of stainless steel. Steel is used so that the aft ring will resist loads proportionately with the forward ring of the lower shrouding to which it is attached.

Stainless steel was also selected for the lower shroud panels because of the excessively high temperatures in this area from the rocket blast and possible fires in the engine compartment. Due to the high strength and formability required for the ring frames and stringers, it is necessary to heat treat the steel after forming. Complications were encountered during the heat treating because of the warping of the material, resulting in a loss of required tolerances. An effort to minimize this problem led to the selection of 17-7 PH type stainless steel for the panel framework which makes use of precipitation harding and results in less warping during heat treating. 17-7 PH also has a low coefficient of expansion in the heat treated condition and is expected to cause a minimum amount of stresses or warping due to elevated temperatures during flight.

Since the lower shroud panel skins are not as highly stressed as the framework, heat treated material is not required. An option of 17-7 PH annealed, type 431 or type 410 was considered acceptable for lower shroud panel skins; however, type 410 stainless steel is preferred, because its coefficient of expansion is nearer that of 17-7 PH in the heat treated condition and is considerably less expensive. Application of PR 1910, or approved equal, to the inside of the shroud would considerably decrease the maximum temperatures to be expected.

The lower outboard panels are composed of three different conical shapes with the stringers being placed on straight line elements of the cones. These panels must be removed to place the missile in the test stand. Final assembly of the panel to the missile must be completed after testing.

The lower inboard shroud is composed of two different cones with stringers placed on the straight line elements of the cone as in the lower outboard panel. The aft edge of this panel is indented toward the center of the missile to allow clearance of the launcher's stationary support arms at liftoff. The panel also serves as the vertical support for the outer portion of the heat shield. This panel, like the lower outboard, also has a portion of the heat shield attached to it.

The upper inboard panel consists of an .032-inch aluminum skin supported by an arrangement of stringers and arced frames. This panel contains a quick cable release mechanism, a man-size access door, and two holes for the lateral tension ties which stabilize the missile on the static test stand. The portion of the panel immediately below the outrigger is removable in order to assemble the panel to the missile without removing the handling ring.

The construction of the upper outboard panel is of similar construction to that of the upper inboard panel except that it has only one cutout, 30 inches wide, for installation of the retrorockets. A separate small panel will be designed to fit closely around the retrorocket and fill the larger cutout after the rockets have been installed. The portion of this shrouding immediately below the outrigger will also be removable in order to assemble the shroud to the missile without removing the handling ring. Conical fairings are used to fair both the upper outboard and upper inboard panels into the 70-inch daimeter tank skirts.

f. (U) Propellant Tanks

Basically, all of the tanks are of a semimonocoque construction, out of skin segments approximately 58 inches in width, with hemispherical heads at both ends. The skin thicknesses vary according to design requirements which are based on container pressures as well as longitudinal loads.

The material used for the skin is 5456-H24 aluminum alloy. It has a low weight to strength ratio, favorable mechanical properties, and gives better elongation in the welded condition.

The heads are spun into hemispherical shape out of 5086-H34 aluminum alloy. This alloy has mechanical properties better than those of 5052 alloy but inferior to 5456; however, the latter cannot be used because it is difficult to spin form. The spherical shape for the heads was adopted to simplify fabrication.

Mounted in each container is perforated, accordion type, anti-slosh devices similar to those used successfully in the JUPITER. Use of these simple (but high total weight) devices was adopted because extensive test programs to prove the proper functioning of more complicated (but lighter) approaches could not be accomplished in time.

An interflow pipe manifold is provided to connect to each of the five lox containers at the bottom. The lox interflow manifold is 12 inches in diameter and forms an integral system of 6061-T6 pipe, with a .125-inch wall thickness, welded together. Machined flanges are provided at points where connection to the containers and fill and replenishing lines are required. The fuel interflow manifold interconnects the four 70-inch fuel tanks. It is also fabricated of 6061-T6 material; however, the pipes are only 10 inches in diameter. Machined mating surfaces are also provided in the fuel manifold.

g. (U) The Upper Stage Adapter Section

The upper stage adapter and spider beam assembly is a beam and tube truss, located at the forward end of the containers, which holds the cluster rigidly together and provides support points to the upper stage (Fig. 4).

The spider beam is assembled of eight 20x8-inch 7075-T6 extruded I-beams with 1/2 inch flanges and 5/16 inch webs radiating outward from a single hub. The assembly is done with mechanical fasteners to avoid warp and stress that could be caused by welding. The forward end of each lox container is fastened rigidly to the spider beam to absorb compression loads from the upper stages. The fuel containers, 2.5 inches shorter than the lox tanks, are connected by sliding dowel pins which give lateral but no longitudinal support. This design allows the lox containers to shrink longitudinally when the lox is loaded. The upper stage adapter proper is a tubular truss system attached to the spider beam. It provides a 220-inch diameter mounting pad 90 inches forward of the spider beam. Materials used in the adapter are 7075-T6 aluminum alloy and 4130 steel. The 90-inch dimension was considered adequate for containing a recovery package and allowing an adequately sloping adapter shroud.

The aerodynamic shroud between first and second stage was subcontracted for design to North American Aviation. It will be fabricated of spacemetal (301 stainless steel sandwich previously described for the firewall). The shroud is connected only to the upper stage adapter and is cantilevered from the eight points on the adapter. Ring frames at the mating edge, the break, and the lower periphery strengthen the adapter against aerodynamic loading. Connecting points to the spider beam are not provided in order to allow for deflections and relative motions in that area. The main purpose of the shroud is to serve as an aerodynamic fairing between the cylindrical second stage and the clustered first stage.

h. Load Analyses

(1) (U) Introduction

Numerous studies have been made to determine load and bending characteristics for the various SATURN vehicle configurations considered. Most of these studies were invalidated with the acceptance of 220-inch diameter, all-hydrogen upper stages, and presently new studies are being completed to provide the changed load and bending information. Summarized below are the results of three studies which present data on the loads, bending moments and deflections of the SATURN vehicle. The results of the first study presented are obsolete, but since new information is not yet available, the data are presented in order to reflect the general order of magnitude estimates for wind forces and resulting bending moments to be encountered by SATURN type vehicles.

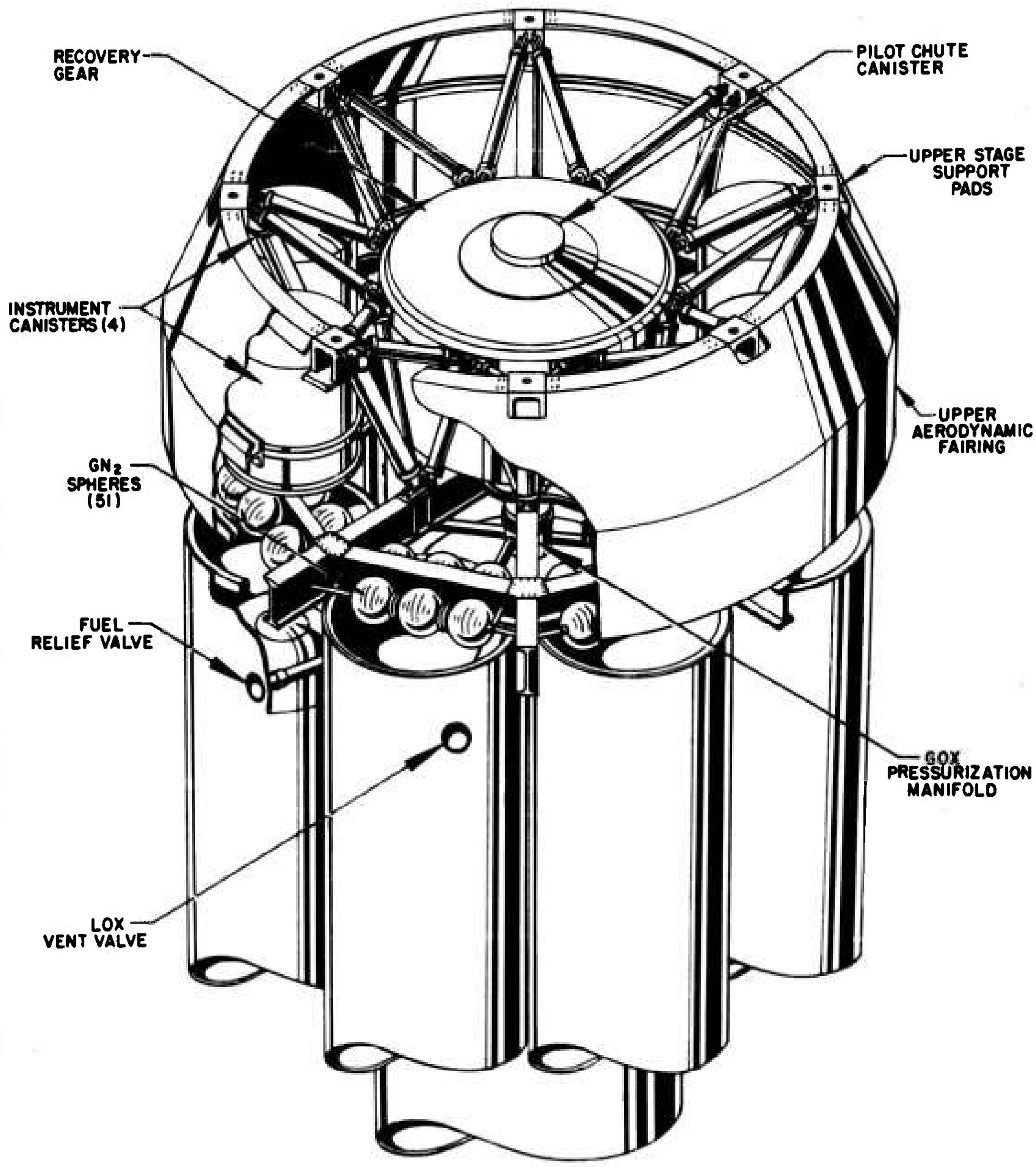


FIG 4
UPPER STAGE ADAPTER AREA
SATURN BOOSTER

(2) (S) Bending Moments and Deflections Due to Wind While Missile is Installed on the Launcher

The bending moments and deflections imposed on the erected SATURN vehicle by winds were studied for the following configuration:

Over-all length	2,509 inches
2nd stage diameter	160 inches
3rd stage diameter	120 inches
Payload weight	20,000 pounds
Total vehicle weight, fueled	1,200,000 pounds

The data obtained for this particular configuration is presented as it has one of the more severe characteristics of the various configurations studied. A secondary moment due to the weight of the vehicle and approximate launcher arm deflections were included in the results.

The assumptions made in this study include: value of air density as .002378 pound $\frac{sec^2}{ft^4}$, a steady wind gradient relative to the 10-foot level according to MIL-STD-210, and constant drag coefficients of eight for the clustered tank section and four for the remaining portions. Vortex shedding motions were neglected, however, the bending moments and deflections include a factor of 1.5 to account for increased loads due to vortex shedding. The tie-down points at station 123 were assumed to be 39 feet above the ground. Stiffness values (EI) were assumed as 26.6x10 in-lb in^2 for the booster, 31.4x10 in from station 889 to 1200, 33.2x10 in from station 1200 to 1740, 81.0x10 in from station 1740 to 2264 and 30.0x10 in from station 2264 to 2480.

Figure 5 presents a plot of the bending moment (M_b) at station 123 for wind velocities from 0 to 70 knots. Maximum bending moment reached is approximately 60x10^6 in-lb.

Figure 6 presents the ratio of the bending moment (M_s) at any missile station to the bending moment (M_b) at station 123, and a plot of the missile deflection resulting from a 20-meter/sec wind. The maximum deflection under these conditions can be seen as 10.3 inches at the tip. Contributing to this are the wind moment at 76.3%, moment due to weight displacement at 9.7%, and support arm deflection at 14%. The M_s/M_b plot, in connection with Figure 6, can be used to determine the bending moments at any missile stations for various wind velocities.

FIG. 5. BENDING MOMENT AND DEFLECTION

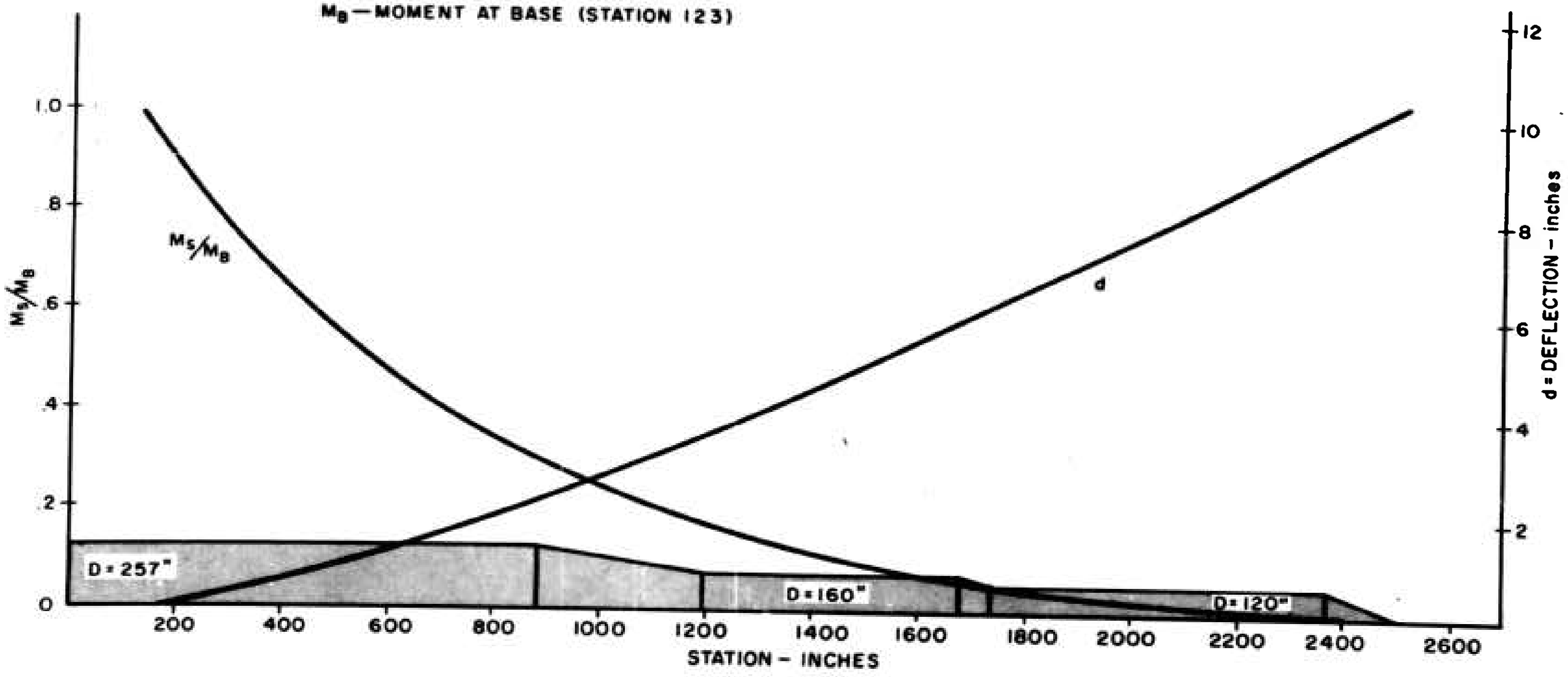
ON LAUNCHER

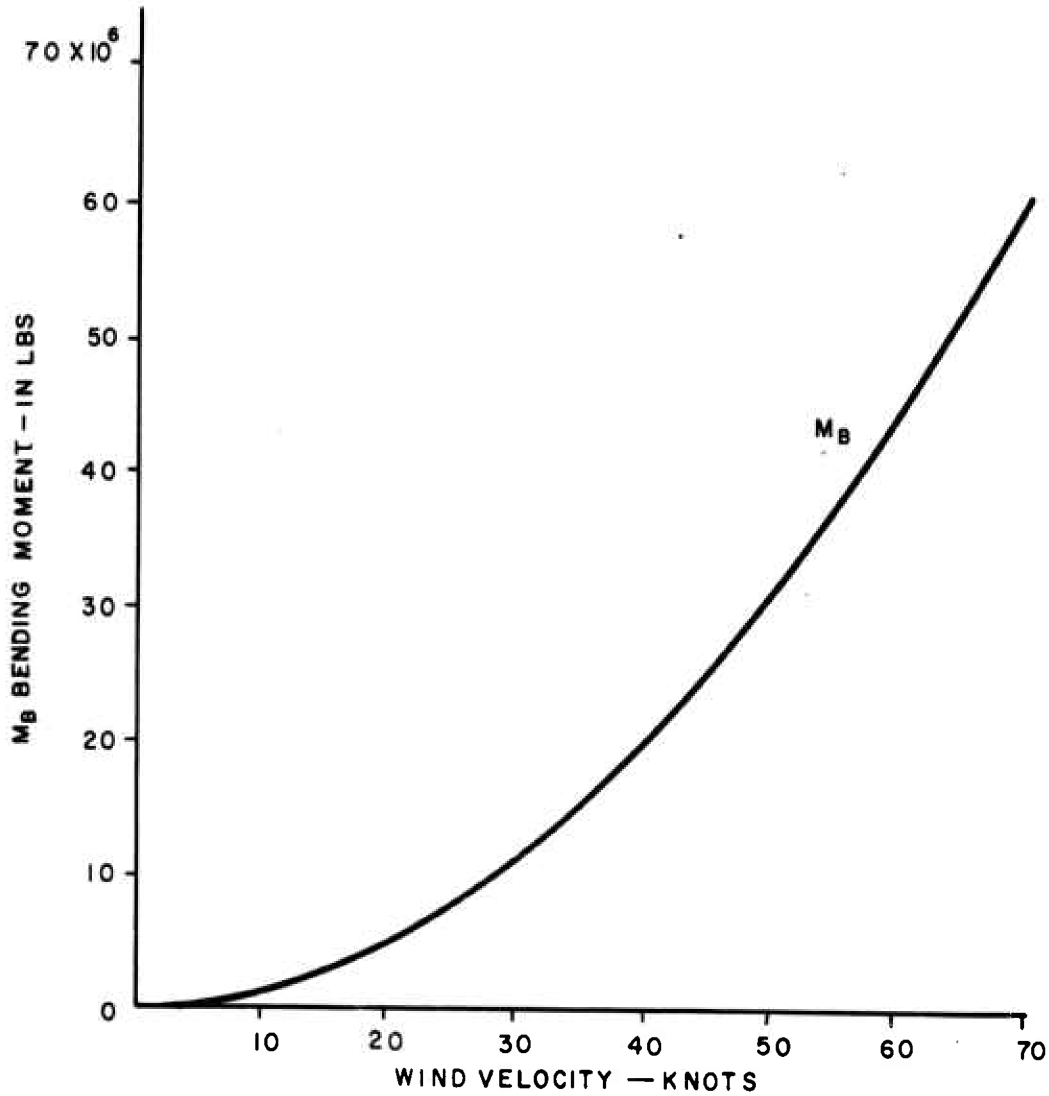
RATIO OF MOMENTS AND DEFLECTION VS STATION

20 M/S WIND USED IN DEFLECTION - CALCULATIONS

M_S — MOMENT AT ANY MISSILE STATION

M_B — MOMENT AT BASE (STATION 123)





ON LAUNCHER

FIG. 6. BENDING MOMENT VS WIND VELOCITY

(3) (C) Maximum Loads During Ascent for a Four-Stage Vehicle

The maximum loads for the powered phase of a 4-stage, lox-H_2 upper stage vehicle with a length of 2614 inches and a liftoff weight of 1,152,962 pounds were investigated. Results obtained for the time t = 71 seconds where maximum aerodynamic pressure conditions are encountered are presented in Figure 7. The second and third stage configurations with a 220-inch diameter, and a fourth stage diameter of 120 inches were used. Individual engine thrust level for this case would be 188K. The trajectory conditions for this configuration at t = 71 sec would be

q = 4.85 psi	β = 4.35°
M = 1.5	η = .153 g
Alt = 11.2 km	ϵ = .0000906 in -1
α = 4.3°	λ = 1.975 g's

The analysis considered the effect of the 8 outer tanks on the remainder of the missile. The shear, bending moment and longitudinal force diagrams for the total missile have the reactions of the forces due to these outer tank imposed at station 189 for both fuel and lox tanks, at station 803.446 for the lox tanks and at station 800.446 for the fuel tanks. The loads presented are due to aerodynamic and inertia effects only as internal pressure effects were not considered.

At t = 71 seconds, the maximum bending moment of -20.96×10^6 inch-lb is experienced at station 1160. The maximum moment on the booster is -18.02×10^6 inch-lb at station 882, and maximum longitudinal force was found to be -1.584×10^6 lb experienced at station 162.4.

(4) (S) Dive Phase Loads on Booster

To aid the studies of the booster recovery program, dive phase loads on the booster were calculated at maximum dynamic pressure and at the expected altitude of parachute opening. These results will vary somewhat with change in trajectories. Due to the numerous attitudes that the booster might re-enter, the aerodynamic data used in this report was calculated from data covering the re-entry cases of α = 0° and α = 90°. At time of parachute opening the booster angles of attack could of course be other than 0° or 90°, and in this study values of 90°, 135°, 170°, and 180° were considered.

The assumptions made in the study included a booster weight of 119,000 lb and a main chute cluster force of 322,000 lb at deployment. The maximum pressure for the booster with a constant angle of attack of 0° is 13 psi while the constant 90° angle of attack gives 2.76 psi. Since it is believed that neither of these cases will hold true, a maximum dynamic pressure of 7 psi was selected to approximate the case.

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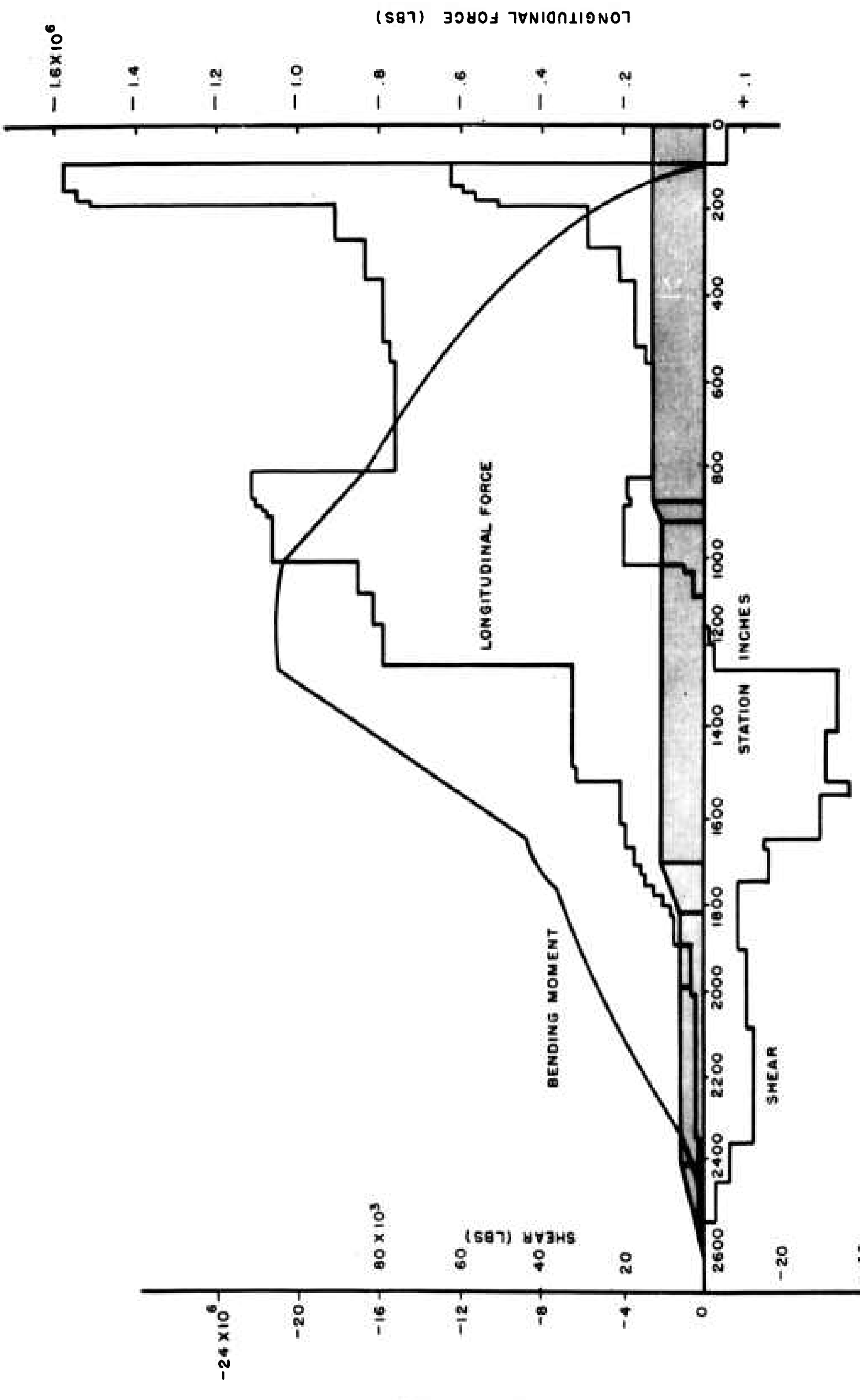

ASCENDING PHASE
BENDING MOMENT, LONGITUDINAL FORCE AND SHEAR VS STATION

FIG. 7

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Data for only the 90° angle of attack at both maximum g and parachute opening are presented here as they reflect the worst conditions encountered.

The results (Fig. 8) reflect that the maximum longitudinal load factor is less than that known for the ascent phase; however, some stations are in tension, whereas they are in compression during ascent. Also, the largest shear and bending moments imposed on the booster occur during maximum dynamic pressure rather than at the time of main parachute opening.

i. (C) Weight, Center of Gravity and Moment of Inertia

The weight and center of gravity of the SATURN booster configuration have been revised according to the most recent design information (Table 1). The booster dry weight is presently estimated to approach 85,000 lb. This reflects a considerable increase over early estimates. It should be considered, however, that the earlier estimates were based on very preliminary information, whereas the current figure is based on comparatively detailed design information. The 85,000 lb dry weight reflects actually a decrease in the booster's structural weight. However, the weight estimates of cables and miscellaneous hardware components have increased enough to make up the difference. An aft shift in the booster's center of gravity was encountered due to the decrease in weight of the upper stage adapter section. It should be noted that the weight of the recovery package, recovery rockets and rocket brackets is not included in the 85,000 lb.

The weight breakdown of the booster at separation is given including a 2% lox reserve; however, aeroballistic flight reserve allowances have not been made (Table 2). Also, no allowance has been made for the possible additional lox and fuel trapped in the containers as a result of one engine out operation. A booster schematic indicating stations and moist points is shown in Figure 9.

j. (U) Structural and Materials Testing

(1) Structural

The over-all structural test program was briefly described in the first semiannual technical report. The objectives of and the approach to the test program have not been changed. The primary difficulty in this area has been that early restrictions in funding have prohibited timely delivery of test hardware. The only component on which testing has been initiated is a shortened 70-inch

FIG. 8A
DIVE PHASE
BENDING MOMENT AND TOTAL NORMAL LOAD FACTOR VS STATION
$\alpha = 90°$, $\eta = 20.32$ g's, $e = -.0156$ in^{-1}, $q = 7.0$ psi

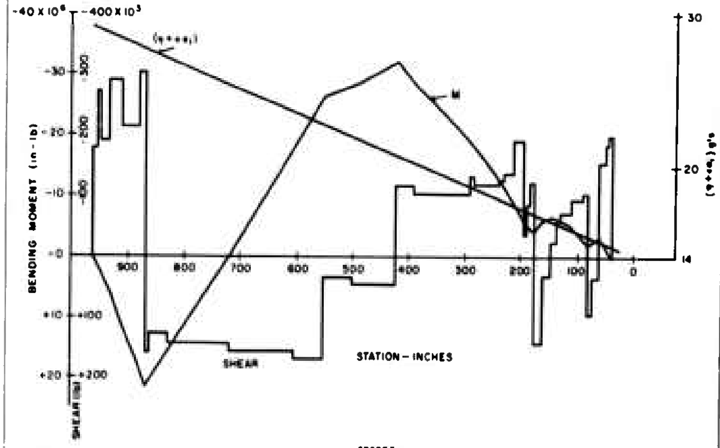

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FIG. 8B
DIVE PHASE
BENDING MOMENT AND TOTAL NORMAL LOAD FACTOR VERSUS STATION
$\alpha = 90°$, $\eta = 7.64$ g's, $e = -.0243$ in^{-1}, $q = 3.0$ psi
CHUTE FORCE = 322,000 lbs

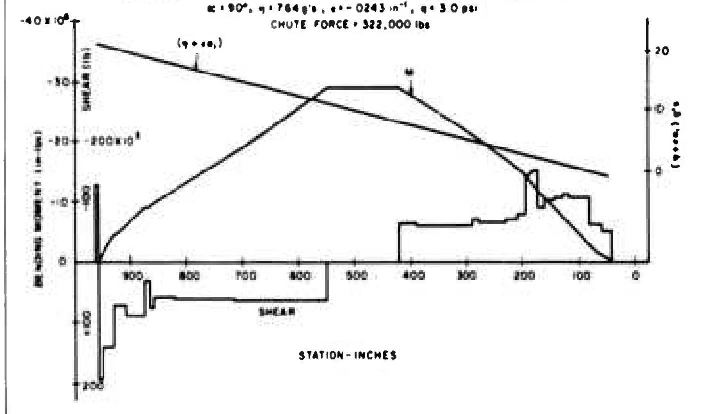

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TABLE 1

SATURN BOOSTER PRELIMINARY WEIGHT BREAKDOWN

		WEIGHT (LB)
Structure		53,500
Propulsion System		23,000
Guidance and Control and Measuring Equipment		6,000
Contingency		2,500
	TOTAL	85,000

DETAILED BREAKDOWN

STRUCTURE

A.	105 Inch Diameter Lox Container Section	6,175
B.	70 Inch Diameter Lox Container Section No. 1	2,900
C.	70 Inch Diameter Lox Container Section No. 2	2,900
D.	70 Inch Diameter Lox Container Section No. 3	2,900
E.	70 Inch Diameter Lox Container Section No. 4	2,900
F.	70 Inch Diameter Lox Container Section No. 1	1,750
G.	70 Inch Diameter Lox Container Section No. 2	1,750
H.	70 Inch Diameter Lox Container Section No. 3	1,750
I.	70 Inch Diameter Lox Container Section No. 4	1,750
J.	Tail Section Assembly	10,900
K.	2nd Stage Adapter Section	5,675
L.	2nd Stage Adapter Shroud	2,500
M.	Engine Compartments	160
N.	Fire Wall	440
O.	Heat Shield and Support	1,467
P.	Flame Shield and Support	325
Q.	Tail Shroud and Heat Protection	2,791
R.	Instrument Canister (4)	600

TABLE 1 (Continued)

S.	Cable Ducts (4)	100
T.	Antislosh Devices	2,000
U.	Ice and Rain Shield	350
V.	Pain and Corrosion Protection	500
W.	Miscellaneous Brackets	917
	TOTAL	53,500
PROPULSION SYSTEM		
A.	Rocket Engine System	13,492
B.	High Pressure Supply System (52 cu ft)	1,800
C.	Suction Lines (Including Flow Meters)	4,676
D.	Tank Interconnecting Lines	599
E.	Vent Lines and Tubing	1,000
F.	Fire Fighting System	1,000
G.	Purge System	100
H.	Control Supply and Gear Box Press ($1\frac{1}{2}$ cu ft)	75
I.	Miscellaneous	258
	TOTAL	23,000
GUIDANCE AND CONTROL AND MEASURING EQUIPMENT		
A.	Instrument Panels (4)	200
B.	Guidance and Control System	310
C.	Power Supply System	265
D.	Electrical Network System	2,584
E.	Measuring Network System	1,547
F.	Telemetry System	290
G.	Range Safety and Tracking	394
H.	Preflight Cooling Ducts	100
I.	Air-Bearing System	29
J.	Mounting Brackets and Miscellaneous	281
	TOTAL	6,000

TABLE 2

SATURN BOOSTER AT SEPARATION WEIGHT SUMMARY

	WEIGHT (LB)
Dry Booster	85,000
Recovery Rocket Attachment Brackets	500
Recovery Rockets	4,000
Recovery Package	7,000
Trapped Propellants*	
1. Gox in Lox Containers	3,900
2. Wet Weight in Engines	1,410
3. Suction Lines	6,422
4. Interconnecting Lines	2,833
5. Heat Exchangers	352
6. Miscellaneous Lines	343
7. Hydraulic Actuator Fluid	30
8. Nitrogen in Lox Ullage	1,030
9. Nitrogen in Fuel Ullage	100
10. Pressurizing Nitrogen	728
2% Lox Reserve - Fuel Depletion	
1. Reserve at Bottom of Containers	9,022
2. Reserve at Top of Containers	9,022

*Does not include lox and fuel trapped in the event of one engine out.

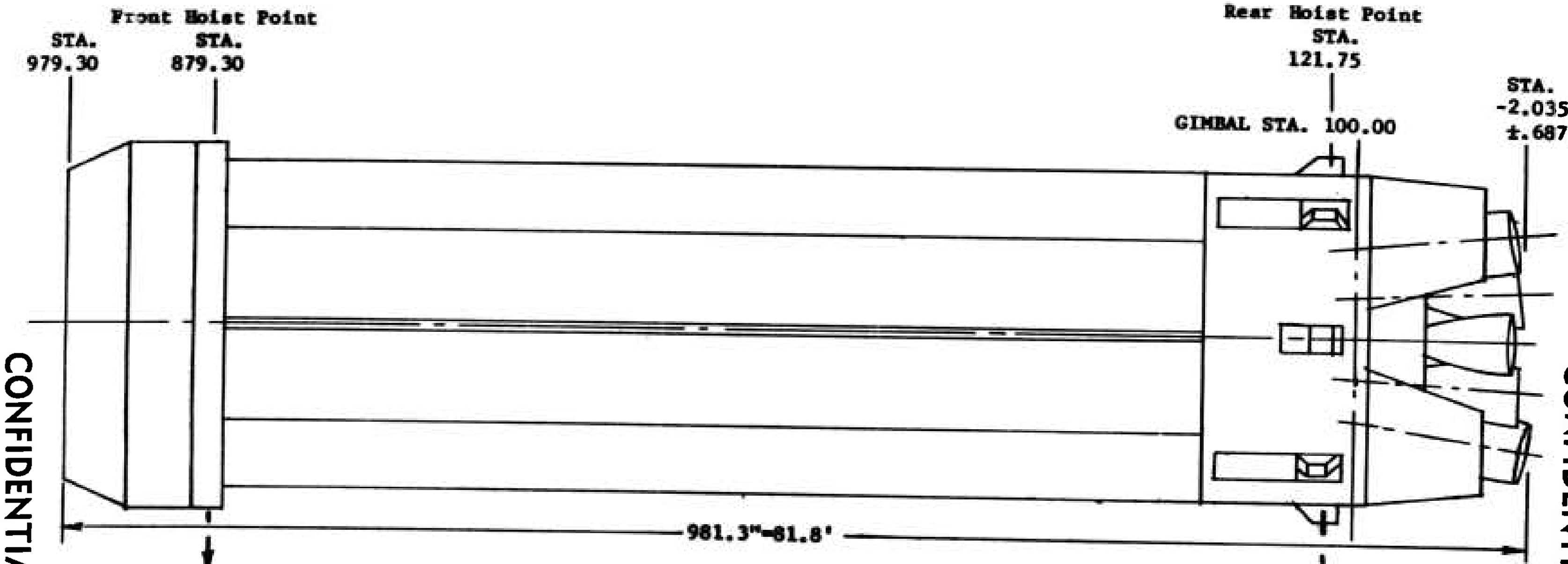


FIG. 9

SATURN BOOSTER SA-1 PRELIMINARY MASS CHARACTERISTICS

ITEM	WEIGHT (LB)	C. G. STATION	MASS MOMENT OF INERTIA KgMSec^2	
			PITCH	ROLL
Dry Booster	85,000	390	261,400	14,500
Dry Booster With Recovery Rocket Brackets	85,500	389	262,300	14,700
A. Front Hoist Point Load	30,165	879.30	-	-
B. Rear Hoist Point Load	55,335	121.75	-	-
Dry Booster With Recovery Package and Rockets	96,500	419	328,400	15,700
Booster at Separation				
A. With residual lox at bottom of tanks	122,720	387	360,400*	19,100*
B. With residual lox at top of tanks	122,720	431	401,700*	19,100*

*Assuming the inertia of the propellant as a point mass.

container. It has been completely instrumented and is being subjected to load testing.

In preparation for structural testing of the tail section and a shortened propellant container section, the design of a structural test tower was completed. Installation of the tower by the contractor was accomplished late in December. The tower will allow simulation of the static firing condition, the launch condition, the maximum dynamic pressure condition, and the load conditions just before cutoff. The completed tower is shown in Figure 10.

(2) Materials

The design of a closed base in a missile inherently results in significant heating of the base. In the case of clustered engines in vehicles such as SATURN, this heating condition is aggravated. Two significantly different areas in the SATURN base require heat protection. The first of these is the flameshield located between the four inboard engines at essentially the base plane of the vehicle. The second area includes the heat shield located at the rocket nozzle throat, and such other areas as the inboard side of the outboard rocket engine expansion cones, some portions of the shroud, and a few structural members.

(a) Flameshield

A theoretical investigation of the base heating in the star region of the SATURN booster, made in conjunction with actual experience of Convair and Lockheed on the ATLAS and POLARIS missiles, indicated severe heating of this region would occur. The conditions of heating rates between 40 and 120 BTU/ft^2-sec and gas velocities up to sonic were presented. It was indicated also that the peak heating conditions would not occur at cutoff but rather at approximately 30% of mainstage and then remain essentially constant.

In selecting a material to protect the star region consideration was given to the concepts of (1) heat-sink, (2) radiation, and (3) ablation. The heat-sink concept was eliminated because of the weight penalty attached. The radiation concept, i.e., the use of a highly polished surface of a material with high reflectivity, was tested and found to fail because of the long exposure times and the high temperatures. The radiating surface melted and the reflectivity initially present was reduced to ineffectiveness. The ablation concept was selected on the basis of the capability to block effectively the heat transfer to the structure by both surface ablation and

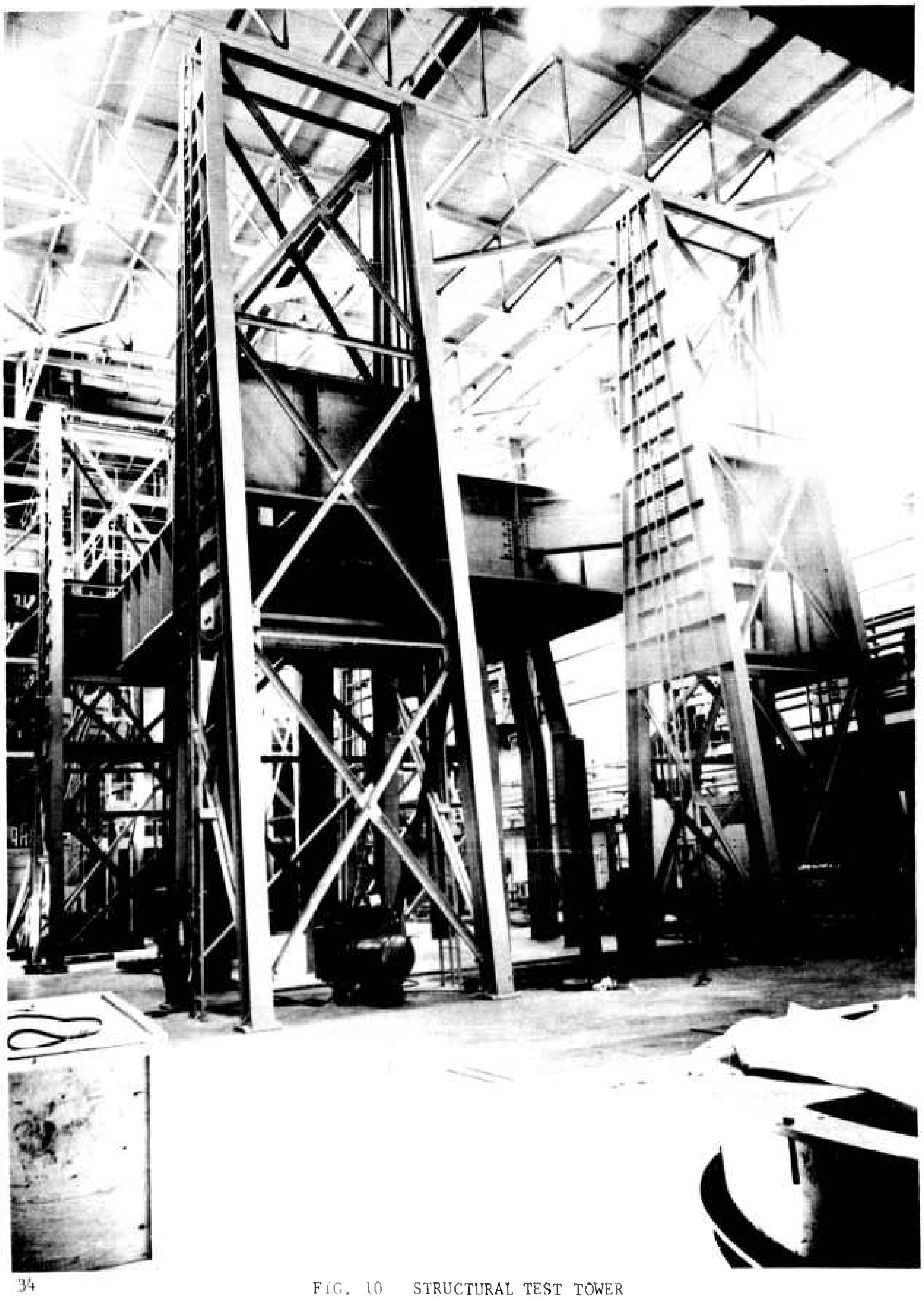


FIG. 10 STRUCTURAL TEST TOWER

subsurface insulation.

Since the environment is not such to cause ablation in the lower heating rate extremes, care was taken to insure that the material selected would function effectively at the two heating extremes indicated by the theoretical analysis. Scaled testing of several materials was done under the two extremes. To simulate the low heating rates and subsonic velocity, and air-oxygen-acetylene blast burner was used. The specimen was held such that the surface was parallel to the direction of gas flow. Under this condition of exposure, the burner was capable of producing heating rates up to 45 BTU/ft^2-sec.

Simulation testings of materials in the upper heating rate extreme was done by exposing samples to the exhaust gases of a small liquid propellant rocket motor. The specimens were exposed at an angle of 25° with the center line of the exhaust gases. Heating rates over a range of 80 to 160 BTU^2-sec were obtained. The gas velocity was Mach 2.5.

The results of all test indicated an asbestos flock reinforced phenolic resin molding compound was the most applicable under the wide heating range. Other materials tested included fiberglass and silicon dioxide (Refrasil) reinforcements, both randomly and schematically oriented, in phenolic resin. Ablating ceramics were studied also but the susceptibility of these materials to mechanical shock and vibration failures removed them from serious considerations. The recorded temperature rise on the back surface of a one-inch thick test panel of the asbestos reinforced phenolic molding compound showed that after three minutes exposure, a rise of 12°C was the highest obtained under all test conditions. This material, manufactured by the Cincinnati Testing Laboratories, is designated CT-301. Since the maximum size plate available commerically is a 9 in x 9 in size in the 1 in thickness, the star region is insulated by a series of these plates designed to have overlapping joints. The cost of manufacturing tooling to produce a one-piece or four-piece star region was outside of the financial boundaries of the development. However, with sufficient funds, the manufacture of a one piece star region is entirely feasible.

(b) <u>Heat Shield</u>

The heating of the heat shield and related components is not expected to be of the same degree of severity as that anticipated for the star region. It was necessary that the material be capable of withstanding exposure for approximately two minutes to a temperature up to 1400°K while maintaining a back surface temperature below 900°K. Although not mandatory, it was desirable to use a material in these areas which could be applied after assembly by brush or spray techniques. These requirements limited the investigation to a few materials. Advantage of the Lockheed experience on POLARIS was taken in selected the materials for investigation. These materials included commercial coatings manufactured by Products Research Company (Material PR-1910), Dow Corning (DC-150, DC-501 materials) and Minneapolis Mining and Machinery Corporation (X-3-0167 material).

The testing of the materials was done in three ways. First, the materials were subjected to radiant heating, second to blast burner conditions outlined previously, and finally to the rocket motor exhaust conditions.

From the test results, the PR-1910 material, a filled silicone rubber, was shown to be superior. The 3-x-0167 material showed considerable promise but as the "x" in the designation indicates, this material is still developmental and could not be used at this time.

Results with the PR-1910 showed excellent adhesion, good insulation, simplicity of application and the coating did not spall when exposed to the rocket motor exhaust. The mechanism of insulation employed involves a chemical reaction dissipating heat and causing an outgassing of the material which creates a foam. After three minutes of exposure of a $\frac{1}{4}$ in thick coating on a 1/16 in aluminum plate to the blast burner, with a heating rate of approximately 30 BTU/ft^2-sec, a back surface temperature rise of the order of 155°C was observed.

The coating is applied by troweling or bushing. The one undesirable feature is the lack of close control on the thickness, however, a minimum thickness of $\frac{1}{4}$ in can be obtained easily. The coating will air-cure but requires approximately 96 hours. It can be patched if scarred but the 96-hour cure must be included. This material was recommended for use on the SATURN heat shield, the inboard side of the outboard engine expansion cones and all structure exposed to base heating. It was not recommended for use in an inclosed area which may contain an ignition source since the products of combustion of the PR-1910 will support combustion if the source is sustained.

2. (C) Propulsion

a. Introduction

The components and subsystems of the SATURN booster propulsion system have undergone extensive testing during the past six months. Testing of the H-1 engine at Rocketdyne utilized three different test positions, allowing simultaneous progress in three areas. One test position was devoted almost exclusively to the production acceptance testing of H-1003 through 1009. These engines, after satisfactory acceptance testing for use on SA-T, were shipped to ABMA and performance tested with booster configuration hardware. These engines were also used at ABMA to establish start and shut-down transients and to obtain general operating characteristics at the 165K level. A second test position at Rocketdyne was used for support testing; some short term R&D testing; and, in the latter months, duration testing, preliminary flight rating tests (PFRT) and model demonstration tests. The third position was used for additional R&D studies, including the gas generator linked valve, blender simplification, dry starts, etc. In addition to the main engine test positions, the components test stand was used for testing of the turbopump, gas generator, oronite blender, hypergol cartridge, turbine spinner, and the various engine valves.

In the past six months, as the result of Rocketdyne and ABMA tests on the complete engine and components, several improvements have been completed in various stages of design or test. These items include a mechanically linked gas generator valve, to assure a specified and repeatable lox lead in the gas generator, the incorporation of the unprogrammed slotted manifold to reduce tube splitting, an integral lox dome and elbow to reduce several joints, an aspirator for exhausting the turbine gases of the outboard engines into the main engine exhaust stream, and an improved lox seal. A summary of current engine development problems is shown in Table 3.

Work was continued at ABMA on other subsystems design of the propulsion system. Initial development testing of the heat exchanger has been completed. Delivery of additional propellant suction lines and wrap-around assemblies for use on SA-T was accomplished and the qualification testing of these items was initiated at a vendor's plant in order to conduct the tests at the high vibration level anticipated for the SATURN.

Various qualification tests have been conducted on the major mechanical components of the pressurization system. Initial studies have been conducted for an advanced pressurization system.

Studies and tests are also being conducted in the areas of tail heating, compartmentization, and fire fighting techniques.

Tests of propellant container interflow, sloshing, and vortexing were also continued.

TABLE 3

CURRENT ENGINE DEVELOPMENT PROBLEM STATUS

Area	Problem	Action
1. Thrust Chamber	1. Nozzle tube splits	1. Slotted manifold thrust chamber was designed to ease this problem and present testing indicates that the design objectives will be met. Additional features such as a tapered manifold and internal coating are being investigated.
2. Turbine Spinners	1. Ballistics consistency	1. Initiation of an improved quality control program.
	2. Ignition system	2. Development of a Microloc 500-volt initiator and an exploding bridge-wire system.
	3. Temperature sensitivity	3. Incorporation of a heater blanket.
3. Hypergol System	1. Moisture may accumulate downstream of cartridge due to condensation during standby.	1. Blow-off covers or possibly a downstream purge system is being considered.
	2. Contamination of igniter fuel and main fuel valve control system by hypergol.	2a. Servicing of valve after five tests and discarding line after 20 tests recommended.
		2b. A hypergol cartridge which slides after diaphragm rupture and diverts some upstream flow to the main fuel valve is being designed.
		2c. Viton A main fuel valve O-rings are now standard for H-1. All engines in the field will be retrofitted when hardware becomes available.

TABLE 3 (Cont'd)

Area	Problem	Action
4. Main Fuel Valve	1. Sticking and binding	1a. Use of viton A is standard. 1b. Installation of an orifice in the MFV control line to eliminate dwelling of the valve and to minimize entry of hypergol into the valve actuator. 1c. Actuator link is being reworked to prevent binding.
5. Engine	1. Marginal turbine spinner overlap	1. The turbine spinner overlap is being increased.
	2. Joint leakage	2. An H-1 engine leakage investigation program is progressing.
6. FAMU	1. Low mixture ratio of Oronite to RP-1 has been encountered. Also, there is considerable scatter in data. Filling and measuring difficulties have complicated the problem.	1. Filling measuring procedures have been refined to insure best obtainable results. The allowable mixture ratio band has been widened to 2.75 ± 0.75%. The nominal Oronite metering orifice diameter has been increased to 0.028 inc.
	2. Leakage past full indicator plunger. Dimensions are such that the O-ring on the shaft can travel out of the hole.	2. Redesign of plunger configuration to incorporate a stop which insures that the O-ring does not come out has been retrofitted to all engines in the field.
	3. Moving parts	3. The spool may be eliminated by tapping off fuel from the downstream side of the sequence valve and turning the fuel additive unit upside down.
	4. Overheating	4. Redundant thermostat requested by ABMA.

TABLE 3 (Cont'd)

Area	Problem	Action
7. Gas Generator and Turbine	1. Sequencing of gas generator fuel and lox control valves has been such that gas generator sequencing and turbine erosion have resulted. Over 25% of engine malfunctions have been caused by this problem.	1. Testing is being accomplished with various opening pressures. A mechanically linked gas generator propellant valve and dry start sequence may alleviate this problem.

b. H-1 Engine Development

(1) System Design and Testing

(a) H-1 Engine Testing. The H-1 engine has been tested for a total of 14,965.7 seconds in the R&D configuration and 7,529.2 seconds in the production configuration. Table 4 shows the breakdown of the engine testing (Rocketdyne and ABMA) by engine, time and number of tests. The basic single engine parameters are shown in Table 5. Performance of the 165K engine versus altitude is shown in Figure 11. The control system, which times the opening and closing of the various valves during start and shutdown operations, was improved in its repeatability of operation during this period. No complication of the simplified pressure ladder sequence (Fig. 12 and 13) has been necessary. In addition the hypergol ignition system (triethyl aluminum, TEA) was developed into a highly reliable thrust chamber start technique. To date there has been no failure-to-ignite with a hypergol cartridge (Fig. 15).

The basic design of the H-1 engine will allow improvements from other engine development programs to be easily incorporated. One of these was an integral lox dome and elbow, which was attempted on engine H-10 but resulted in a vibration, or "buzz", of the lox dome at about 875 cps and 50 to 60 g's. In order to prevent delay, it was decided to use the lox dome assembly with eight attaching bolts until further development work could be accomplished.

Fuel-only lubrication of the gearbox is under development and may be used in the H-1, thus eliminating the fuel-additive blender unit and ground charging and servicing equipment.

The basic difference between the H-1 engine and the S-3 type engine is the mounting of the turbopump as a portion of the engine below the gimbal point, the adoption of a pressure-ladder sequence for ignition cutoff, employment of a solid propellant turbine spinner start system, hypergolic ignition, extensive reduction of the electrical system, and fuel-additive lubrication of the gearbox.

(b) Engine Ignition. Tests using the hypergol solution triethyl aluminum (TEA) have shown better ignition characteristics than the triethyl boron, but leave a contaminating aluminum oxide deposit. A mixture of TEA-TEB (4%/96%) has satisfactory ignition characteristics, does not leave deposits but becomes marginal in its operation at low temperatures. Experience at ABMA has shown satisfactory operation with the 100% TEA and its use will be continued.

Even though the hypergol solution has always ignited the engine, it is desirable to provide a redundant source of ignition because of the large magnitude of danger from an explosion if an engine discharges unignited propellants. The danger exists because the turbine exhaust

TABLE 4

H-1 ENGINE TESTING

R&D Engines[1]

Engine No.	Time	No. Tests
H01	201	6*+10
H02	1962.6	52
H03	2681.4	58
H04	287.8	7
H05	1338.3	26
H06	1197.6	27
H07	3618.6	56
H10	166.5	10
H11	3090.0	65
H13	414.3	25
H14	76.0	2
	14,965.7	338

*Water Blow-Down Tests

Delivered Engines[2]

Eng. No.	NAA		ABMA		Totals	
	Tests	Time Sec.	Tests	Time Sec.	Tests	Time Sec.
H1001	4	668.0	5	562.4	9	1230.4
H1002	4	329.9	3	330.2	7	660.1
H1003	5	464.1	8	252.8	13	716.9
H1004	3	445.0	3	312.7	6	757.7
H1005	9	715.5	2	251.0	11	966.5
H1006	3	514.6	2	250.0	5	764.6
H1007	3	667.3	3	401.1	6	1068.4
H1008	5	387.8	2	251.2	7	639.0
H1009	10	472.4	4	253.2	14	725.6
	46	4664.6	32	2864.6	78	7529.2

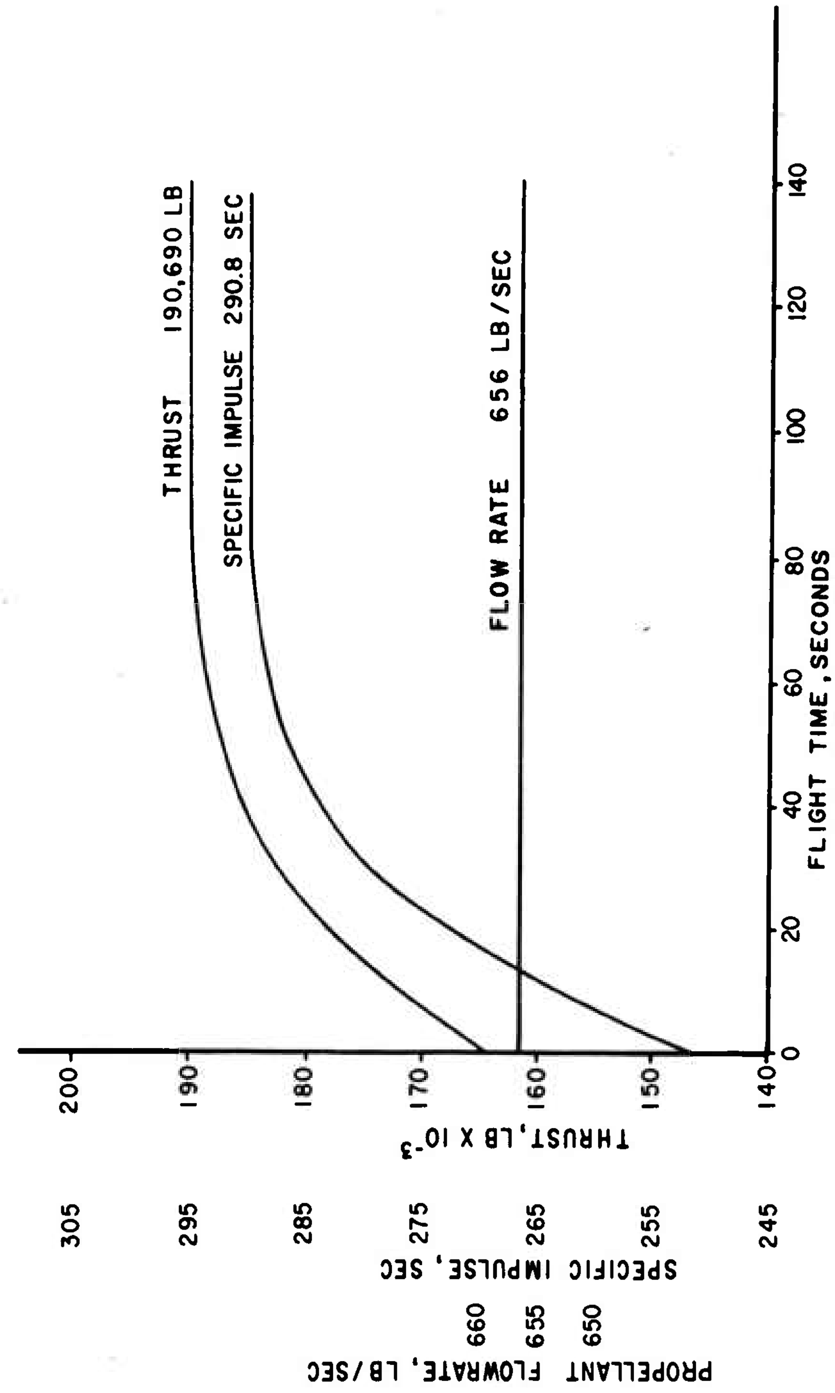
Total H-1 Engines Testing: 416 Tests, 22,494.9 Sec.

1. Through 4 Dec 1959.
2. Through 19 Dec 1959.

TABLE 5

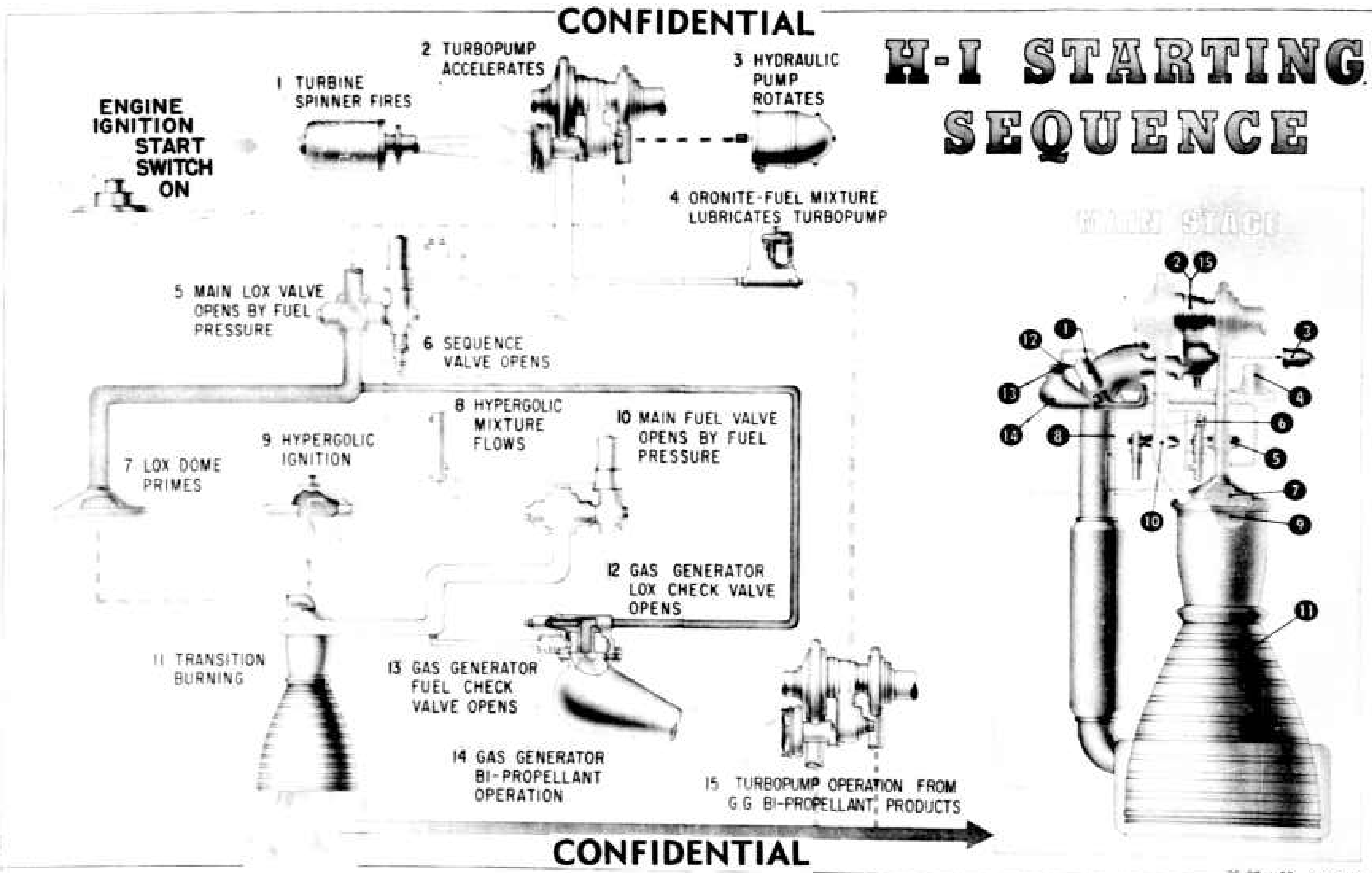
H-1 SINGLE-ENGINE PARAMETERS
(H-1010 and Subsequent Engines)
As of 7 January 1960

	165K	188K
Thrust at sea level, lb	165,000 ± 3%	188,000 ± 3%
Thrust at 120,000 ft	190,690 ± 3%	213,700 ± 3%
Sea level specific impulse, sec	248.0	252
Total propellant flowrate, sea level	. . .	. . .
Thrust chamber and gas generator, lb/sec	655.71	735.91
Fuel flowrate (with lube additive fuel flow), lb/sec	200.6	255.0
Lox (without heat exchanger), lb/sec	455.1	510.9
Oxidizer	lox	lox
Temperature, °F	-297.4	-297.4
Density, lb/ft^3	71.38	71.38
Fuel	RP-1	RP-1
Temperature, °F	60	60
Density, lb/ft^3	50.45	50.45
Engine mixture ratio, w_o/w_f	2.27	2.27
Oxidizer pump inlet pressure, psia	78	98
Fuel pump inlet pressure, psia	50	40
Weight, lb		
Outboard		
Wet	1891	1891
Dry	1647	1647
Inboard		
Wet	1680	1680
Dry	1457	1457



ESTIMATED NOMINAL ALTITUDE PERFORMANCE
ATMOSPHERIC PRESSURE EFFECT ONLY
H-I ROCKET ENGINE (165 K LEVEL)

FIG. 11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H-I STARTING SEQUENCE
ENGINE IGNITION START SWITCH ON
1 TURBINE SPINNER FIRES
2 TURBOPUMP ACCELERATES
3 HYDRAULIC PUMP ROTATES
4 ORONITE-FUEL MIXTURE LUBRICATES TURBOPUMP
5 MAIN LOX VALVE OPENS BY FUEL PRESSURE
6 SEQUENCE VALVE OPENS
7 LOX DOME PRIMES
8 HYPERGOLIC MIXTURE FLOWS
9 HYPERGOLIC IGNITION
10 MAIN FUEL VALVE OPENS BY FUEL PRESSURE
11 TRANSITION BURNING
12 GAS GENERATOR LOX CHECK VALVE OPENS
13 GAS GENERATOR FUEL CHECK VALVE OPENS
14 GAS GENERATOR BI-PROPELLANT OPERATION
15 TURBOPUMP OPERATION FROM G G BI-PROPELLANT PRODUCTS
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FIG. 12

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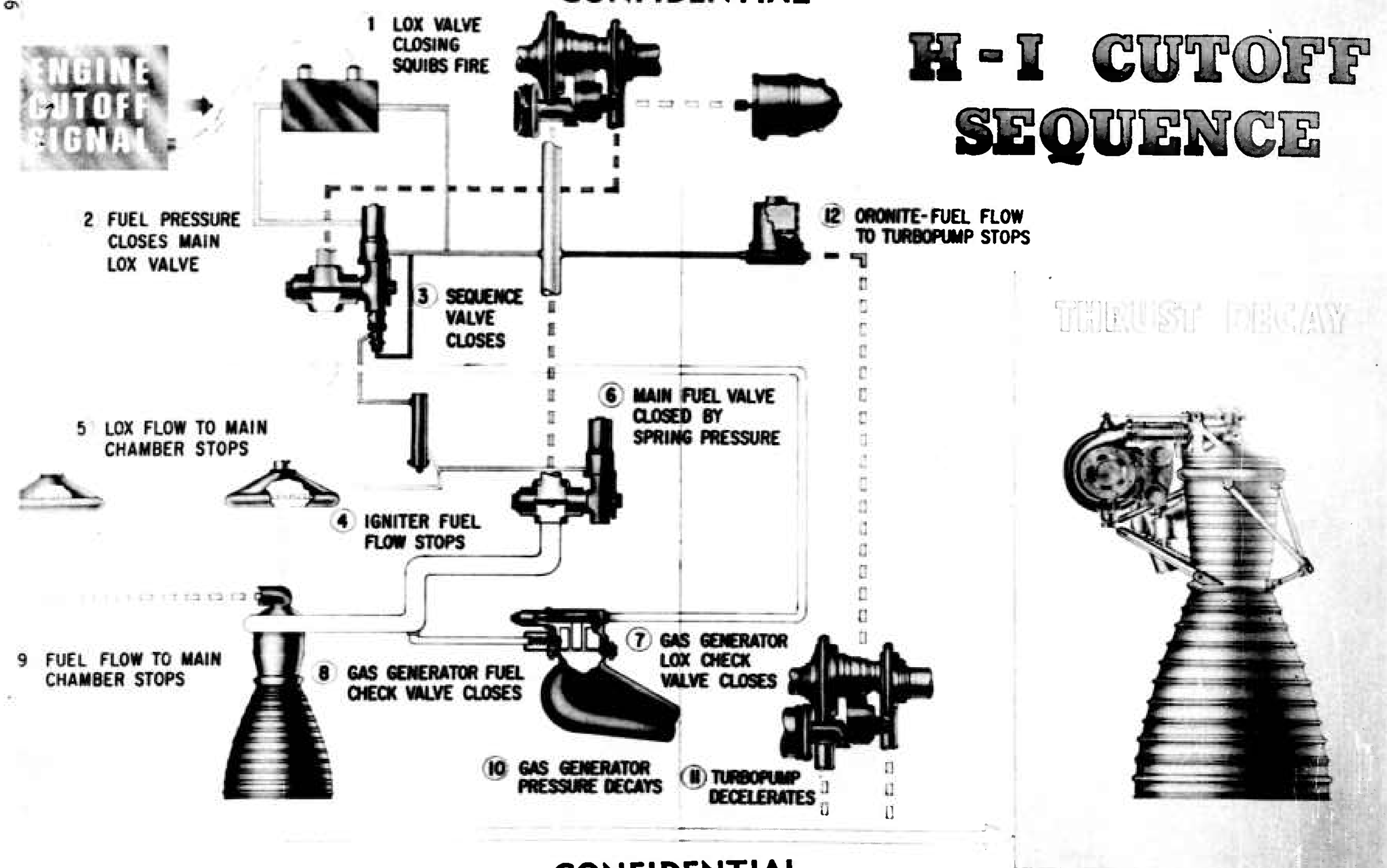

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FIG. 13

gases created by the solid propellant spinner are discharged through the aspirator into the engine. Initial tests for a redundant ignition source to back up the hypergol were concentrated on a pyrotechnic ignitor. Results of these tests were completely unsatisfactory as the ignition flame was drowned by the high initial lox flow and could not mix with the ignitor fuel, resulting in no ignition. Additional studies are under way to determine a satisfactory means of providing dual ignition sources without decreasing engine reliability and still present a highly satisfactory performance.

Because of the 6-degree cant of the outboard engines in the SATURN cluster, it is desirable to obtain starting information with the engine in this position. A series of ten tests was conducted by holding the engine in a canted position (up to 6°) by use of the gimbal actuators for all starts conducted in this study. No adverse effects were noticed in starting, although a slight air pocket was noticed in the fuel system when the turbopump was higher than the high-pressure fuel bleed.

(c) Engine Control. For the initial H-1 engine control system, a lox or gaseous oxygen leak near the main lox valve control line could cause the fuel to freeze in the control system, preventing a normal engine shutdown. To lessen the opportunity for such an occurrence, a "series control system" was designed in which the main lox valve opening control line was integrated into the control line for the FAMU supply line. This setup provides a continuous flow of control fluid lessening the possibility of the fluid freezing. The series control system was evaluated and as no detrimental characteristics have been encountered, the system will be incorporated into the production engines.

(d) Electrical Control System. The electrical system consists of the start and cutoff controls of the engine, several engine component heater units, and a checkout of critical conditions to give a "preparation complete" indication. The electrical system also includes the safe arming system, which prevents accidental firing of the turbine spinner initiators, and the engine safety shutdown circuitry, such as the overspeed trip, (static test only) the thrust o.k. switch (indicates satisfactory thrust), and rough combustion detection devices (ground function only). Other electrical circuits may also initiate shutdown. These include the redline cutoffs, observer cutoffs, and programmed cut-off during flight. Safe arming is a requirement of the electrical control system since the H-1 engine is pressure sequenced and capable of going into mainstage when a solid propellant gas generator initiator is fired after the propellants have been tanked. The H-1 electrical system connects the eight engines in parallel so that integrated or independent operation is possible. Development of an in-flight combustion stability monitoring device is in progress, and the possible elimination of some heaters by using boattail heating is under investigation.

To increase starting reliability in the eight-engine cluster, a dual firing circuit has been designed whereby one turbine spinner initiator on each engine is fired from one source and the other eight initiators are fired from another source. These two sources supplying one amp at 500 v per initiator are independent secondary windings of a high voltage transformer. The primary is switched to use a minimum of control relays. This dual firing system allows the engines to start even though a malfunction of one circuit occurs, since the H-1 will start by firing one initiator in the turbine spinner.

Because of the anticipated high temperatures, and to prevent physical damage from a minor explosion, etc., the critical engine control cables are inserted inside a stainless steel flex hose and covered with a special temperature resistant insulation which will withstand the extreme temperatures.

The control system makes the H-1 engine unique by eliminating the electromechanical controls and components required on earlier engines. The pneumatic control system of these earlier engines has been eliminated and its functions given to the main fuel line discharge pressure. The pressure-operated sequence of this engine also greatly reduces the functional monitoring concept required with earlier engines. Also, the lube oil system has been eliminated by adoption of a fuel additive mixture unit.

No electrical equipment is required for start or cutoff sequencing of the engine except an electrical signal to the turbine spinner initiators for start and an electrical signal to the Conax valve, closing the main lox valve stopping the engine.

H-1 controls are limited to those parts which govern the start-stop sequences. They consist of the main lox valve, main fuel valve, Conax valve, gas generator lox and fuel valves, turbine spinner initiators, and fuel and lox orifices.

(e) Dry Jacket Starting. The dry jacket start has been considered the most desirable starting sequence for H-1 since the initial tests in the program. Early attempts to achieve dry starts with the 1.2-second duration turbine spinner, although successful, indicated that a longer duration spinner would be necessary if optimum start conditions were to be obtained. The primary problem was to achieve a high enough spinner power level (and subsequently, engine prime speed) to avoid chugging during the thrust build-up transient. This chugging was found to occur with both types of starting sequences at lower prime speeds.

The dry start, while advantageous, is not mandatory for successful H-1 engine performance, therefore, further work on dry starts was suspended until a satisfactory long duration spinner was available. The advantages of the dry start lie in the ease of preparing the engine for a firing and in the start characteristics. With a dry start, the measured prefill procedures during countdown are eliminated. During start, the propellant feed system is allowed to stabilize prior to engine prime. The lox flow to the thrust chamber has a more uniform quality at mainstage start. Oscillations in the fuel system occurring during main fuel valve opening are allowed to dissipate before chugging or high-frequency combustion instability can occur. The feed system recovers more easily from pump cavitation effects and allows the gas generator inlet parameters to stabilize sooner. Tests indicate that early stabilizing of the gas generator system can be very important to gas generator transients and result in a lower sensitivity of the system. Higher prime speeds are available with dry jacket starts, giving higher control system pressure and resulting in a more positive sequencing and a more reliable main fuel valve opening. Elimination of the spongy, variable-volume air pocket in the jacket which exists with the fuel-filled start may result in a more repeatable and inherently more stable thrust buildup. The dry jacket start also allows the removal of the thrust chamber fuel manifold purge system provided thrust chamber cleaning procedures are carefully followed.

(f) Single Engine Testing With Heat Exchanger. Engines H-1006 and H-1007 were the initial production engines with ABMA supplied heat exchangers to be acceptance tested at Rocketdyne. During these tests a pressure less than atmospheric, observed during ignition and cutoff transients at the inlet duct, caused slight structural damage to the heat exchanger shell.

Future heat exchangers will incorporate several bands around the heat exchanger shell to prevent cavitation due to this small internal pressure drop during start and cutoff.

(1) H-1 Component Development

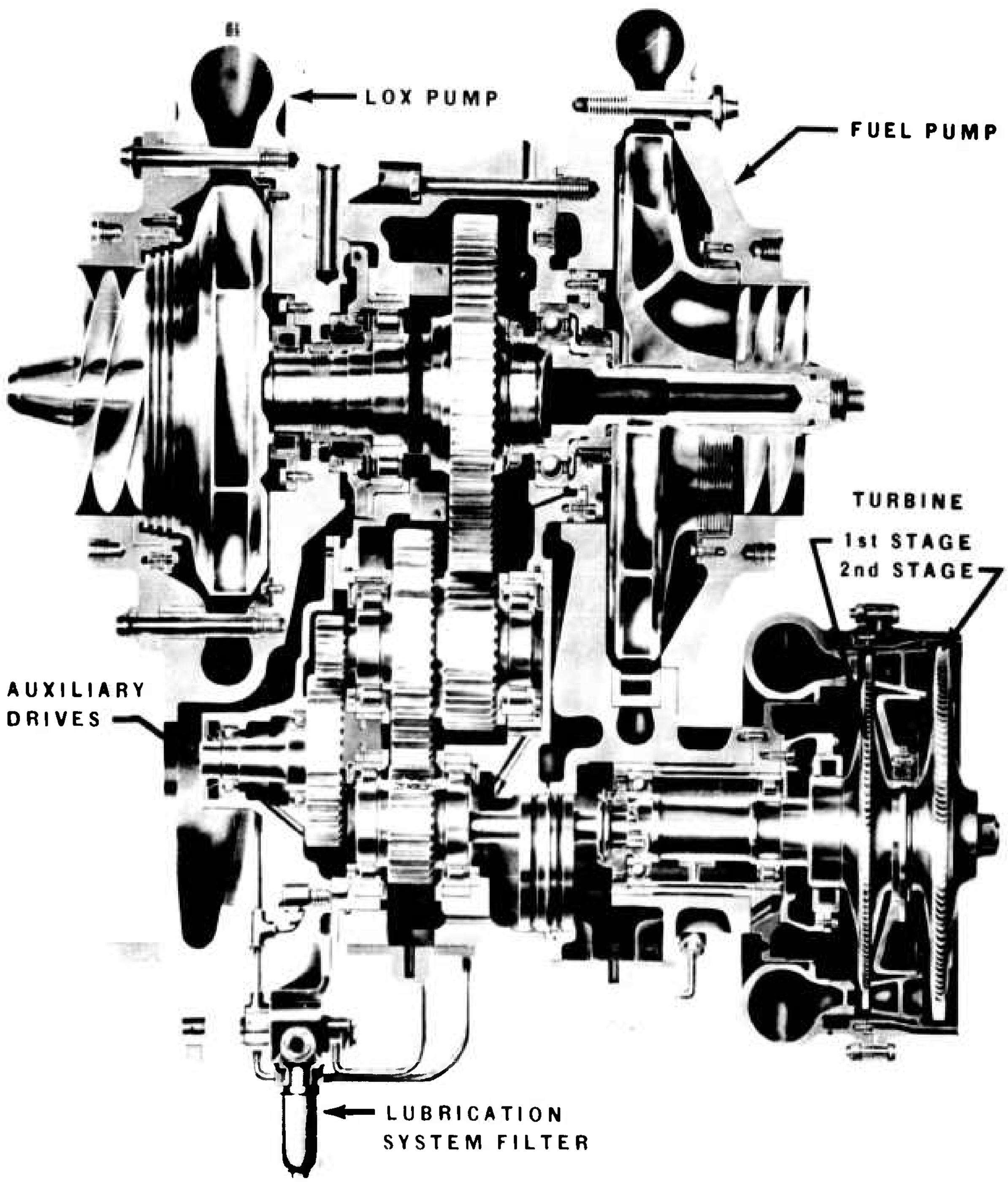
(a) Thrust Chamber. A basic S-3 thrust chamber was adopted for the H-1 engine. This thrust chamber was reinforced with additional bands and stiffening hoops to provide for the increased thrust level at which it eventually will be used. Also a lower turbopump mount was welded to the chamber and ears were added to the thrust ring and manifold for attachment of the outriggers. Brackets for the fuel additive mixture unit and other components were welded to the combustion zone of the chamber. The upper turbopump support is attached to supports bolted to the lox dome.

To reduce the pin holes and splits in the tubes that form the thrust chamber and to increase the anticipated life of a thrust chamber, a series of tests is being conducted on the use of a coating for the inside wall of the chamber. A zirconium oxide coating is presently being tested.

(b) Turbopump. The H-1 engine uses the Mark III turbopump assembly (Fig. 14). The basic turbopump has been proved through use in the JUPITER, ATLAS, and THOR engine systems. Some new features developed for the turbopump are the one-piece (Mod-1) gearbox, incorporating full-depth gears with integral bearing interraces; a low-pressure turbine exhaust hood seal design; and fuel-additive lubrication.

H-I TURBOPUMP CONFIGURATION

FIG. 14



The Mark III turbopump has been designed into the H-1 engine package so that the gearbox is horizontal instead of vertical. This has necessitated other minor modifications such as drain plugs.

The extensive testing of the fuel-additive lubrication system has shown the technique to be an improvement over the elaborate system required for oil lubrication. Tests conducted with the H-1 engine show that the fuel-additive mixture in combination with the improved gearbox allows the pump bearings to run cooler and perform more reliably.

(c) Conax Valve. The Conax valve (Fig. 15) is comprised of two normally closed, explosive operated valves in parallel. Each side of the valve has flow restriction after firing. The time delay from electrical signal until the Conax valve opens is in the order of 0.001 sec. This valve is designed to have a 100% no-fire current rating of 0.5 amp and an operating current of 3.0 amp. The bridgewire resistance is 0.7 to 0.14 ohm and is normally checked out with a current of 0.010 amp.

It is characteristic of the Conax valves that small particles of metal are sheared loose from the body of the valve when it is fired. These particles could contaminate the main lox valve actuator piston. Further studies are in progress to reduce the shrapnel from the diaphragm of the Conax valve.

A study was conducted to determine the minimum cutoff impulse that could be obtained while maintaining the specified lox pump inlet and outlet surges.

The effects of various valves and lines on cutoff impulse are shown in the following table.

Items Used	Delay (Average) Cutoff to Main Lox Valve Leave Position	Approximated Cutoff Impulse Using Same Thrust Level and Main Lox Valve Closing Rate
120-grain Conax valve with orificed closing control line	270 m sec	82,000 lb-sec
120-grain Conax valve with H03 simulated 3/8-in. closing control line	75 m sec	50,000 lb-sec
Solenoid valve with 3/8-in. closing control line	165 sec	64,000 lb-sec
60-grain Conax valve with 3/8-in. closing control line	140 sec (not repeatable)	60,600 lb-sec

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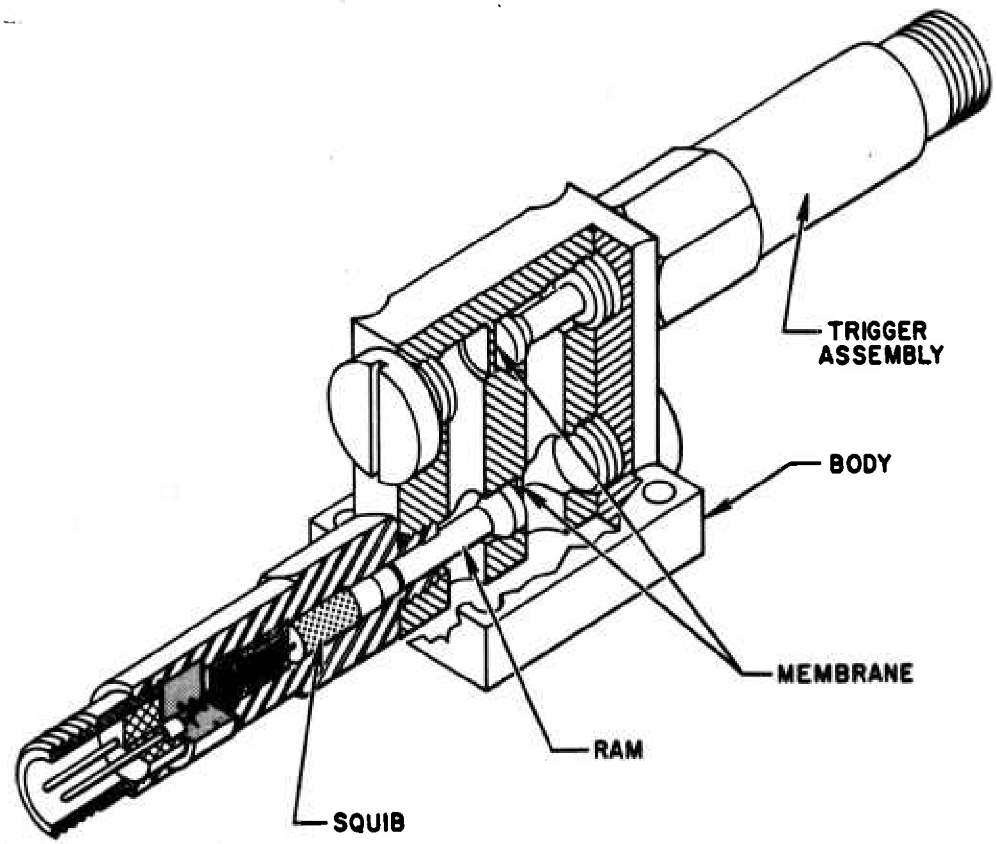

FIG. 15 CONAX SQUIB-ACTUATED VALVE

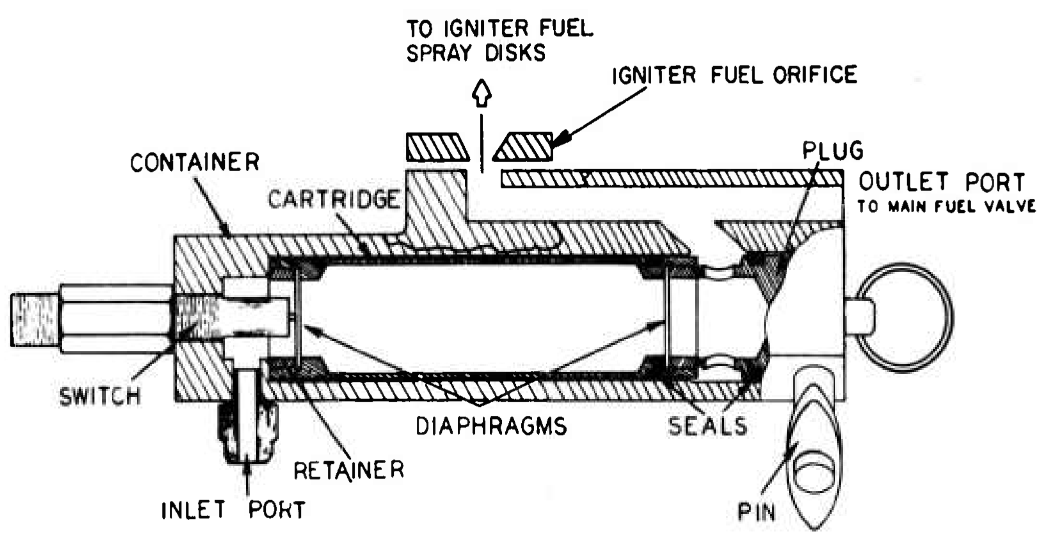

FIG. 15 HYPERGOL CONTAINER

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Items Used	Delay (Average) Cutoff to Main Lox Valve Leave Position	Approximated Cutoff Impulse Using Same Thrust Level and Main Lox Valve Closing Rate
60-grain Conax valve with 1/2-in. closing control line	110 sec	55,100 lb-sec
60-grain Conax valve with 1/4-in. closing line	800 m sec (not repeatable)	169,500 lb-sec
*Estimated nominal 120-grain Conax valve with 1/2-in. closing control; no restrictor on closing side	50 m sec	42,000 lb-sec

*The 120-grain Conax valves were not available during the latter part of the testing. For this reason, the values of this valve with 1/2-in. closing control line were estimated. The final values for use on the prototype engine were the 120-grain Conax valves with the 1/2-in. closing control line. This was estimated to give a cutoff impulse value that should remain within the requirements of 70,000 lb-sec maximum for the 165,000-lb thrust level of the H-1 model specification.

(d) Aspirator. The aspirator configuration to be employed on the outboard four engines of the cluster (Fig. 16 and 17) is intended as a means by which momentum from the high-velocity thrust chamber exhaust gases may be transmitted to the low-velocity turbine exhaust gases. In this manner, the recirculation of turbine exhaust gases in the missile boattail during flight will be reduced.

The aspirator is essentially a shroud of circular cross section and longitudinally varying curvature that is attached to the nozzle "hat bands" and extends a few inches beyond the nozzle exit plane.

The design of the R&D aspirator configuration was similar to the geometry of those aspirators used with the ATLAS sustainer engine systems. The ATLAS engines with aspirators have functioned satisfactorily without apparent influence on system performance.

(e) Solid Propellant Gas Generator (Turbine Spinner). A solid propellant rocket motor (Fig. 18) is used to supply hot gases to the H-1 turbine to initiate the start sequence of the engine. The hot gases spin the turbine and ignite the liquid propellant gas generator in a bootstrap fashion. The spinner propellant is a composite propellant using ammonium nitrate as an oxidizer and a copolmer of butadiene with methy vinyl pyridine (MVP) as a fuel and binder.

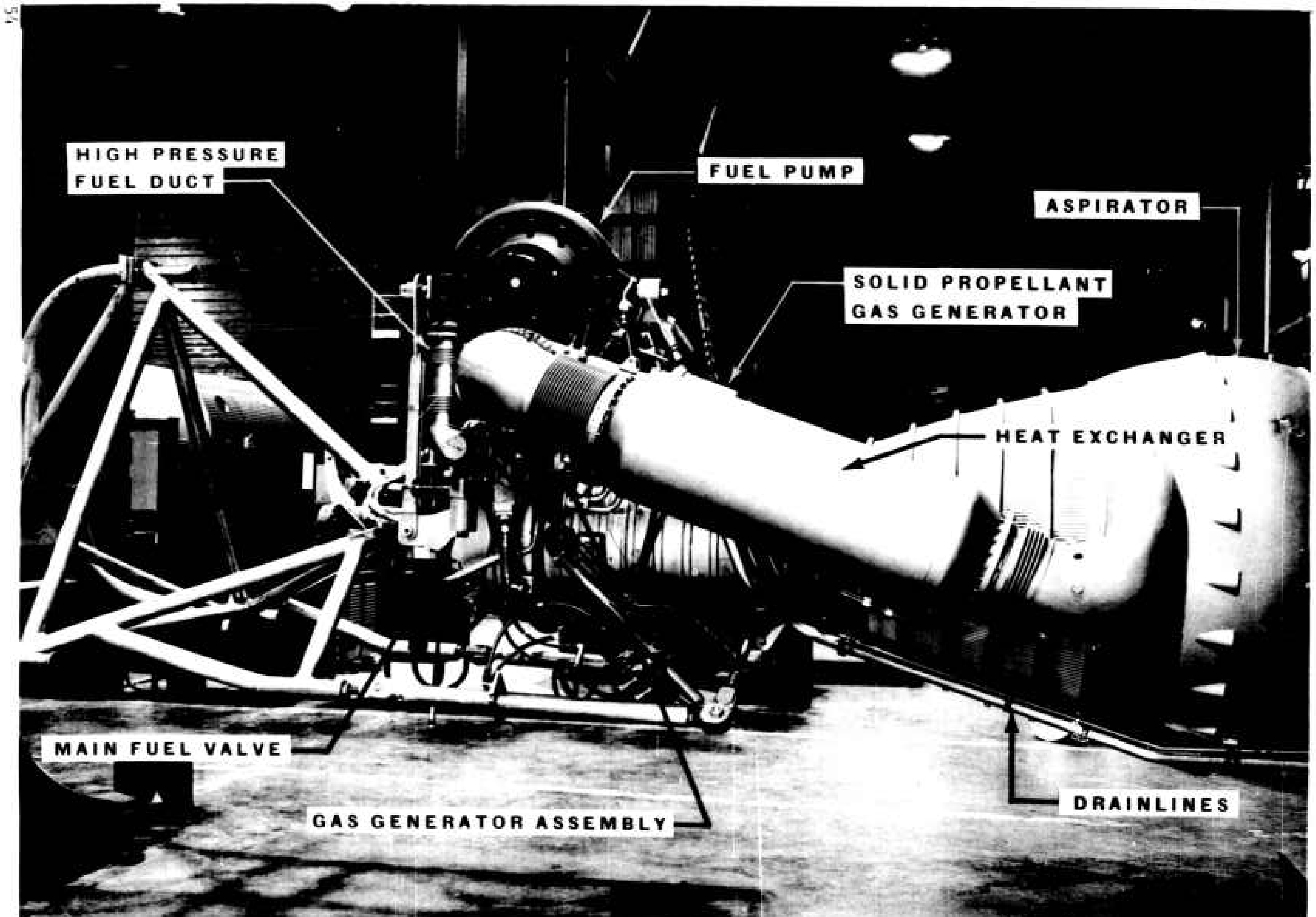


FIG. 16. ENGINE MOCK-UP (FUEL SIDE)

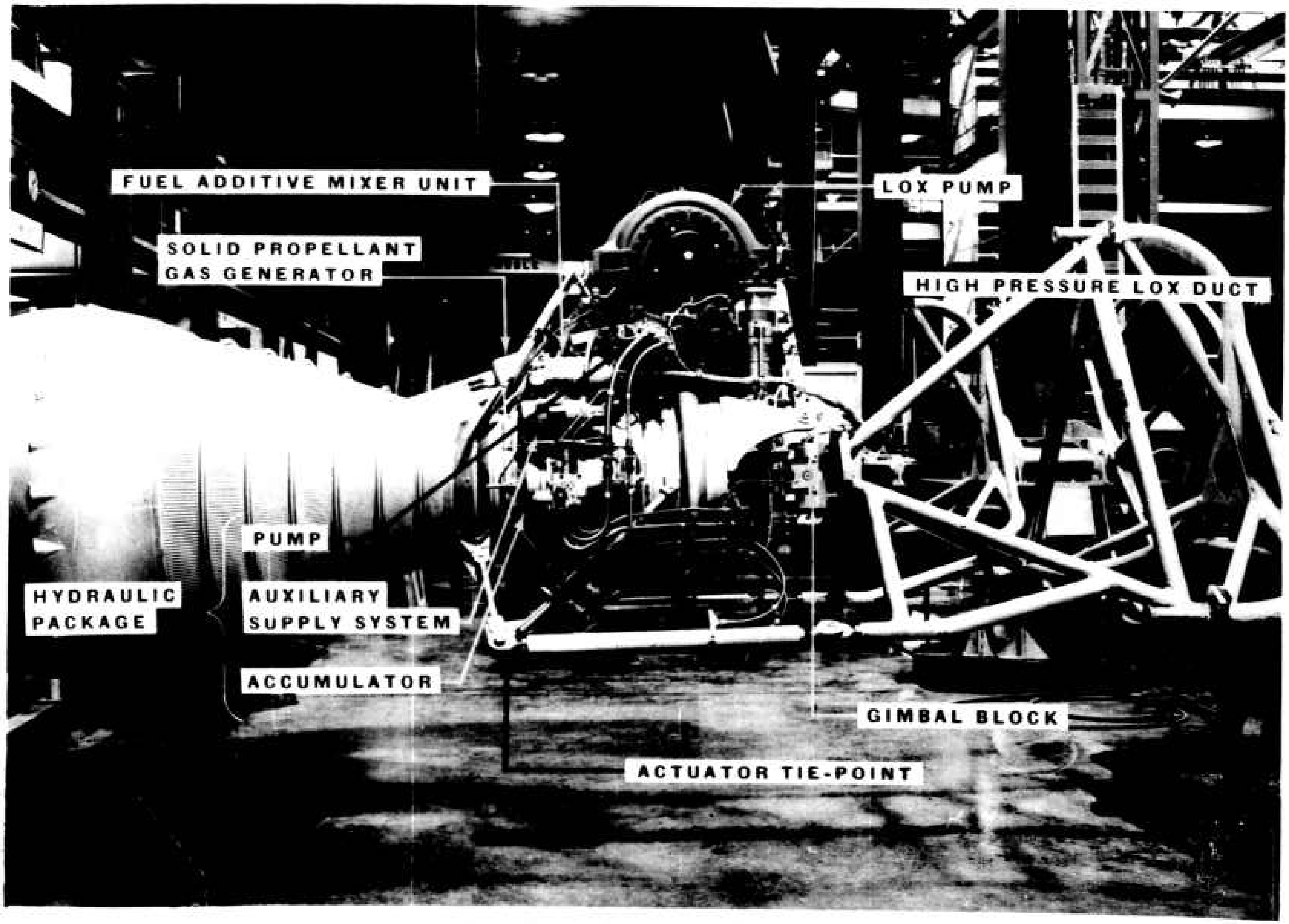


FIG. 17. ENGINE MOCK-UP (LOX SIDE)

FIG. 16 SOLID PROPELLANT GAS GENERATOR (TURBINE SPINNER)

The motor, composed of three grains housed in a steel case, is ignited by a 28-volt pyrotechnic initiator, burns for 1.05 seconds and creates a pressure of 800 psig. The Type A spinner initially used in the H-1 engine program is now in the process of modification for increased reliability. A polyethylene bag is now used to contain the igniter pellets, and it, in turn, will be retained in the head end by a plastic disc. Other improvements will be the removal of the metal spacers, nuts, and bolts; the use of a 500-volt ignition system; and closer propellant manufacturing tolerances. Investigation is also being conducted on the possible use of a fixed nozzle rather than the present removable one.

(f) Gas Generator Fuel Cracking Check Valve. In the interim before the redesigned gas generator mechanically linked fuel and lox valve is available, a test program was initiated at ABMA on the present GG fuel cracking check valve which has not readily met check-out specification at low temperatures. This valve must be predictable in operation to prevent improper start sequencing. In the test program, three valves were tested under both pressure shock loading and a steady increase in pressure over a temperature variation of 40°F to 0°F. Temperature had little effect on the pressure required to open the valve during shock loading. However, the temperature did change the pressure at which the valve opened during a slow steady increase in pressure. Further tests are under way to find a satisfactory solution.

A summary of the component follow-on programs to improve engine reliability can be seen in Table 6.

(3) ABMA Single Engine Testing

Testing of the H-1 engine at ABMA during the last six months consisted primarily of performance and calibration testing of the engine using flight configuration suction lines and wrap-around assemblies. The majority of engines were checked in the outboard configuration to demonstrate the reliability of the H-1 engine in the outboard configuration. Various test objectives during these tests included: investigation of the reliability of the suction lines and associated braces; observation of the gimbal performance of the outboard engine actuators; study of turbine spinner performance; and general engine performance in long duration tests.

c. Propellant System

(1) Propellant Suction Lines and Wrap-Around Assemblies

The propellant feeding system must provide the pump with the propellants at the proper rate to prevent cavitation of a magnitude sufficient to influence engine performance. Since the turbopump is mounted on the thrust chamber (Fig. 16), the propellant feed lines must follow the motions of the pump without imposing undue loads on the pump housing or the hydraulic gimablling system. The engine has nominally

TABLE 6

COMPONENT FOLLOW-ON PROGRAMS

Area	Engines H1001-H1009	H1010 and Subsequent	Final Design Considerations
Thrust Chamber	1. High-strength dome bolts	1. New outrigger (due to new gimbal)	1. Loop tube thrust chamber study (turbine exhaust disposal into thrust chamber)
	2. S-3 type modified for turbopump support mount	2. Increase fuel bootstrap takeoff to 1-1/4-in. dia.	
	3. Outriggers (new design) 45° from S-3 existing location	3. New forward pump mount support fittings to take flight loads	2. Incorporate features to minimize possibility of tube splitting
	4. All outboard chambers to have provisions for aspirator heat shield at exit manifold		
	5. Bracket for hydraulic system		
	6. Relocate exhaust duct bracket due to heat shield bracket		
	7. Redesign fuel inlet elbow (bolt accessibility)		
	8. Slotted fuel manifold		
Injector	1. Type 190M	1. No design change	1. Type 191
	2. Check valve welded open and recessed below injector lands		2. Flush-mounted igniter spray nozzles (buttons)
			3. Free draining
Lox Dome	1. S-3 type modified	1. Forged elbow to dome	1. Same

TABLE 6 (Cont'd)

Area	Engines H1001-H1009	H1010 and Subsequent	Final Design Considerations
Gimbal Assembly	1. S-3 modified for 165,000-lb, 45° actuators	1. Redesign for flight loads	1. Boot to protect the bearings (retrofit)
		2. Extra low tolerance	
		3. Infinite adjustment for alignment	
Ignition System	1. Hypergol cartridge with detector unit	1. New 9-cu. in. container with 6-cu. in. cartridge (retrofit)	1. No design change
	2. 6-cu. in. hypergolic container	2. Redesign to incorporate feature to prevent main fuel valve contamination	
Lox Pump	1. Mark III modified for horizontal drainage	1. Volute casting material change to Ten's-50	1. Diffuser vane redesign
	2. Increase inducer lead to reduce NPSH	2. "K" Monel bolts	2. Tapered inducer
	3. Increased clearance to minimize interference problem	3. Stepped wear rings	3. 8-in. inlet
	4. Inconel "X" lox bolts	4. Cutback impeller	4. 10-vane impeller
	5. Adapter to lox volute washers increase thickness	5. J.M. gasket (3 place) A286 stud and nut replacing safety wire and locking bolts	
Fuel Pump	1. Mark III modified for horizontal drainage	1. Volute casting material change to Ten's-50	1. Diffuser vane redesign
	2. 4130 cad-plated washers	2. Volute bolt heat-treated to 180-200 thousand psi	2. 8-vane impeller
		3. Residual fuel drain	3. New lead inducer
		4. Adapter O-ring seal	

TABLE 6 (Cont'd)

Area	Engines H1001-H1009	H1010 and Subsequent	Final Design Considerations
Turbine	1. Mark III modified	1. One-piece exhaust flange	1. Turbine one-piece manifold and bearing housing
	2. Split backup ring	2. Welded Skinner Seal-joint	2. 12-stud mounting flange
	3. Turbine shroud rework (welded)	3. Larger thermocouple port	3. O-ring mounting flange seal
	4. 5B30 bolts		4. One-piece shaft seal housing
	5. Pins to prevent rotation		5. Electronic overspeed trip
	6. Fixed quill shaft		
Gearbox	1. Mod 1 modified for horizontal drainage	1. Change material to Ten's-50	1. Redesign gearcase internal lube system
	2. Lube additive requirements	2. Drain manifold redesigned to one-piece configuration	2. Wider "A" and "B" gears
	3. Pressurized	3. New main shaft (nickel plated)	3. Remove ring jets and install orifice inserts
	4. No. 2 bearing carrier modification		4. Four additional gearbox to fuel volute mounting bolts
	5. Shot-peened "A" pinion		5. Study of treated gears and bearings with fuel only lubrication
	6. Floating mating rings		
	7. New ring jet sealing configuration		
Lube Blender	1. New design (111-cu. in. volume)	1. New filter 100 mc	1. Gearpump (study). Elimination of oronite and use of fuel only lubrication

TABLE 6 (Cont'd)

Area	Engines H1001-H1009	H1010 and Subsequent	Final Design Considerations
		2. Positive stop on indicator plunger	
		3. New discharge fitting including flush port (retrofit)	
		4. Horizontal drain feature	
Turbine Spinner	1. Lightweight, reusable, bolt-on	1. No design change	1. Larger grain for dry jacket start (retrofit)
Main Lox Valve	1. H-1 design	1. No design change	1. Position transducers with different resistance (retrofit)
	2. Install position transducers		2. 40% cam for control system actuation
	3. Locate thermostat on housing instead of bracket		
	4. Provision for heat exchanger take-off		
Main Fuel Valve	1. H-1 design	1. Viton "A" O-rings	1. Position transducers with different resistance (retrofit)
	2. Install position transducers		
	3. Tap both ports for fuel valve opening and draining		

two degrees of freedom; however, relative motions between the pump, thrust chamber, and thrust structure require that the propellant feed or suction lines be capable of six degrees of freedom. The suction lines are also subjected to loads due to acceleration, vibration, and surge pressures. They must also compensate for thermal contraction and relative motions between the propellant containers and the thrust structures. The straight propellant lines (from the tanks to the plane of the pump) and the wrap-around assemblies are made of 321 stainless steel. Most of the bellows are of 19-9 DL stainless steel. The lines are designed for a nominal pressure of 140 psia and to withstand transient surges of 400 psia.

A design which meets the requirements of motion, translation and rotation and imposes relatively small loads on the pump housing was selected (Fig. 19). This design uses three gimbal joints in an arrangement having their axes mutually perpendicular to one another. The components have been designed and qualified for operation under internal pressure, extremely low temperatures, vibration, shock, impulse, and fatigue or life cycle. Prototypes have been assembled and used in hot firings of single engines on the test stand since May 1959. During those firings, measurements of motions were made to investigate the transient conditions of starting and stopping. Thus far in the engine test program the suction lines have satisfied the design requirements.

The suction lines are now undergoing qualification tests as complete assemblies. A static load test to simulate flight acceleration loads is being run to determine deflections and stresses due to these conditions. The tests are being run with the suction lines positioned for all expected misalignments and pump motions.

Vibration tests are underway on the complete assembly in accordance with the provisions of the "Preliminary SATURN Shock and Vibration Requirements." Under these requirements, the suction line is installed in a test fixture with the upper flange fixed and the pump inlet flange attached to a bracket on the vibration table. The frequency range - 20 to 2000 cps - is then scanned for five minutes. This scan is made twice on each of three major axes and the resonant points are noted. The suction lines will then be subjected to five minutes of vibration at each resonant point in its respective plane. The machine used to excite these lines has a 25,000-lb-force output.

(2) <u>Propellant Container Pressurization</u>

(a) <u>Present System</u>. In line with the overall SATURN approach of using available and flight tested systems wherever feasible on the early vehicles, the present pressurization scheme is the same as the JUPITER's, i.e., use of unheated gaseous nitrogen stored in fiberglass spheres under 3000 psi for the fuel containers and gaseous oxygen for the lox containers. The gox is derived from lox passed through eight heat exchangers (one per engine) that utilize the turbine exhaust gases. The SATURN pressurization scheme is required to provide a lox

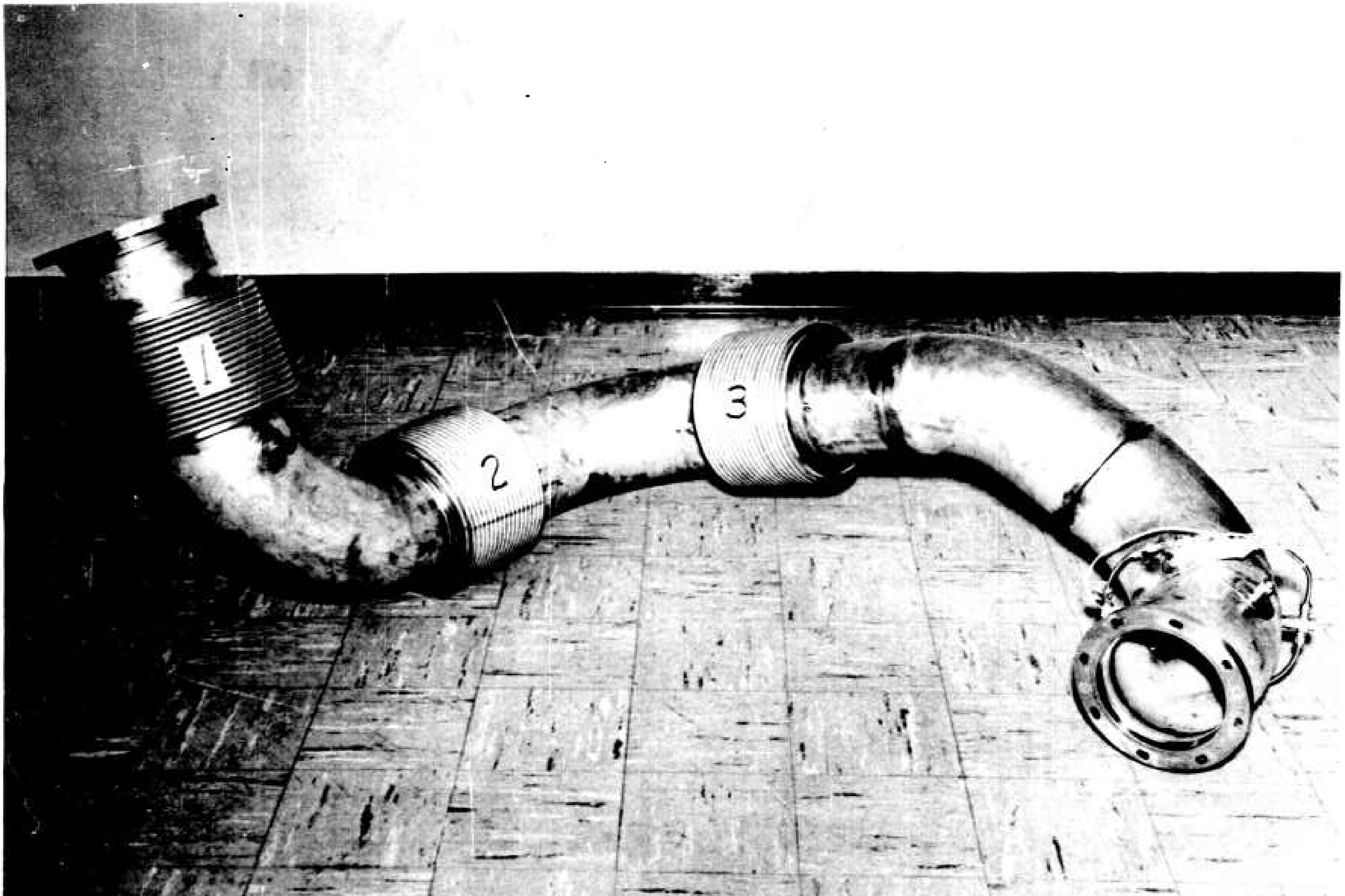


FIG. 19. OUTBOARD LOX WRAP-AROUND ASSEMBLY

container pressure of 56 psia and a fuel container pressure of 17 psig for the initial flight vehicles with an individual engine thrust of 165K. In these vehicles, the pressure level is determined by the minimum net positive suction head (NPSH) required for a 4-4 engine cutoff scheme.

A progressive reduction of the NPSH requirements for the H-1 engine is anticipated due to proposed modification of the propellant pumps. Consequently, in later booster flights, which will use a 188K engine, the minimum fuel and lox container pressures will be determined by both the NPSH requirements during cutoff and the structural requirements at the point of maximum dynamic loading and/or recovery conditions. It is anticipated that the lox container gas pressure will be the same as present, due to the lower pump requirements, thus allowing continued use of present hardware if the basic pressurization scheme remains unchanged. The fuel pressure, however, is to be an absolute value rather than a gage value, since this provides a more constant pressure during flight. Although it was not the ideal choice, gage pressure regulating hardware was used on the first vehicles, due to availability.

The lox and fuel vent valve and some associated pressurization hardware had to be modified from JUPITER hardware for SATURN application. One redesigned item that required extensive testing for flight qualification was the heat exchanger. In order to carry through the philosophy of "engine-out" capability, a heat exchanger is included for each of the eight engines. The stainless steel heat exchanger (Fig. 20) used in development tests contained two, concentric, helical coils of 3/4-inch diameter stainless steel tubing with a total of 38 to 39 square feet of coil surface. The hot exhaust gas has a temperature of 870-875°F, a flow rate of 13 lb/sec, and a static pressure of 13 psig. The lox has a flow rate of 3.6 lb/sec at 660-665 psig inlet pressure. Heat exchangers of the above design produced gox at a rate of 3.6 lb per second with a temperature of approximately 400°F initially and decreasing to 150°F at 335 psig pressure thus satisfactorily meeting system requirements.

The biggest difficulty encountered during development testing of the SATURN heat exchanger was the occurrence of pressure oscillations within the coils on some of the tests. The effect of these oscillations on the engine or pressurization system performance has not yet been fully determined. However, these oscillations are considered undersirable and potentially detrimental. The oscillations occurred in the first four or six seconds of the test when the heat exchanger flows are attempting to reach a state of equilibrium. When occurring in a total time of about 90 seconds, the oscillations built up rapidly to as high as ±250 psi then decayed. The frequency was approximately 1/3 cycle per second.

Results of the heat exchanger tests indicated that the biggest factor affecting the occurrence of oscillations was the external "cleanliness" of the heat exchanger coils. Initial tests with a new or cleaned heat exchanger showed a pattern of most severe oscillation followed by a reduction in severity and cessation of the oscillations as the heat

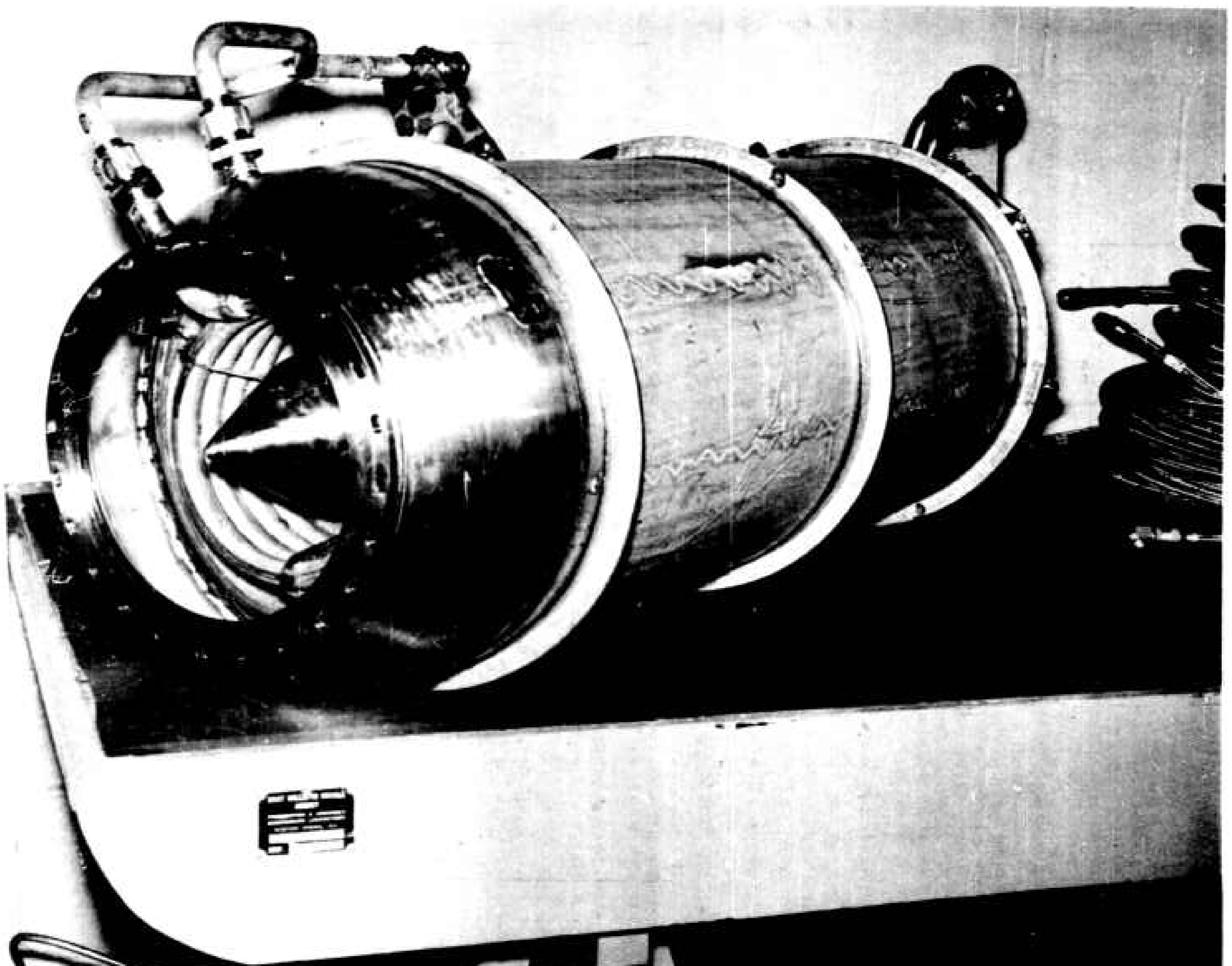


FIG. 20 HEAT EXCHANGER

exchanger coils became coated with carbon deposits from the turbine exhaust gases. Tests are now in progress to determine a suitable artificial coating or mechanical device for the new heat exchanger coils thus retarding the occurrence of oscillations during the flight operations of heat exchanger and eliminating any need for "running-in" the heat exchanger.

(b) Investigation of Advanced Systems. Feasibility studies of advanced pressurization systems for the SATURN booster have been in progress during the past reporting period in an effort to combine the best features of several possible schemes. It is anticipated that contracts will be let to develop the advanced system that best satisfies the SATURN booster requirements and that this system will be flown on later R&D vehicles upon completion of qualification testing of this new type of pressurization system.

The ideal pressurization system is a monocomponent system that is packaged above the tankage and away from the adverse environment of the engine area. The system would generate an inert gas of low molecular weight with no contaminants and at a temperature in the range of 500°F for use as a pressurant on fuel and oxidizer. The ideal system would have short feed lines, no filters, no rotating machine, a minimum of valves and controls; yet be controllable over a complete flow range. In addition the basic design features should provide maximum reliability, minimum space, weight and scaling for variable requirements.

Three catagories were considered and were compared with the present system which weighs about 10,000 pounds. The systems are:

1. Pressurization utilizing a solid propellant in three different schemes.
    a. Solid propellant exhaust (cooled or uncooled) introduced directly into the lox and fuel tanks.
    b. Solid propellant exhaust cooled in a heat exchanger and used directly on the fuel.
    c. Solid propellant exhaust to evaporate LN_2.
2. Storable monopropellant - Decomposition products cooled by a heat exchanger and used in the fuel tanks. Other gases of the heat exchanger are used for the lox tanks.
3. Gases other than N_2, with low molecular weights, utilized for both lox and fuel tanks.

The more advantageous schemes were investigated and compared with both the present system and the ideal system. A factor that reduces the effectiveness of the solid propellant schemes is the structural temperature limitation. The pressurant should not have an inlet temperature of over 500°F. The particular systems investigated and the merits and disadvantages of each are listed below:

1. Solid Propellants

a. Use of a lithium azide (LiN_3) combined with a suitable oxidizer and binder and burned as a solid propellant. Seventy-five percent of its weight is given off as N_2 at 1600°F. The exhaust is then cooled to 500°F by LN_2 injection.

Advantages: (1) Light-weight system (≈ 6000 lb)
(2) Single pressurant for both propellants

Disadvantages: (1) Controlled only by venting
(2) Lithium reacts with moisture to form hydrogen gas
(3) State of the art development required for a solid-propellant grain.

b. Use of solid-propellant exhaust gases to evaporate liquid oxygen or nitrogen in a heat exchanger; the gaseous oxygen or nitrogen is then fed to the lox tank at a temperature of 500°F. The solid-propellant exhaust is cooled in the heat exchanger and a portion ducted to the fuel tanks. In this scheme approximately 90% of the exhaust gases are dumped overboard.

Advantages: (1) Single heat exchanger
(2) Solid propellants readily available

Disadvantages: (1) Controlled only by venting
(2) Weight of system ≈ 8000 lb
(3) A multigas system is required
(4) CO_2 formed by solid propellant would carbonize a fuel (UDMH) forming a white chrystalline precipitate.

c. Use of a solid propellant to evaporate LN_2, supplied by a high pressure tank, in a heat exchanger. The gaseous nitrogen pressurizes both propellants and the solid propellant exhaust is dumped overboard.

Advantages: (1) Ease of controllability
(2) Compact packaging on top of containers
(3) Availability of solid propellants

Disadvantage: High weight of system ≈ 10,000 lb.

(All the above schemes use a solid-propellant motor in a position above the propellant container. This could introduce difficulties in disposing of the waste gases.)

2. The most advantageous system investigated in the storable monopropellant catagory was the hydrazine decomposer which generates hydrogen, nitrogen and ammonia. The decomposition products are

given off at a temperature of 1150°F and passed through a heat exchanger which superheats LN_2 to 500°F for lox tank pressurization. The hydrazine decomposition products provide a compatible pressurant for the fuel tanks.

Advantages: (1) Ease of controllability
(2) Decomposition products more compatible with fuel than solid propellant exhaust products

Disadvantages: (1) High system weight ≈ 11,000 lb. This weight could be substantially reduced by use of a pump (instead of GN_2 pressure) to feed the N_2 and hydrazine to the heat exchanger or by decomposition of a different type hydrazine
(2) Unreliable initiation of hydrazine decomposition requires use of a pyrotechnic ignitor.

3. The final catagory investigated was use of gaseous helium which would be stored in 3000 psia containers and cooled prior to launch by LN_2. During flight the helium is fed through a heat exchanger in the turbine exhaust. The heated gas is recirculated through regenerative heat exchangers in the helium spheres and then fed into the fuel and lox tanks at 500°F.

Advantages: (1) One Pressurant for fuel and lox
(2) Lowest weight system ≈ 4000 lb

Disadvantages: (1) Use of turbine exhaust in engine area
(2) Limited helium supply.

At this time it appears that a scheme involving the use of a solid propellant such as lithium azide presents the greatest potential for overall improvement in the SATURN pressurization system. Additional studies will be conducted before selection of a new system for the SATURN booster.

(3) Propellant Level Sensing and Slosh Measurements

The large liquid propellant quantities of the SATURN booster has necessitsted the development of a series of accurate, reliable and rugged pressure switches. These switches are required to prevent propellant overfill and initiate booster engine cutoff. The switches consist of twisted Bourdon tubes, inductance pick-offs, appropriate DC to AC inverters and trsnsistor switching circuits. These switches have no moving parts except for the Bourdon tube which has neither mechanical rubbing cr sliding surfaces. The anticipated accuracy expected is from 1/2 to 1-1/2% including pressure differentials and environmental conditions. Exchange of a single fixed resistor alters the pressure switch point.

Malfunction of the automatic lox or fuel filling control system would cause overfilling which is not allowable as the booster is not provided with overflow drains. To prevent overfilling, differential pressure switches will be installed in one fuel and one lox tank to sense propellant heads. When either propellant head exceeds the pre-setting, a signal from the pressure switch will stop propellant loading.

Variation in the mission of each vehicle requires an adjustable pressure switch. The lox and fuel switches are supplied with five pre-determined settings.

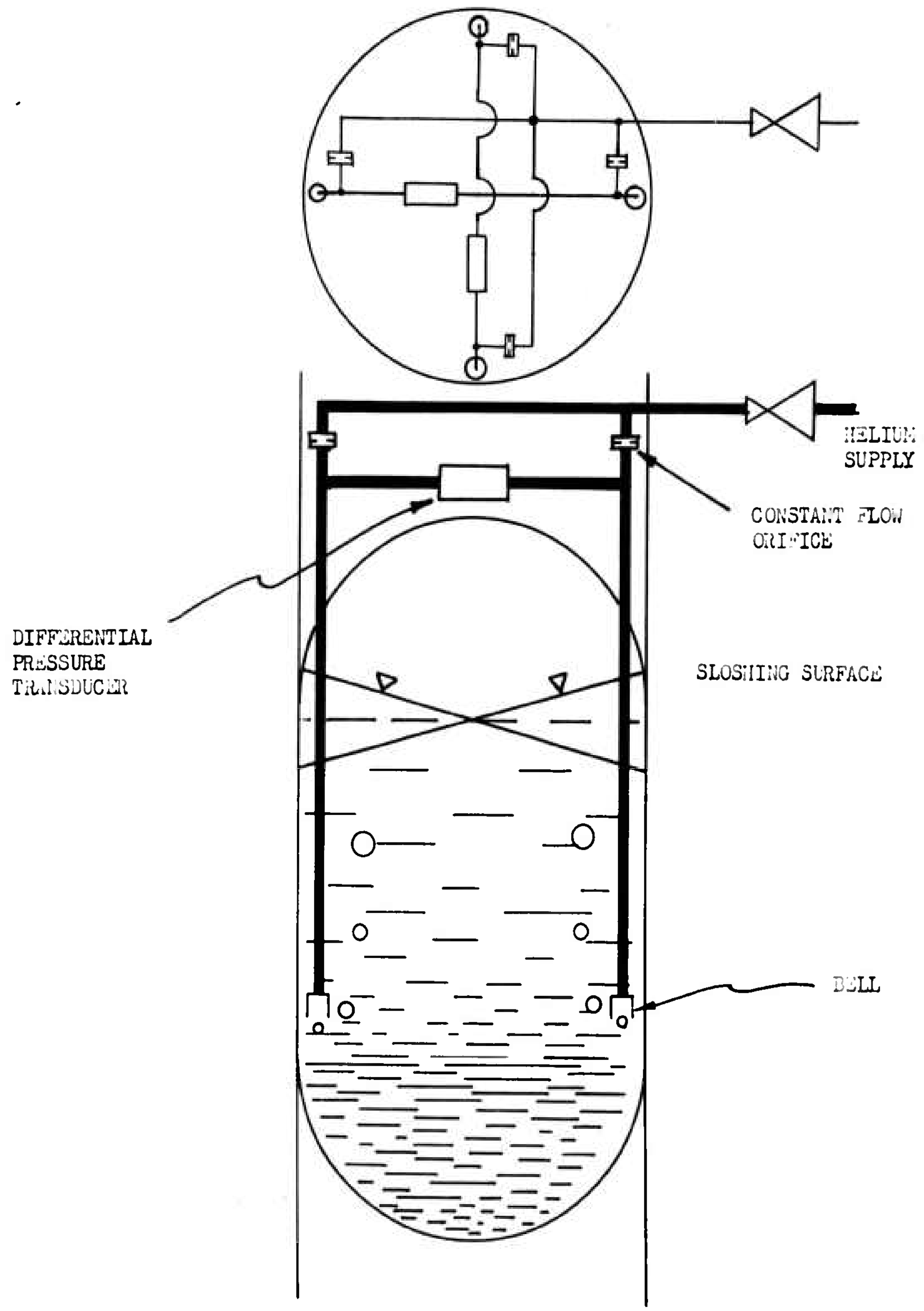
A back-up system to the pressure switches is a lox and fuel over-fill float switch system. Float switches will be installed below the tank vents of the No. 2 fuel tank and the center lox tank to prevent overfilling of propellants into the tank vents. Activation of these switches shuts off the propellant loading system.

Because it is desirable to have a minimum of residual propellants in the containers after cutoff, a scheme of cutoff initiated by propellant level, rather than vehicle velocity, has been selected for the initial flight vehicles. A shutdown scheme with cutoff of the four inboard engines followed by the four outboard is planned for control purposes. To accomplish the proper cutoff sequence, lox and fuel level sensors will be installed near the bottom of the propellant tanks. The sensors will be set to initiate cutoff of the inboard engines when the weight of propellant remaining equals a burning time of six seconds for the four outboard engines. The propellant, either lox or fuel, that reaches its sensor first will initiate cutoff, thereby preventing lox or fuel depletion. The lox level will be sensed by two discrete level gages (capacitance or acoustic type). Two gages are carried in case an engine that contains a lox level gage is prematurely cutoff.

The fuel level will be sensed by two differential pressure switches. The second switch provides a safety margin in the same manner as the second lox level gage.

The SATURN booster propellant tanks will be equipped with fixed accordian-type baffles. To determine the effectiveness of the antislosh devices, two sets of differential pressure measurements will be made in the center and number four lox tanks and the number two fuel tank (Fig. 21). Pressure transducers will measure the change of propellant height or head near the tank walls due to sloshing. To sense the pressure, two tubing probes with enlarged ends will be run within five inches of the bottom of the straight section of the tank. A differential pressure transducer will be connected between the two probes.

A helium purge system consisting of a regulator and a constant flow or sonic orifice will force the propellant out of the probe. The pressure that is required to force the propellant out of the probe will be equal to the pressure of the propellant at the probe end. The probe



TUBING ARRANGEMENT FOR SLOSHING MEASUREMENT
FIG. 21

will be purged by a constant helium flow of 1.25 cubic feet per hour. At this flow rate, the pressure drop in the tubing is negligible.

The constant flow orifice will be a very low flow rate centered metal filter. The filter will behave in a similar manner to a small hole orifice but will not be subject to clogging.

d. Environment Studies

(1) Compartmentization

To prevent a fire resulting from a fuel leak or a leak in the turbine exhaust duct from damaging an adjacent engine, the tail area of the SATURN, between the heat shield and firewall, has been divided into separate engine compartments (Fig. 2). This is to retain the engine-out capability. The desirability for a thermal barrier between engines has been shown by past tests in which engines have had a small fire caused by a leak in turbine exhaust or fuel lines yet have continued to function. These compartments will function in two ways: (1) they will prevent the direct impingement of flames from the small fire upon the heat sensitive components and critical wiring of an adjacent engine, and (2) in event of a major fire, which causes the loss of an engine, it is felt that the compartment walls will slow down the fire until the engine is shut down.

After studies and heat loading tests of several types of fire and heat, inconel wire-reinforced asbestos was selected as the material for use because of its capability of high heat resistance, low thermal conductivity and ease of assembly. This asbestos material may be coated with the silicone rubber paint. This material has been successfully tested at a temperature of 1700°F for three minutes.

(2) Fire Fighting

In the multiengine SATURN booster it is desirable to incorporate means of detecting and extinguishing a fire which results from engine malfunctions or line leakages. The first phase of fire fighting, the detection, has been investigated with no conclusive results to date. Several schemes were studied including optical, mechanical and electrical detection devices, but all of the suggested methods to date have serious drawbacks; principally the detection of a small fire and subsequent engine shutdown from a leak that does not cause malfunction of the engine. Work is also being conducted on the fire extinguishing phase of fire fighting, but to date no reliable and lightweight solution has been determined. Extinguishing agents considered include foams, gases, and dry powders. Some experimental testing was conducted using a dry powder (sodium bicarbonate) on fuel only and fuel and lox fed fires. Further testing and evaluation is required to determine the adaptability of this system for missile use.

As the entire concept of in-flight fire detection and extinguishing is new in the missile field, further investigations are required to find a reliable system that can be incorporated in the SATURN booster design.

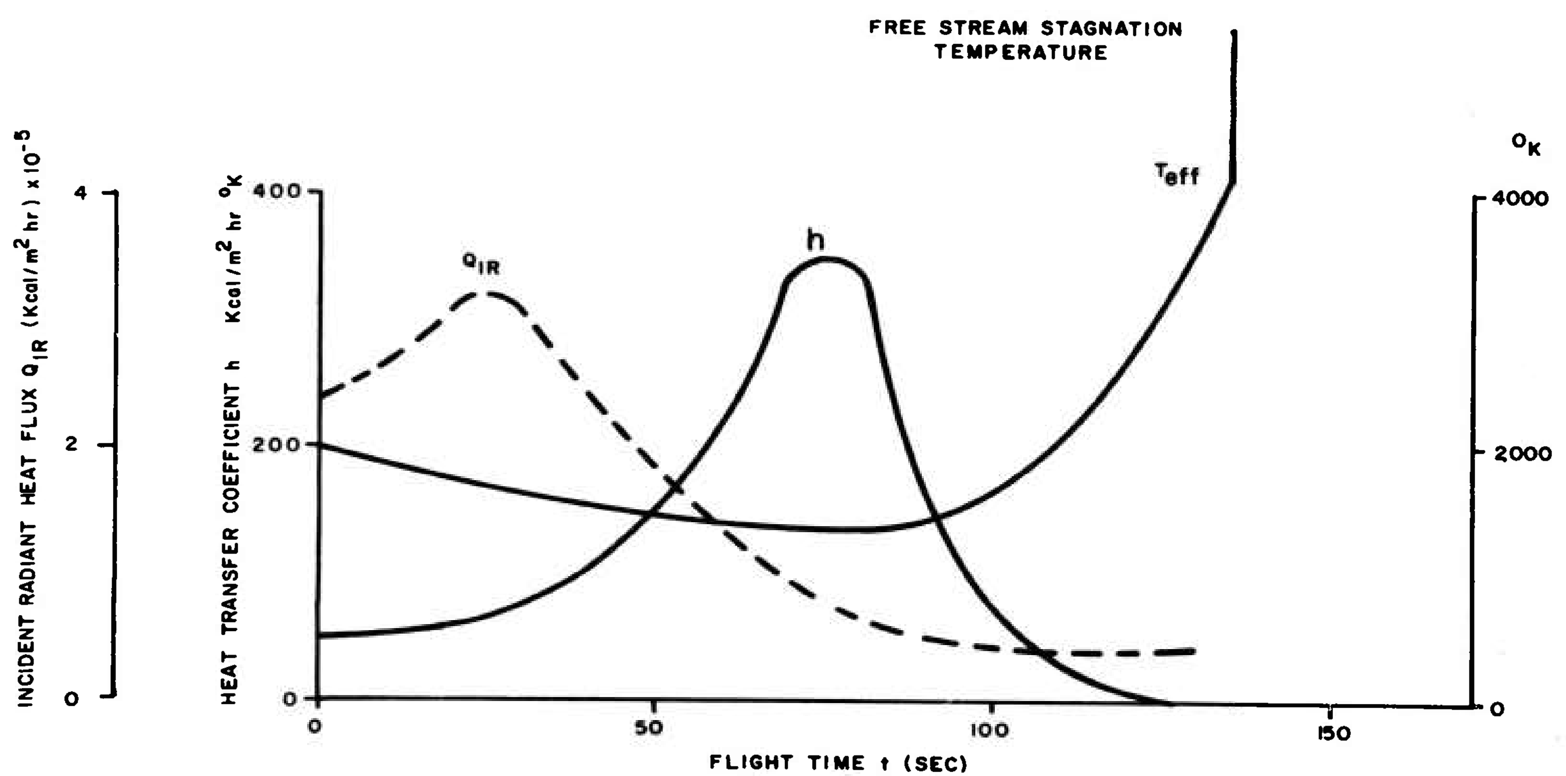
(3) Base Heating

The anticipated heat flux at the base of the SATURN booster during flight and on the test stand is of such a magnitude that the tail area (engine components and structural members) must be protected. The high heating potential is due to two factors: (1) radiation from the rocket exhaust gases and (2) a back-flow of the rocket and turbine exhaust gases into the boattail. The first condition is present in any tail design or engine arrangement, but the second condition is a function of a particular engine arrangement and tail area design. The initial booster design in the tail area will incorporate heat and flame barriers at two stations (Fig. 2). The immediate area around the gimballed outer engines and the fixed in-board engines will require a flexible material. The lower flame and heat barrier (flame shield) is primarily designed to prevent a backflow of rocket exhaust gases into the tail's center from the four inner engines when these hot jets interact due to under-expansion of these engines at higher altitudes.

The upper flame and heat barrier (heat shield) is designed to protect the heat-sensitive engine components and structural members from radiation heating and the back-flow of any hot gases (turbine or rocket) into that area. The present designs for the flame and heat shield were performed without the assistance of hot-jet wind tunnel data on the SATURN tail configuration. Wind tunnel tests of a SATURN booster incorporating eight 250-lb thrust motors is now underway at the NASA's Cleveland Facility and the tail design will be reviewed upon completion of these tests.

In order to provide a "safe" heat and flame shield design that allows for the unpredictability of flow conditions over a wide range of conditions in a flight trajectory, a conservative approach was taken until wind tunnel data became available and design optimization was feasible. The data shown in Fig. 22 derived from assumed gas temperature and flow conditions were used to determine the required heating protection.

Initial investigations for a heat shield design considered ablative materials, rockide coated steel, honey comb steel with a gold coating, and a silicon rubber paint on steel. Based upon the experience of the POLARIS missile with a silicone rubber paint in the tail area and upon ABMA's testing of this material, it was selected for use on the flame shield and also for inside portions of the tail shroud. A material of this type will be applied to a .040-inch stainless steel plate in a thickness of about 1/4-inch to form the heat shield consequently reducing the ambient air temperature between the heat shield and fire wall (Fig. 2) to a maximum of 200°F at cutoff. Tests were conducted with various samples of fiberglass cloth, Refrasil cloth, and an asbestos cloth



VARIATION OF INCIDENT RADIANT HEAT FLUX Q_{IR} HEAT TRANSFER COEFFICIENT h AND EFFECTIVE TEMPERATURE T_{eff} WITH FLIGHT TIME t FOR THE SATURN MISSILE

FIG. 22. ASSUMPTIONS FOR SATURN BASE HEATING ENVIRONMENT

reinforced with inconel wire for use as the flexible material required around the engine throat area. Refrasil cloth (two 4-inch layers) has been selected for use.

In the design of the flame shield, two severe conditions had to be overcome: the pressure and temperature on the flame shield resulting from the back-flow of the inner four fixed engines. The back-flow is due to the jet interactions resulting from the exhaust flames spreading with altitude. The back-flow of gases is estimated to produce a static pressure of about 3 to 5 psi greater than ambient pressure and a maximum heat transfer coefficient of 600 K cal/m^2/hr°K. With certain assumptions on gas flow directions and an estimate of gas temperature and radiant heat flux, 2800°K and 330,000 K cal/m^2hr respectively, a convective heat flux of 580,000 to 1,500,000 K cal/m^2hr is produced, assuming the flame shield to be at temperatures of 1800°K and 3000°K. As the higher temperature (3000°K) of the flame shield corresponds to the surface temperature of an ablating material, CTL-301, a design using this material was adopted. The present design uses a sandwich type structure consisting of 1.3 inches of CTL-301, and a .125-inch steel sheet to carry the pressure loads. All supporting structure of the flame shield will be coated with the silicone rubber paint used on the heat shield and shroud.

The fire wall is located above the engines and below the outriggers (Fig. 2). It is designed to prevent flames and excessive heat from a fire in an engine compartment from damaging the mass of electrical equipment above the fire wall and weakening structural members located in the same area.

The portion of the fire wall between the 105-inch corrugated barrel and the shroud consists of metal, sandwich type construction with an overall thickness of about 5/32 of an inch. The circular 105-inch portion under the barrel will be aluminum sheets of .08-inch thickness. The entire fire wall will also be covered with the silicone rubber type paint.

(4) <u>Future Objectives</u>

A possible method to reduce SATURN booster weight, simplify tail area design, and increase accessibility to the engines would be to change from the present completely closed design to an open tail area having no aerodynamic shrouds and no heat shield. Flight and wind tunnel experience must be accumulated on the present configuration before this approach is attempted; because of the many possibilities it offers, some preliminary studies, covering the environmental area, have been conducted on methods to protect the critical engine components while decreasing weight and increasing accessibility. Based upon current heating potential estimates, a scheme which covers each individual engine with a bag is under investigation. Materials selected for consideration include asbestos (reinforced with inconel wire) and a sandwich design with a thin outside wall of Refrasil cloth (4 mm), a low density insulation material (foamsil, Refrasil, batt and cerafelt), and a layer (2 mm) of asbestos reinforced with inconel wire on the non-exposed surface. Insulation thickness of 15 mm produced inside temperatures of 600°F, 425°F, and 200°F, respectively. Another approach would be to coat the exposed side of an insulative cloth with a silicone rubber paint. In the near future, extensive temperature and vibration tests will be conducted on the materials listed, as well as a comprehensive weight comparison between the present configuration and a future open tail design to determine the overall benefits of an open tail area from the environmental viewpoint.

3. Control System

a. (C) General Control Scheme

The basic concept of the control system for the SATURN vehicle has not been altered during the period covered by this report. The SATURN will use the ST-90 (Fig. 23) adapted from the JUPITER and modified to a digital output. Minor changes, however, such as the introduction of a SATURN guidance repeater, and the elimination of velocity cutoff (for the initial booster recovery flights) have been incorporated in the control system as shown in Figure 24.

The many configurations studied during the past six months have delayed the finalization of several portions of the control system represented by the dotted lines in Figure 24.

Selection of the C-1 configuration has allowed work to start on design of the dummy second stage and initiation of a final round of studies covering vehicle sloshing, structural bending frequencies and control frequencies. The shorter length of the C-1 vehicle (as opposed to earlier configurations) coupled with the possible increase of missile stiffness calls for a re-study of the requirements for the rate gyros and phase shaping filter network. Other areas that remain to be finalized are the type of angle-of-attack meter (boom-type or local), and whether or not control accelerometers are included in the flight system.

b. (U) Engine Hydraulic System

The SATURN engine actuator system is an independent, closed loop hydraulic system designed to eliminate the need for an external hydraulic pressurizing source. Its major components are two electrically controlled, hydraulically operated actuators, a main hydraulic pump, an auxiliary motor pump, an accumulator reservoir assembly, a manifold assembly, and necessary filters, check, relief, and bleed valves, and monitoring instrumentation. The filling and purging system installation provisions are located on the dome of the thrust chamber.

Each of the eight SATURN actuators (Fig. 25) is an integral unit consisting of an equal area, double acting cylinder, and electro-hydraulic servo valve; an internally mounted potentiometer; externally adjustable snubbers; a manually operated prefiltration valve; a pressure operated cylinder bypass valve to relieve hydraulic locking of the piston for manual or mechanical actuation; a differential pressure indicator; a filter; a fine trim adjustment for externally setting the potentiometer and a mechanical locking device to hold the piston at midstroke. The actuators are attached on one end to an assembly mounted on the engine thrust chamber and to the missile frame on the other. When the electrically controlled servo control valve on either the pitch or yaw

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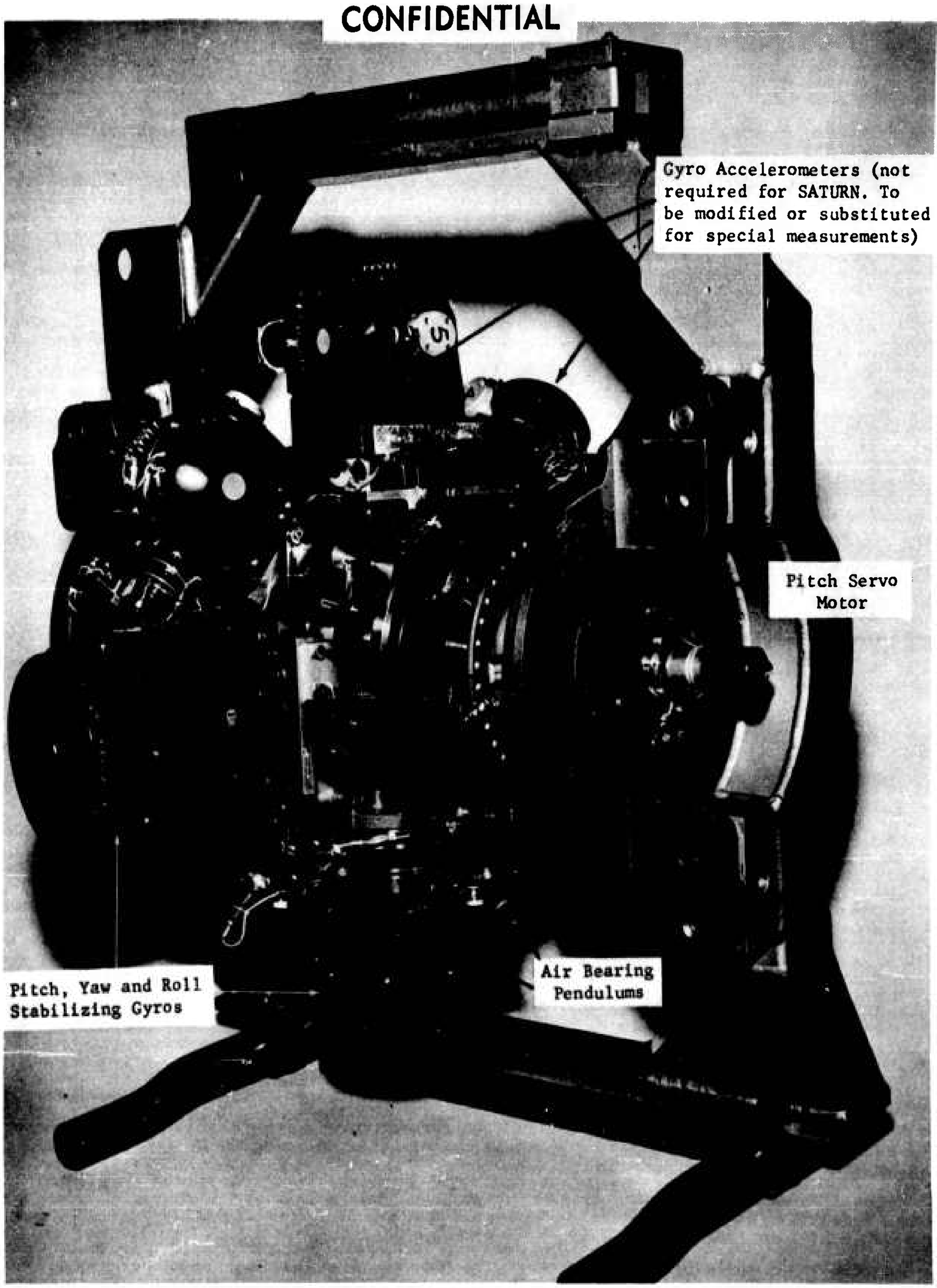

FIG. 23. JUPITER ST-90 STABLE PLATFORM TO BE MODIFIED FOR SATURN USE

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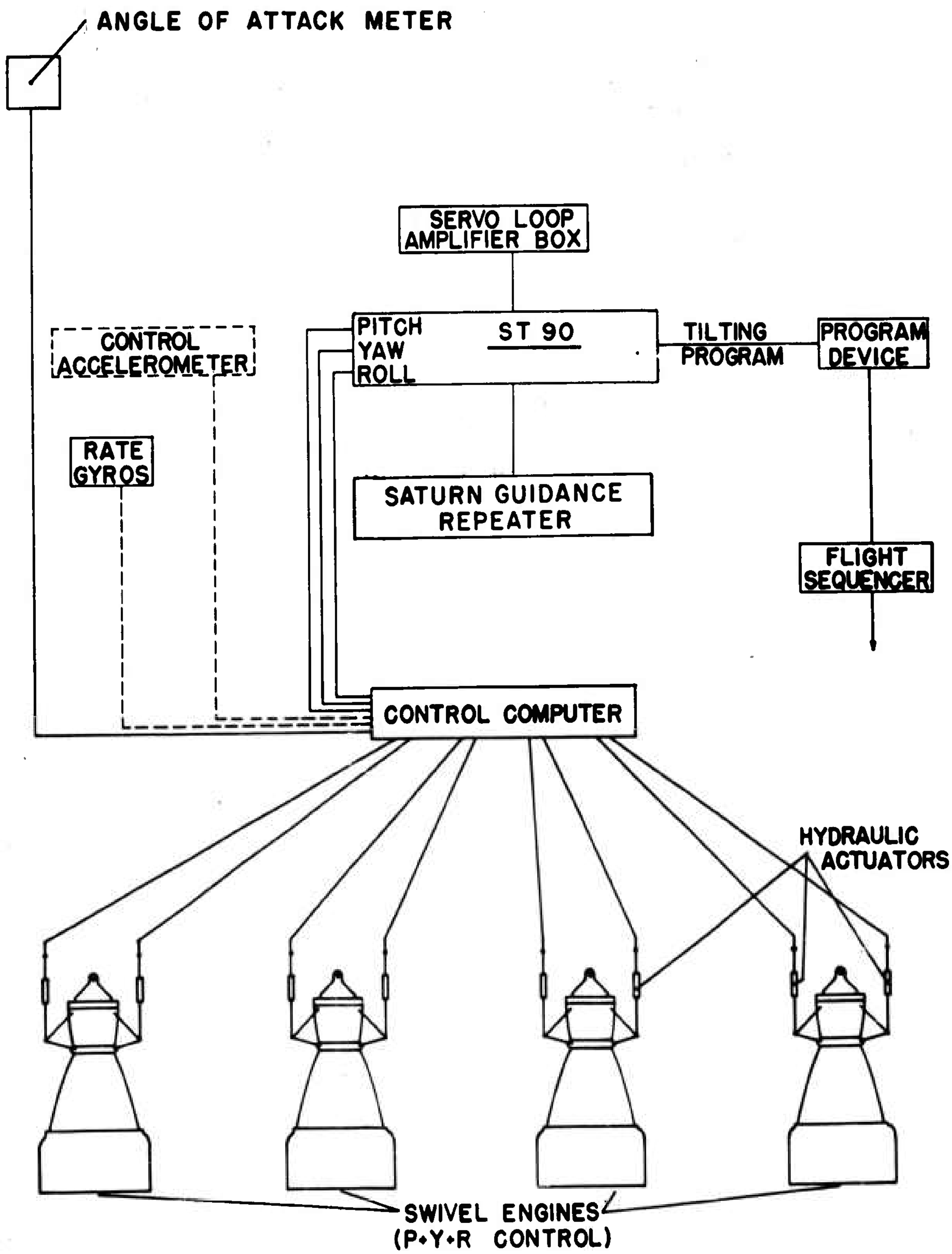


FIG. 24.BOOSTER CONTROL SYSTEM

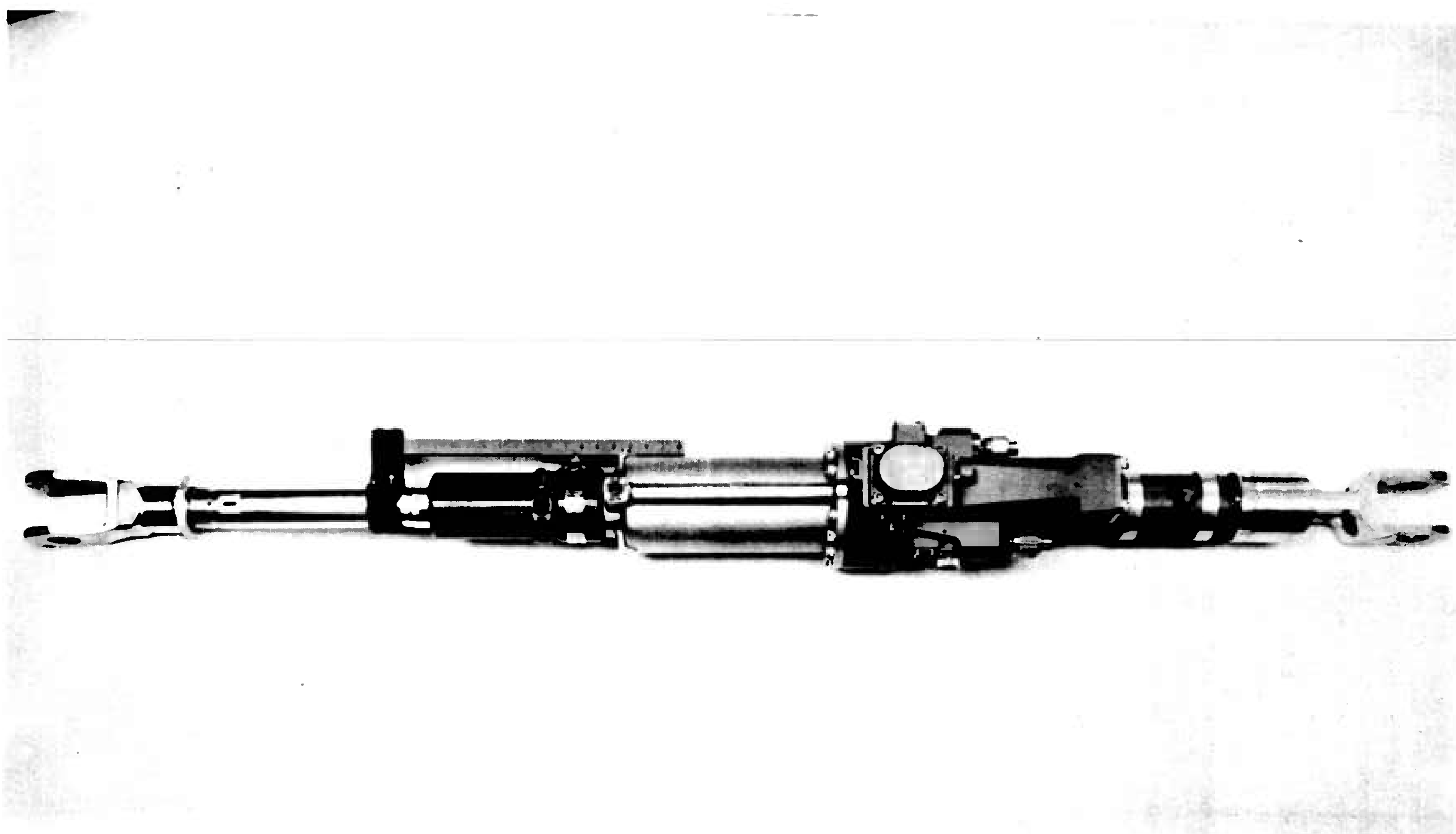


FIG. 25 ENGINE ACTUATOR

actuator receives a pure signal, a spool in the valve operates and allows delivery of high-pressure oil to one end of the actuator cylinder. At the same time, a return connection from the other end of the cylinder to the low-pressure side of the hydraulic circuit opens developing a differential pressure across the actuator piston. This differential pressure causes extension or retraction of the actuator arm at a rate proportional to the signal input to the valve and to the location of the valve spool. Piston velocity is a direct measure of the rate of hydraulic oil flow to the actuator. The main hydraulic pump driven by the engine accessory drive pad at 4,250 rpm delivers hydraulic oil at a rate of 17.5 gpm under 2,900 psi pressure to the engine hydraulic control system. The hydraulic system components are shown mounted on the H-1 engine mockup in Figure 17.

In order to eliminate the need of a ground hydraulic pump for checkout of the "cold" gimbal operations, a small electric motor and pump are mounted on board the booster. The auxiliary pump motor is bracketed to the thrust chamber between the turbopump and the lox position actuator.

The auxiliary pump mounted adjacent to the motor is a variable delivery type unit that delivers 3.0 gpm minimum at 2,900 psig ($\pm$50 psig) outlet pressure and 10,500 rpm.

The other two major components of the hydraulic system are the accumulator and the reservoir. The accumulator is a high-pressure, piston-type, pressure equalized unit of double wall construction. With its mating reservoir assembly, it is mounted to the thrust chamber between the turbopump and the lox position actuator. It delivers hydraulic oil to the actuators upon demand. A relief valve automatically prevents excessive hydraulic pressure in the accumulator. A valve installed in the accumulator is used to charge one side of the accumulator with nitrogen prior to filling, cleaning, and bleeding the system. The reservoir provides storage for hydraulic oil and allows for changes in oil volume due to temperature variation and leakage within the hydraulic system and furnishes oil to the pump inlets. A linear, single element potentiometer, installed in the reservoir, gives an external indication of the internal hydraulic oil volume.

c. (U) Instrument Compartment Cooling System

(1) General Requirements

The GSE external cooling system package must be capable of rejecting heat from the instrument canisters at a rate of 21,170 BTU/hr (6.2 kw), be light enough to be mounted on top of the umbilical tower, and be capable of complete separation from the umbilical pre-cooling valve connection at missile liftoff. The cooling system must be capable of operating continuously for extended periods of time.

The canister cooling system must maintain the canister ambient air temperature between 10° and 40°C during the complete preflight period; cooling is not required in flight. The ST-90 cooling system must maintain the ST-90 ambient air temperature during preflight operation at 25°C ± 2° at the gimbal joints with a drift of no more than 1°C per hour and the entire ST-90 compartment air at 25°C ± 3°.

The Azusa transponder cooling system must maintain the Azusa skin temperature at between 21° and 49°C during pre-flight operations.

(2) Description

The instrument compartment cooling system of the SATURN booster dissipates the heat generated by the guidance, control, and measuring equipment located within the on-board instrument canisters and the heat transferred to the missile from the environment (Fig. 26).

The system is divided into subsystems as follows:

a. Ground support equipment, external cooling system package

b. Missile internal cooling systems:

Instrument canister cooling.

Stabilized platform compartment cooling.

Azusa transponder cooling.

The GSE external cooling system package is the prime unit of the cooling system. It is located on the GSE umbilical tower adjacent to the booster unit. This package furnishes cooled air and nitrogen vapors to the missile instrument canisters where it is effectively distributed to eliminate any "hot spots". The heated air is then exhausted from the canisters and recirculated through the GSE unit. Nitrogen gas formed by the evaporation of liquid nitrogen is utilized for supplementing the cooling and thereby reducing the LN_2 consumption. It is injected into the air ducts and mixes homogeneously with the air. Venting overboard through a pressure relief valve in the canister prevents overpressurizing.

The major components of the GSE package are the cooler assembly, dehumidifier, and blower assembly.

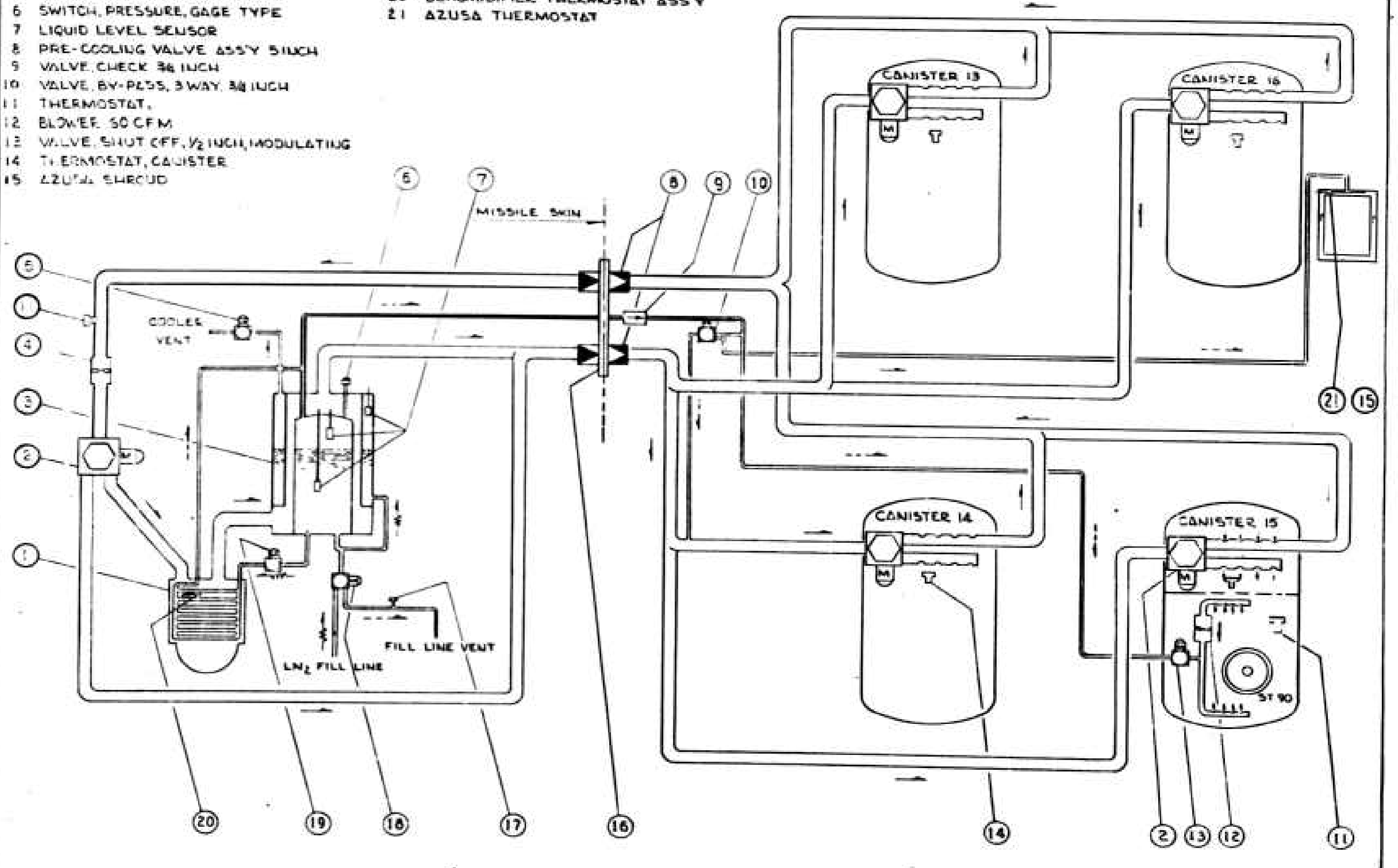
The cooler assembly is a 60-lb capacity liquid nitrogen container with automatic liquid level control. The container serves as a single pass liquid nitrogen to air heat exchanger. An air passage is located between an LN_2 filled inner container with fins and the outer annular container that also contains LN_2.

FIG. 26

SATURN COOLING SYSTEM

COMPONENTS

1 DEHUMIDIFIER ASS'Y
2 VALVE MODULATING IRIS-FLO, 4 INCH
3 EXTERNAL COOLING UNIT ASS'Y
4 BLOWER 300 CFM
5 VALVE 1 INCH MOTOR OPERATED
6 SWITCH, PRESSURE, GAGE TYPE
7 LIQUID LEVEL SENSOR
8 PRE-COOLING VALVE ASS'Y 5 INCH
9 VALVE, CHECK 3/4 INCH
10 VALVE, BY-PASS, 3 WAY, 3/4 INCH
11 THERMOSTAT.
12 BLOWER 50 CFM
13 VALVE, SHUT OFF, 1/2 INCH, MODULATING
14 THERMOSTAT, CANISTER
15 AZUSA SHROUD
16 DROP OFF PLATE
17 LIQUID LEVEL SENSOR
18 VALVE, 3 WAY BY-PASS, MOTOR OPERATED
19 VALVE, 1/2 INCH MOTOR OPERATED
20 DEHUMIDIFIER THERMOSTAT ASS'Y
21 AZUSA THERMOSTAT



Air flowing from the instrument canisters enters the cooler and passes through this air passage where the heat is rejected to the LN_2.

The dehumidifier removes moisture from the instrument canister air thereby reducing the possibility of snow formation in the cooler and ducting. A dual setting thermostat maintains the dehumidifier fin temperatures between -6°C and +6°C by control of the liquid nitrogen flow through the dehumidifier control valve.

A 500 cfm blower (rated at 18 inches of H_2O pressure) circulates the flow of air from the canisters, through the dehumidifier, into the cooler and then back into the canisters. A motor driven vent and relief valve is located in a vent line off the cooler to control its internal pressure.

A three-way control valve regulates the air flow through the cooler or directs the flow to bypass the cooler depending upon the temperature of the air returning from the canisters. This return air is monitored by a temperature sensor located within the return air duct.

Air and gaseous N_2O from the GSE unit enter the missile through separate ducts and are routed to the four instrument canisters.

The ST-90 stabilized platform compartment which is partitioned off from other guidance and control and measuring equipment instruments by an unsealed partition is cooled by nitrogen gas. The flow of nitrogen gas coming from the GSE cooler is controlled by a motor operated modulating control valve located within the ST-90 compartment. The gas is injected into the air stream of a mixing tube located downstream of a 60 cfm blower where it is mixed with air within the compartment and distributed around the ST-90 to maintain a uniform temperature.

Nitrogen gas from the three-way cryogenic motor valve is ducted into a diffuser within the Azusa transponder shroud. An upper and a lower limit thermostat maintain the proper temperature limits on the Azusa transponder skin by control of the three-way motor valve.

(3) General Test Results

Satisfactory preliminary tests have been conducted on the cooling system. Minor modifications to the system are being considered for controlling the relative humidity in the canisters to insure proper functioning of the commutated instruments.

4. (U) Instrumentation

a. Introduction

During the past six months the progress made in instrumentation area of the SATURN booster has been satisfactory and no serious technical problems have been encountered. The design of the telemetry system has been established. It was evolved from hardware developed for the REDSTONE and JUPITER; however, the system was expanded and given a higher capacity for high frequency measurements because of the larger number of measurements required by the eight-engine booster. The development of the SATURN peculiar hardware has been initiated.

The original telemetry antenna concept has been rejected because it failed to meet the requirements of the system. A new concept is being designed and will be ready for SA-1. The end organs for measuring the information to be transmitted by the telemetry system and the range safety devices will be basically the same as those used for the JUPITER.

b. Telemetry System

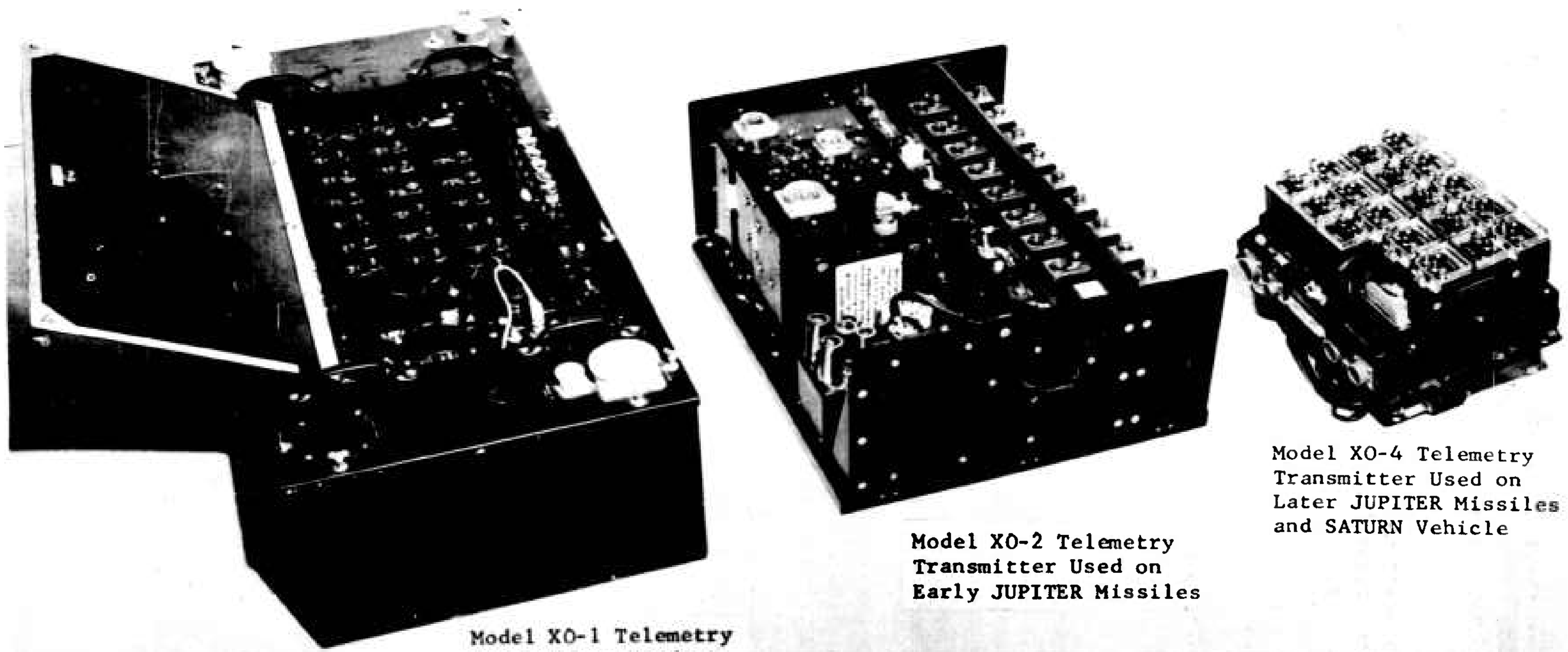
Much of the telemetry system developed for the SATURN booster has evolved from hardware used in the ABMA developed REDSTONE and JUPITER Missiles (Fig. 27). Since the SATURN booster contains a cluster of eight engines, the magnitude of the SATURN measurement program, when compared to that of previous missiles, considerably increases the requirements for telemetry channels. After the data requirements for SATURN were compiled and reduced to essentials, the total number of measurements was determined to be approximately 575. Consequently, some of the data had to be super-imposed and time shared in order to reduce the actual channel requirement to approximately 530. The majority of these measurements is low enough in frequency response to be commutated. However, 21 are vibration measurements that require a frequency response of 1000 cps or more. It is obvious that with the existing FM-FM equipment, the number of telemetry links would have been excessive and a new approach was called for.

In the solution to the problem presented by the SATURN vehicle the general criteria of previous designs was followed, as well as the following additional considerations:

a. The system must be, as far as practicable, compatible with existing receiving and recording equipment.

b. Existing data reduction equipment must be used to the fullest extent.

c. The data must be in such a form that preliminary flight evaluation can be made at the launch site by "quick look" methods.



Model XO-1 Telemetry Transmitter Used on REDSTONE Missiles

Model XO-2 Telemetry Transmitter Used on Early JUPITER Missiles

Model XO-4 Telemetry Transmitter Used on Later JUPITER Missiles and SATURN Vehicle

FIG. 27 **TELEMETRY COMPARISON**

d. In keeping with the principles of evolution and for reasons of reliability, economy, and familiarity of operational personnel, the system should use, whenever possible, proved components now used on the JUPITER.

With the above guidelines and requirements, a system consisting of six independent transmission links and a total capacity of 546 channels (Fig. 28) was devised where a single SS-FM link transmits high response data, a PAM-FM-FM link satisfies the greatest number of channel requirements, three triple FM links increase the number of low-frequency-response continuous data channels, and one FM-FM link completes the data transmission requirements for the system. The four types of subsystem used are described in more detail below.

(1) SS-FM System

The total data bandwidth capability of an IRIG FM-FM link operating at a subcarrier deviation ratio of 5 is approximately 4 kc. This is a baseband frequency utilization of less than 6%, obviously inadequate for wideband data transmission. At this point, it might appear that the best solution for such a problem would be bandwidth reduction by "pre-digesting" a considerable amount of the data onboard. Although systems are being developed by ABMA that will accomplish this to a large degree, there are several reasons why it is not the best approach for this particular application. The number of test flights for the SATURN is very limited with respect to other programs; therefore, all practicable data must be obtained. Data "pre-digestion" is a statistical tool in that it reveals only certain predetermined statistical data. For R&D flights, where it is necessary to reveal the cause of a specific malfunction, statistical data do not supply sufficient information. This is only one of several reasons why onboard data analysis will not be used extensively on the early SATURN boosters.

A subsystem being developed by ABMA for use on the SATURN, utilizing single sideband techniques, will have a baseband utilization efficiency of 60%. Compared to the FM-FM system, this is more than a tenfold increase in efficiency. The system basically consists of 15 single sideband AM subcarriers on an FM carrier, making the letter designation SS-FM appropriate. Figure 29 is a block diagram of the SS-FM subsystem.

The data input of each channel is fed to a balanced modulator and heterodyned with a 455 kc carrier. Neither of the inputs appears at the output of the modulator, only the sum and difference. The output of the modulator is fed to a mechanical bandpass filter which passes only the upper sideband (455-458 kc). A mechanical filter that meets on-missile environmental specifications, and has a shape factor of 1.5, a passband peak-to-valley ratio of 1.2 db, and a 5 db insertion loss has been developed by Collins Radio Company for ABMA. The filter is about 0.4 inch in diameter and about 2.5 inches long. The 455 kc frequency was selected in order to utilize standard tooling in the manufacture of the filters.

FIG. 28

TELEMETRY SYSTEM FOR SATURN

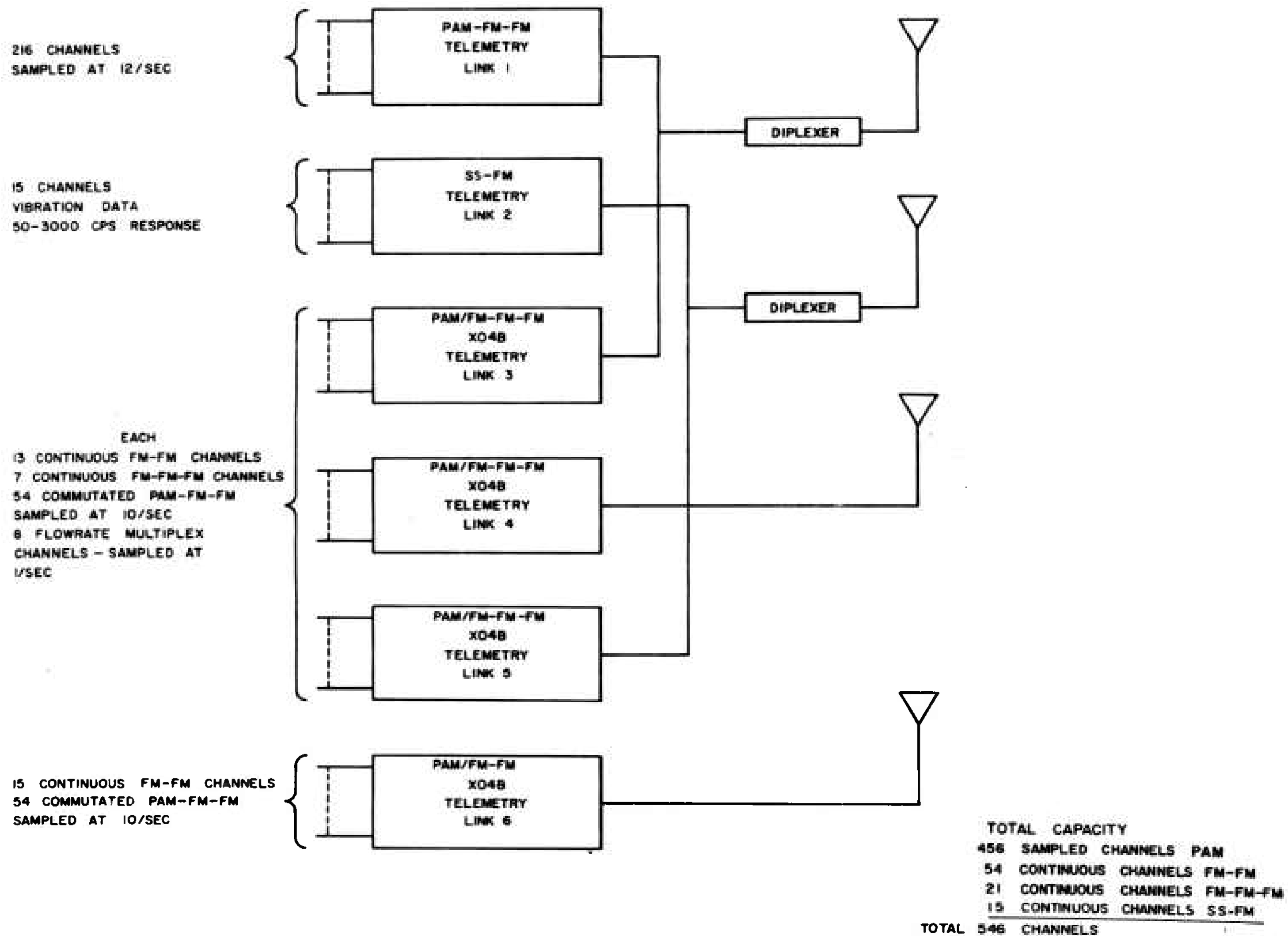
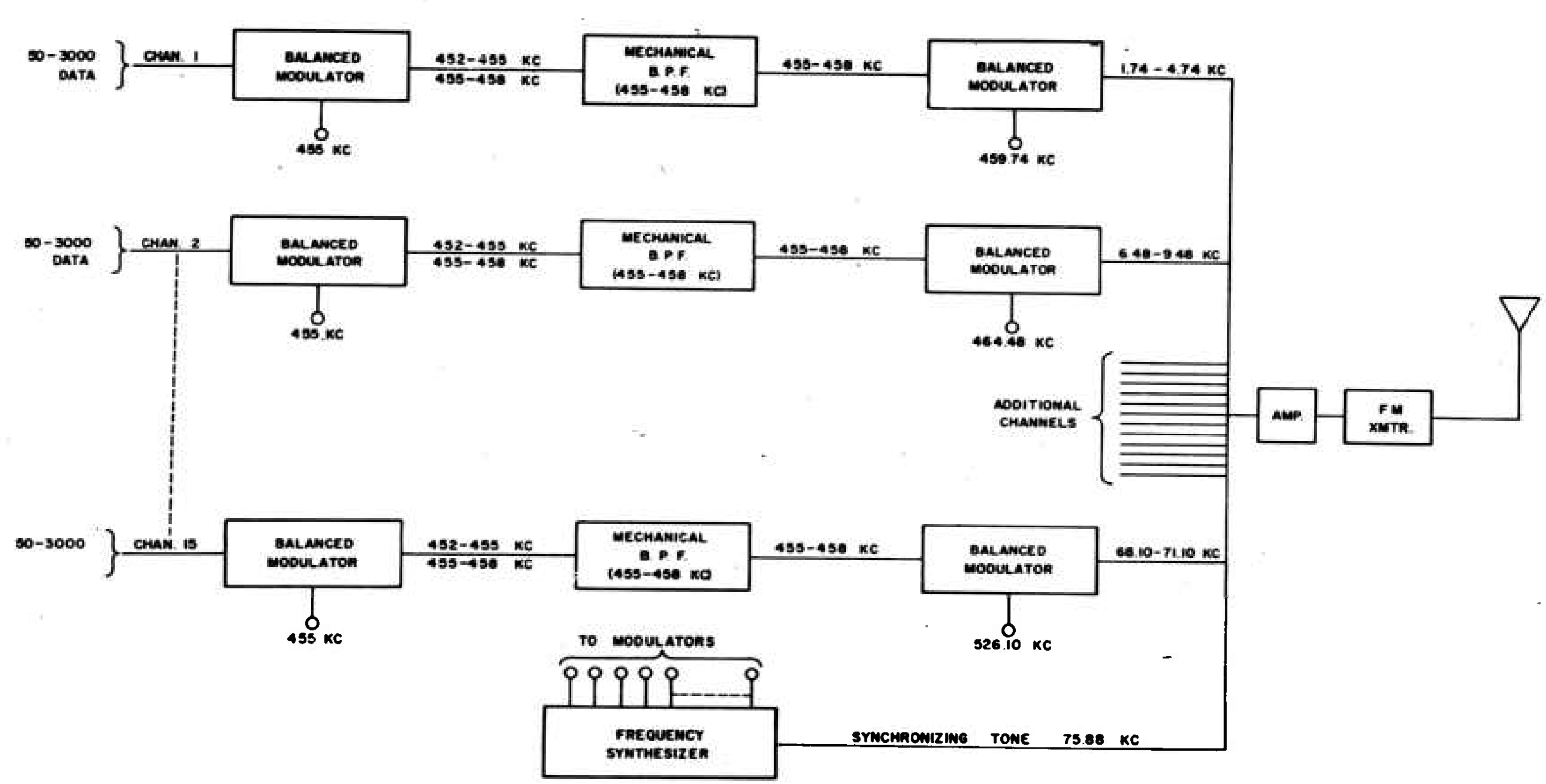


FIG. 29

SS-FM TELEMETRY SYSTEM FOR VIBRATION AND OTHER WIDE-BAND DATA



To minimize the effects of filter drifts that may attenuate the low frequencies, the carrier has been placed approximately 50 cycles inside the passband. As a result, data frequencies below 50 cycles are transmitted double sideband, causing phase distortion at these low frequencies. A scheme that will suppress the vestigial sideband by about 40 db is being exploited. If the circuit proves feasible, the low frequency response will be much less than 50 cycles. The system is presently considered to have a low frequency response limit of approximately 25 cycles. Where lower frequency vibration data are necessary, they can be filtered and applied to FM-FM channels.

The second modulator translates the filter output to an assigned baseband frequency. The baseband position is determined by the tone supplied to this modulator from the frequency synthesizer. The output circuitry of the modulator is sufficiently broadband to permit use of the same modulator design for any channel unit. It should be noted that the channel units are identical, satisfying the serviceability and flexibility requirements of the design criteria. The output of the channel units are mixed, amplified, and fed to a very linear FM transmitter.

The function of the frequency synthesizer is to generate the 15 carriers for the second modulators and a synchronizing tone for the ground equipment. To accommodate a 3-kc information bandwidth and allow a sufficient guardband, a channel spacing of from 4.5 to 5 kc was desired. For convenience in designing the frequency synthesizer, 4.74 kc were chosen, allowing an adequate guardband of 1.74 kc. With this spacing, the baseband spectrum extends from 1.74 to 71.10 kc. To facilitate slaving of the ground demodulators to the on-missile reference signals and to simplify on-missile equipment, all the reference signals are derived from one master oscillator. This oscillator is crystal controlled at 910 kc. A selective spectrum generator produces a spectrum of frequencies that are separated precisely by an equal spacing of 4.74 kc. The desired frequency components of the spectrum generator are separated by a set of comb filters, which are mechanical filters similar in construction to those used in the channel units. The output of each filter is amplified and fed to the relevant modulator.

The synchronizing tone was chosen to be 75.88 kc since it falls just above the highest baseband frequency and is available in the divider chain from the master oscillator. This tone is used as a reference in reconstruction carrier signals in the demodulation equipment. The tone has other functions, the most important being an AGC reference. Since most instrumentation tape recorders introduce amplitude modulation, AGC is necessary to eliminate this effect. The AGC feature is also used as a scale factor in correcting for on-missile automatic deviation control of the transmitter.

Conventional FM-FM telemetry receivers and tape recorders may be used for recording the data, deleting the necessity for purchasing additional equipment for receiving stations. At data reduction stations, the demodulation equipment is essentially the same as the on-missile hardware. The primary difference is additional output amplifiers and AGC detection and correction circuitry.

Later in the program, this system may be used to carry FM subcarriers, sound intensity measurements, commutation wave trains or voice communication. The flexibility of the system will undoubtedly lead to many diversified uses in the future.

The airborne portion of this system will be packaged in a volume of about 300 cu. in., weigh about 15 lb, and consume about 22 watts of power. These figures do not include the power amplifier that may be used for some applications. The amplitude accuracy is about 5%.

(2) PAM-FM-FM System

The greatest number of data channel requirements for SATURN are for data of low frequency response. More than 400 such measurements are to be made.

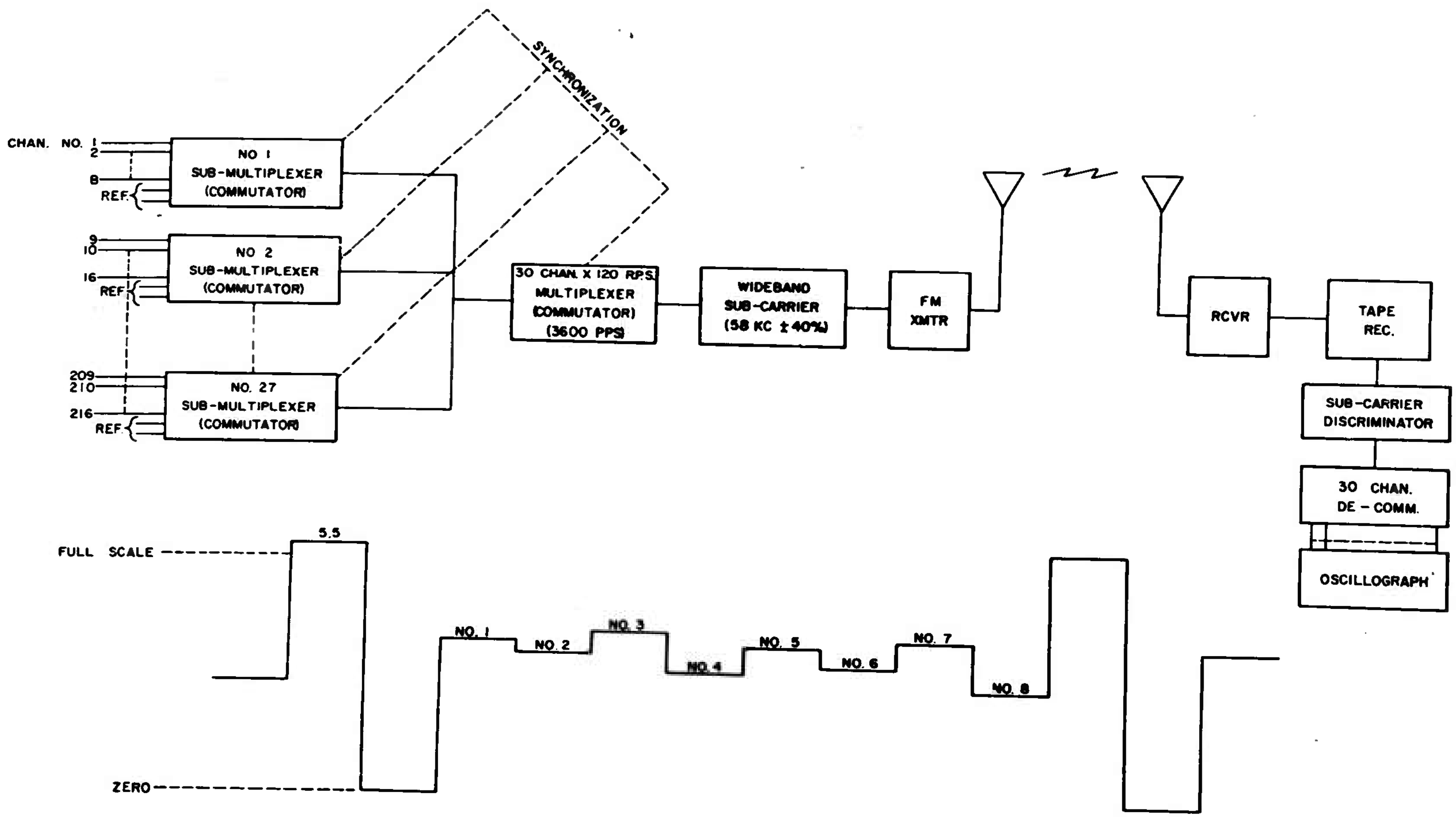
A 216-channel PAM-FM-FM subsystem has been designed for the SATURN which will adapt the system to the increased demand of commutated data channels. Figure 30 is a block diagram of the PAM-FM-FM subsystem.

The heart of this system is a 216-channel transistorized multiplexer. The number of channels was chosen so existing documentation equipment could be used for demodulation and at the same time achieve a quick look feature for comparing the performance of the eight engines. Each of the submultiplexers will sequentially sample identical measurements for each of the eight engines. For example, submultiplexer 1 may sample temperature lox pump bearing on each engine while submultiplexer 2 samples pressure at fuel pump inlet. Two channels of each submultiplexer are used for frame identification and voltage reference making a total of ten channels. The main multiplexer sequentially samples each of the 27 submultiplexers. Three additional channels are used for voltage reference and master frame synchronization. In order to maintain the accuracy that had been achieved by the mechanical commutators, special attention was given to the design of the transistor gates. As a result, a gate that approaches the characteristics of an ideal switch has been developed. The master frame format is identical to the IRIG PAM commutation format operating at a rate of 3600 pps. The Mach data channel is sampled 12 times per second. The data may be handled by the standard IRIG decommutator by playing back the tape recording at one-fourth the recorded speed, resulting in a rate of 900 pps. Some decommutators will handle 3600 pps in real time.

Figure 30 shows how the output of an oscillograph recording of one decommutated master channel would appear. In order to show the individual subchannels, the channel outputs are shown at different levels. If all the engines are functioning properly the channel outputs will appear

FIG. 30

216 CHANNEL PAM-FM-FM TELEMETRY SYSTEM



QUICK-LOOK OSCILLOGRAPH DISPLAY OF CHANNELS 1-8 practically as a continuous line on the oscillograph, enabling engine malfunctions to be detected at a glance.

If it is desirable to separate the individual data channels, this may be accomplished by feeding the output of each demodulator of the decommutator to a subdecommutator. To facilitate this, the submultiplexer frame identifying pulse is 110% of the master frame synchronizing pulse, allowing amplitude comparison techniques to be used for synchronization of the subdecommutator. Most data reduction centers can handle this format with no difficulty with automatic digitizing equipment.

The output of the multiplexer is fed to a 58 kc wideband subcarrier. This frequency was chosen in order to use a commercially available oscillator that will accurately carry the 3600 pps wave train and utilize existing demodulation equipment. For one-fourth speed playbacks, a 14.5 kc ± 40% discriminator plug-in may be used. For real time or full speed playbacks a 50 kc ± 40% plug-in is used. The FM transmitter is the same type used on the other links. Transmitter power of 2.5 watts will be adequate for this particular application if no additional SCO's are used. Low frequency SCO's may be added when required.

The total package for this system will weigh about 20 lb, have a volume of about 400 cu. in., and consume about 20 watts. The system accuracy is better than 3%.

(3) FM-FM Systems

The additional links necessary to complete the data requirements for the system will be made up of four XO-48 packages. The ABMA model XO-48 telemeter package is an FM-FM system with 15 channels of continuous data plus 54 multiplexed data channels. The multiplexed channels are switched by two 27-channel mechanical commutators and applied to the 22 and 30 kc channels, PAM at 10 rps. All data inputs are 0-5 volts. Inflight calibration is in five steps and is started by onboard program command. A special program plug is incorporated in the system so calibration may be withheld from channels where required. Pre-flight calibration is accomplished by command from the checkout console.

The XO-4 consists of components mounted on two chassis. One, the modulator unit, consists of the basic chassis on which is mounted the SCO's mixer amplifier, transmitter, commutator switch, commutator gating unit, blower, calibrator, time delay relays, program plug, 150- and 250-volt power supply, and plugs to receive missile cabling. Weight is about 26 lb. Most components are mounted independently so they may be removed readily without disturbing the other components. The second chassis mounts the SF power amplifier and its associated 350-volt power supply. This unit weighs about 11 lb. The volume of both units is about 800 cu. in. Recent tests of the JUPITER Missile have carried three of the XO-4 packages, and the units have proved, thus far, to be 100% reliable in flight.

(4) FM-FM-FM and Flowrate Multiplexer

To increase the number of low frequency response continuous data channels, three of the XO-48 packages will each have seven additional subcarriers modulating the 40 kc channel. Since these triple FM channels are less accurate than FM-FM, the triple FM data will be selected as those having lesser accuracy requirements. Figure 31 is a block diagram of this subsystem.

An auxiliary unit that plugs into the XO-4 package will contain the triple FM subcarriers as well as a transistorized nine-channel flowrate multiplexer. The multiplexer samples the flowmeter output from each engine sequentially. Each sample duration is approximately 1 second. The ninth channel is blank and is used for frame identification. To identify gating periods, a reference bias is built into the multiplexer. The bias alternates on consecutive gates.

Each engine has three flow measurements: main lox, main fuel, and low flow to heat exchanger. For single engine boosters the bandwidth was available to transmit flowmeter output continuously. In case of the SATURN this would require 24 continuous channels, each with relatively high response. Accuracy requirements prohibit the use of converters unless a means of accurate calibration can be devised. A flowrate multiplexer may be added to achieve this accurate calibration. The flowmeter outputs are converted to DC and applied to commutated channels. The outputs are also sampled at 9-second intervals by the multiplexer. These samples are used to calibrate the converted data when the data are evaluated.

Each triple FM package adds about 7 lb and 100 cu. in. to the XO-4 package.

c. End Organs

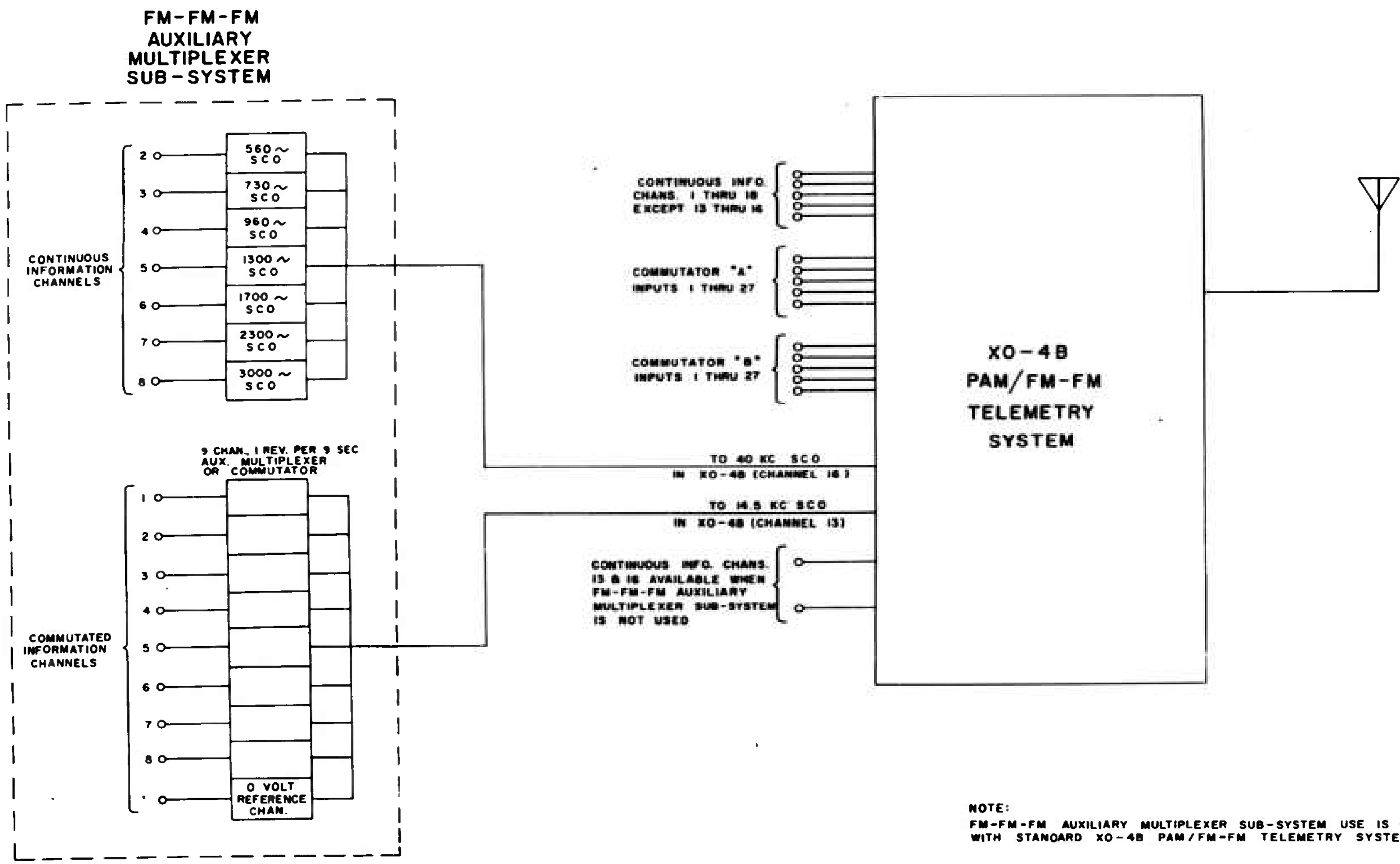
The basic end organs used to obtain the measurements transmitted by the telemetry system are listed in the following paragraphs. Table 7 gives the end organ and adapter requirements.

(1) Flowmeters

Turbine type flowmeters being used on the SATURN booster are similar to the JUPITER flowmeters with specially developed self-locking hanger suspension for added safety and reliability. Early difficulties experienced in tests using the teflon bearings were overcome by design changes. These flowmeters were subjected to eight times rated speed without loss of turbine blades. Development of flowmeters for the SATURN S-1 and S-2 is now complete.

FIG. 31

PAM/FM-FM-FM AND/OR PAM/FM-FM TELEMETRY SYSTEM FOR SATURN



NOTE:
FM-FM-FM AUXILIARY MULTIPLEXER SUB-SYSTEM USE IS OPTIONAL WITH STANDARD XO-4B PAM/FM-FM TELEMETRY SYSTEM

Table 7

END ORGAN AND ADAPTER REQUIREMENTS

Type of Measurement	No. of Meas. of This Type	Types of End Organs Required	Types of Adapters Required
Propulsion	230	18	6*
Temperature	174	24	4
Aerodynamic	13	5	5
Special	40	6	16

*Adapters are required to make various sensors with a wide range of outputs and impedances compatible with the standard telemeter with a standard input of 0 to 5 volts across 100,000 ohms. Only 58 of the 230 propulsion measurements require adapters.

(2) Pressure Transducers

At the initiation of the SATURN program there was no known source for pressure transducers with the ability to withstand vibrations in the range of 50 to 65 G's encountered in the SATURN without excessive errors. Recently several vendors have produced satisfactory potentiometer type transducers. Exhaustive environmental testing was conducted to select the item for flight use.

(3) Temperature Probes

Twenty-four different types of temperature end organs are required for the temperature measurements to be made on the SATURN. Of these, nine will be identical to end organs used in either the REDSTONE or the JUPITER programs. Nine will be similar to the prototype items but will incorporate design improvements or improved materials. The remaining six gages are new and will require considerable testing prior to use. DC amplifiers are required by 153 of the gages.

(4) Statistical Analyzer

The statistical analyzer to be used for onboard partial data reduction in connection with vibration measurements on the SATURN vehicles has been designed, fabricated, and satisfactorily ground tested.

(5) Measurements

Several measurements are called for on the SATURN vehicle which require modification of universal measuring adapters for range change, etc. Most of the special measurements can be made by minor adjustments of standard modules. One special adapter is required for use with a new type pickup used in the rough combustion cutoff.

d. Antenna Systems

(1) General

Originally it was planned to use two different types of antenna systems for the SATURN booster. One type was to give as nearly as possible an omni-directional pattern. This was to be accomplished by using a distributed array of loops or some other types of antenna. Several different arrangements were tried, but difficulties were encountered in obtaining the desired patterns and the idea was abandoned. The present plan calls for all eight telemetering links to be multiplexed into two antenna systems of the second type originally planned. Commercial multiplexers are to be used to feed four links into three "T" feed slot antennas spaced 120° mechanically apart and fed inphase. A total of six antennas will be used in the two arrays. Patterns have been made on this system, and bandwidth and altitude tests have been completed. Temperature tests are in progress at the present time. Vibration tests are to be run as soon as a flight model is obtained since all tests so far have been run using a laboratory model which is much weaker mechanically than the flight model is expected to be. The flight model is scaled in size in order to move the best part of the curve into the 240 to 260 mc range.

(2) Range Safety and Tracking Antennas

Antennas of the type used on the JUPITER are to be employed for the UDOP system. Two antennas are located near fins II and IV and will be diplexed using two printed circuit power dividers, one each for the transmitting and receiving sections of the antennas. Further investigations will be required upon completion of radiation pattern tests.

Work is continuing on the antenna design for the command destruct system, however none has satisfactorily met the present range requirements.

A JUPITER antenna, previously used for C-band radar tracking, has the required frequency range and radiation pattern for use in the SATURN Azusa system.

SATURN tracking radar will employ two S-band and two C-band antennas of the type previously used on the JUPITER Missile.

e. Range Safety Devices (Radio Command System)

The command receiver-decoder employed in the REDSTONE and JUPITER programs is unsatisfactory for use in the SATURN program. The JUPITER environment approached the limit of endurance for the existing equipment; therefore, development of an improved system was initiated early in the SATURN program. The improved unit was required for satisfactory reliability under the expected environmental conditions presented by the SATURN booster. In addition, considerable reductions were realized in weight, displacement, and power consumption. The old command receiver-decoder (AN/DRW-59) displaces 1,000 cu. in., weighs 20 lb, and requires 115 watts of input power. The AN/DRW-13 displaces 35 cu. in., weighs 2.5 lb, and requires 7.5 watts input power. This development is now in its final stages. Evaluation and acceptance tests are now in process and 16 units have been delivered.

5. (U) Fabrication and Assembly

A substantial effort has been expended toward improving the quality and efficiency of structural components fabrication. A major portion of this was the conduction of numerous studies and test programs to determine new welding procedures including selection and special acceptance methods for welding materials to improve strength, soundness and reliability of weld joints in the SATURN structure.

It was found that a number of difficulties encountered during welding according to methods used previously could be eliminated by the electronic evaluation of welding wire surface quality and utilization of low contact resistance measurements as a criteria for sheet cleanliness level. Additionally, improved tooling designs and fabrication concepts are being achieved by studying and experimenting in the following areas:

a. Measuring of magnetic field disturbances caused by the fixture construction.

b. Determination of three-dimensional movement of metal during welding.

c. Study of heat sink efficiency for various back-up bar configurations.

d. Modification of hold down systems and flexible clamps in welding tools.

e. Investigations of welding wheel pickups.

f. Research on automatic welding arc tracking.

g. Study of offset effects in aluminum alloy welds.

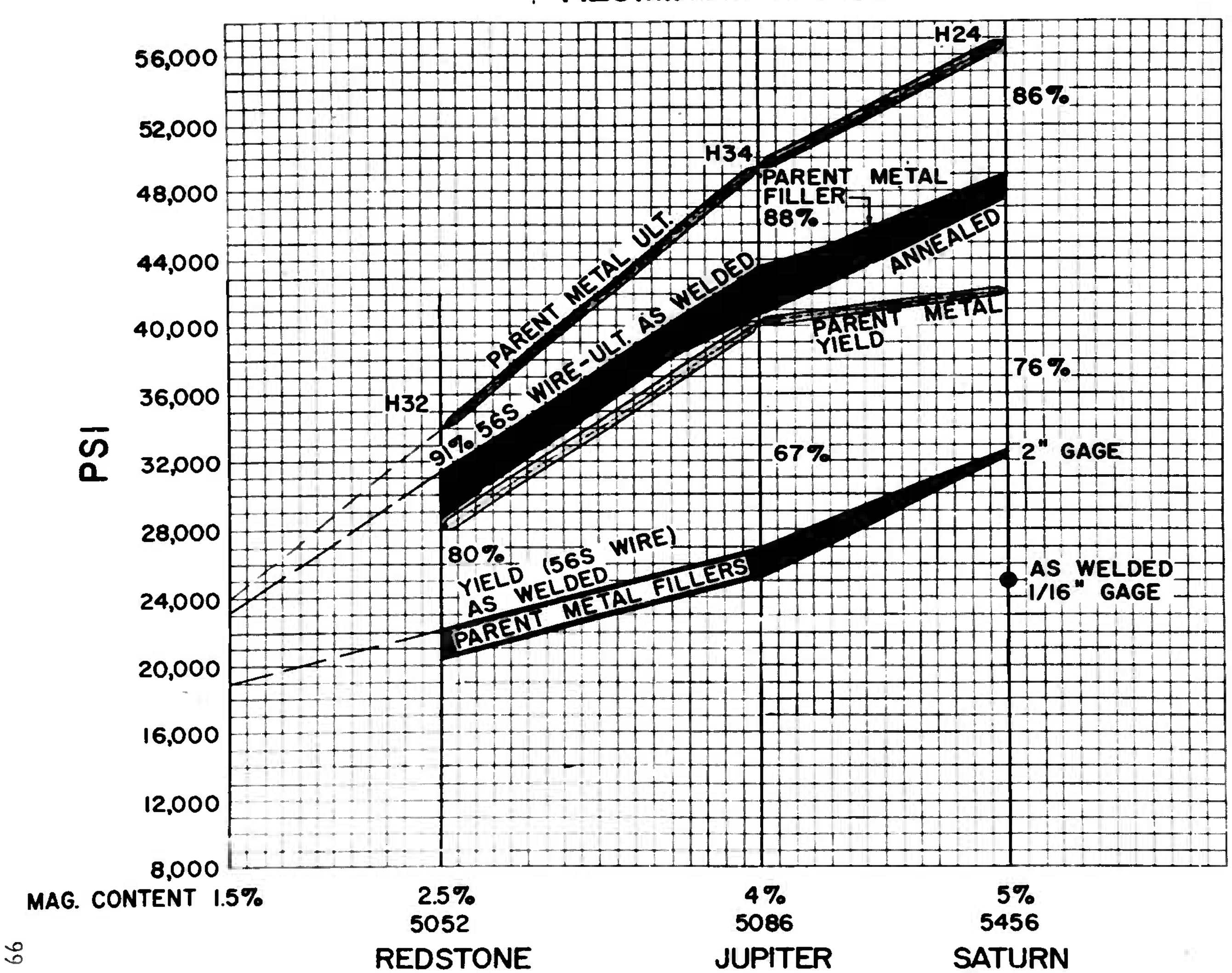
h. Application of plastic lightweight clamp and chill rings.

General results of this effort are reflected in Figure 32 which shows how weld joint efficiency and absolute strength have increased with the newer designs. The SATURN, which makes extensive utilization of the 5456 aluminum alloy, has achieved a weld joint efficiency of 76% for the 32,000 psi yield point.

Tooling and fabrication research has resulted in the development of a continuous spot welding technique, illustrated in Figure 33, which by sending timed 400 KVA electrical impulses through the continuously rotating rollers has reduced the operations time to 25% of that required by previous methods, while achieving considerably improved weld quality.

FIG. 32

COMPARISON OF STRAIN-HARDENING ALUMINUM ALLOY



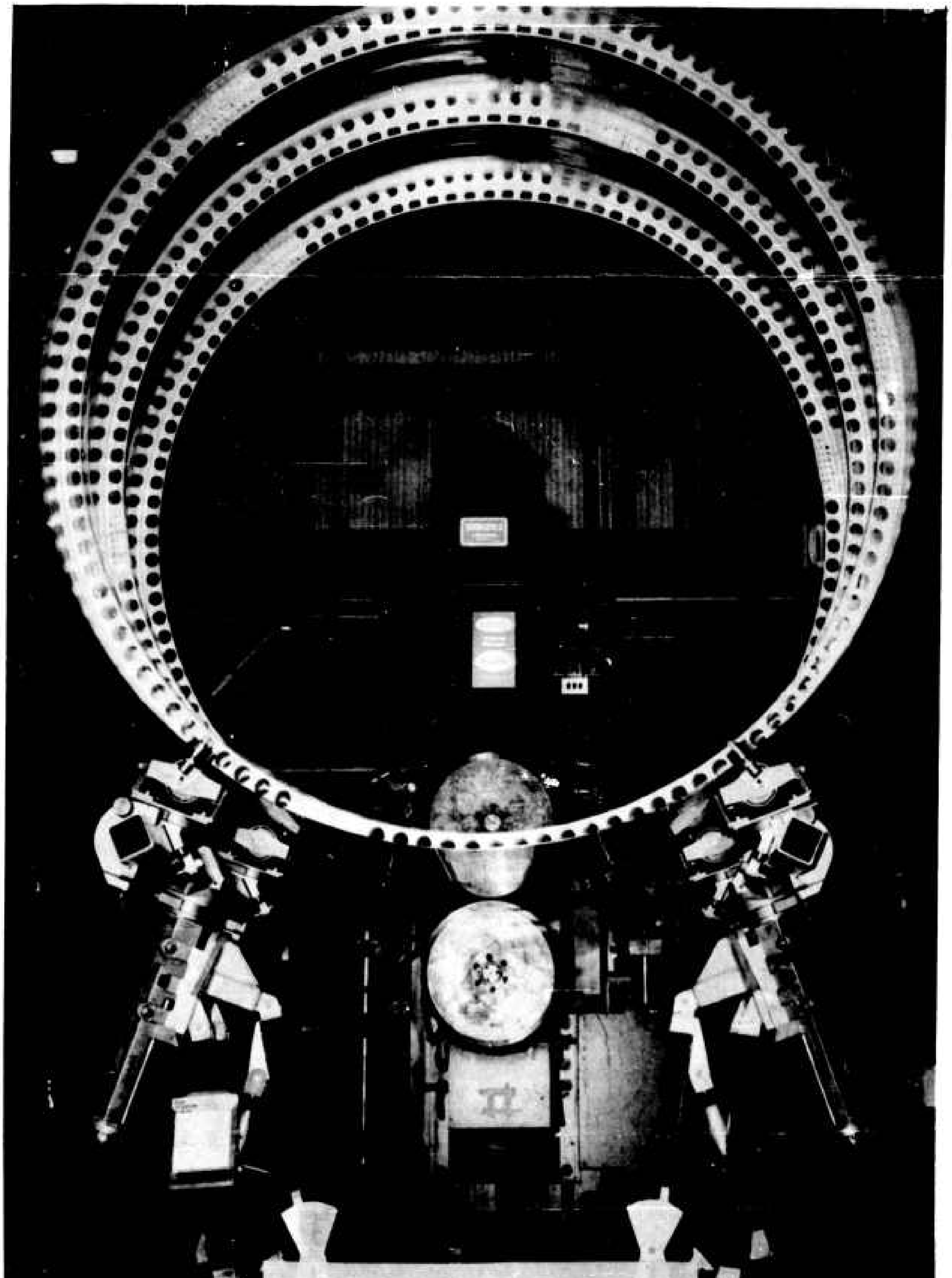


FIG. 33. ROUND ROLL SPOT WELDER

Experiments are being conducted with floating, flexible clamping rings used on the girth weld fixture (Fig. 34) which reduce the relative deflections encountered during circumferential welding between two thin skin segments, and minimize the out-of-roundness problems.

To develop a circumferential welding technique which is independent of the diameters to be handled, an experimental setup using new rolling girth weld concept was employed (Fig. 35). If this concept proves feasible, simplified tooling could be provided for various future missile applications.

Increased utilization of plastics in SATURN tooling for prototype development has resulted in faster and cheaper production and simpler modification procedures (Fig. 36). Parts developed during experimental fabrication were used as molds for dies and templates which are entirely adequate for use in the limited production of experimental missiles. An additional advantage in the use of plastics is weight saving which can be of considerable importance considering the many bulky items employed in SATURN.

Twenty-inch container stiffener rings are formed by means of hydraulic stretch forming (Fig. 37). This operation combines the bending and stretching of the component into one sequence. Sophisticated thin wall cross section members are produced by this method with greater efficiency while very close tolerances are held.

The horizontal mechanical skin milling machine was modified to handle two skin sections simultaneously, doubling the rate of skin milling operations (Fig. 38).

A 10-foot vertical boring mill was converted (Fig. 39) for use as a power shear forming tool in one-piece aluminum bulkhead fabrication. The technique produces hemispherical shapes of consistently uniform high strength characteristics. Chemical milling is applied to SATURN bulkheads in order to reduce structural weight (Fig. 40).

A high energy forming process is used in the fabrication of both the lox interfeed manifold and the lox tank pressurization manifold. Advantages of the explosion forming are that the number of welds are held to a minimum, and the work can be completed without intermediate anneals, still producing a precision formed high strength body. Up to 58% elongation is achieved for one quarter hard material. The successive steps of the operation are illustrated in Fig. 41. A small explosive charge immersed in water contained in a polyethylene bag is used to provide the forces required. A mandrel is drawn through the apertures to complete the operation. The resulting lox manifold is shown in Fig. 42.

Fig. 43 shows one of the 70-inch containers mounted in the hydrostatic test facility where a newly established vertical cleaning

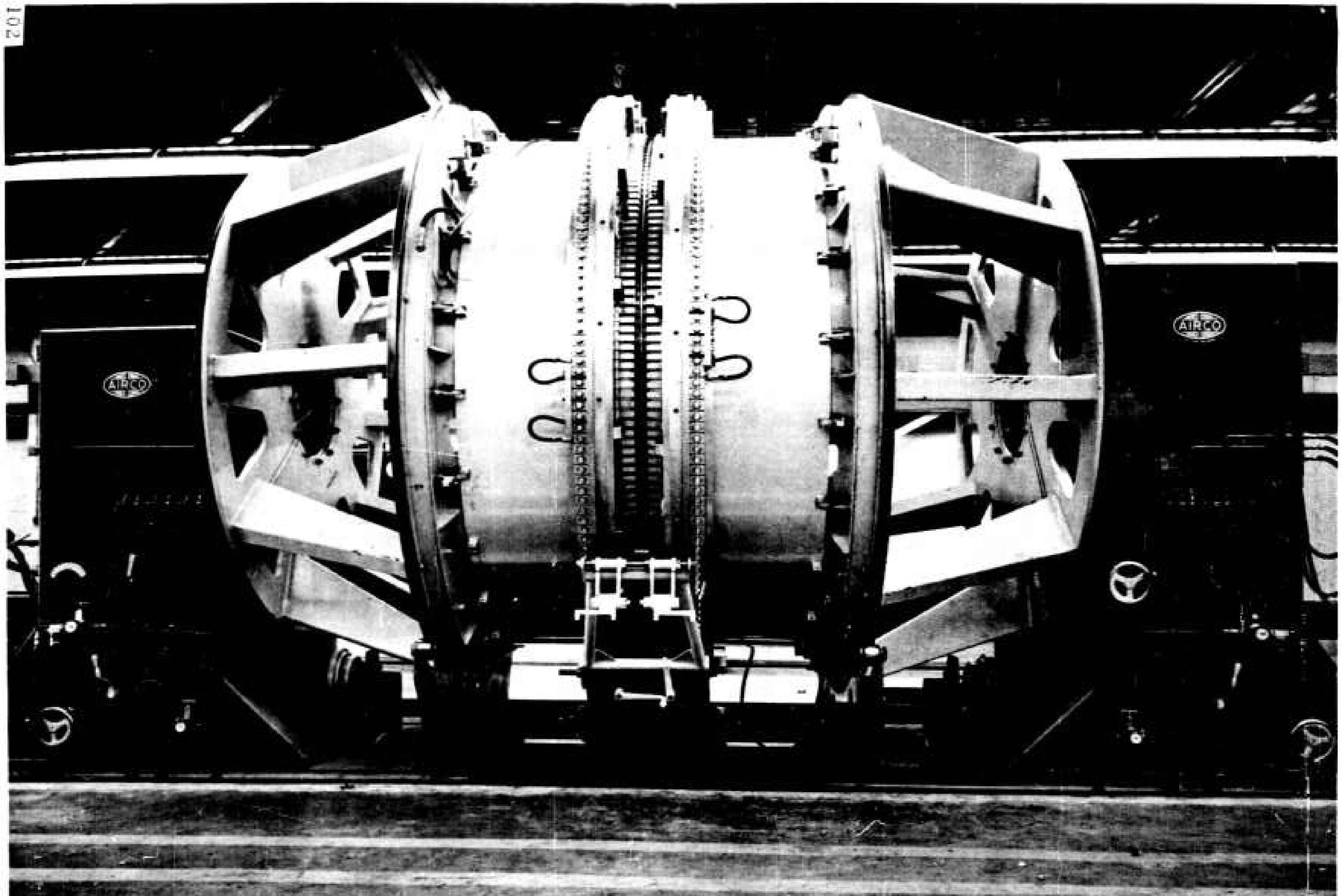


FIG. 34. GIRTH WELD FIXTURE

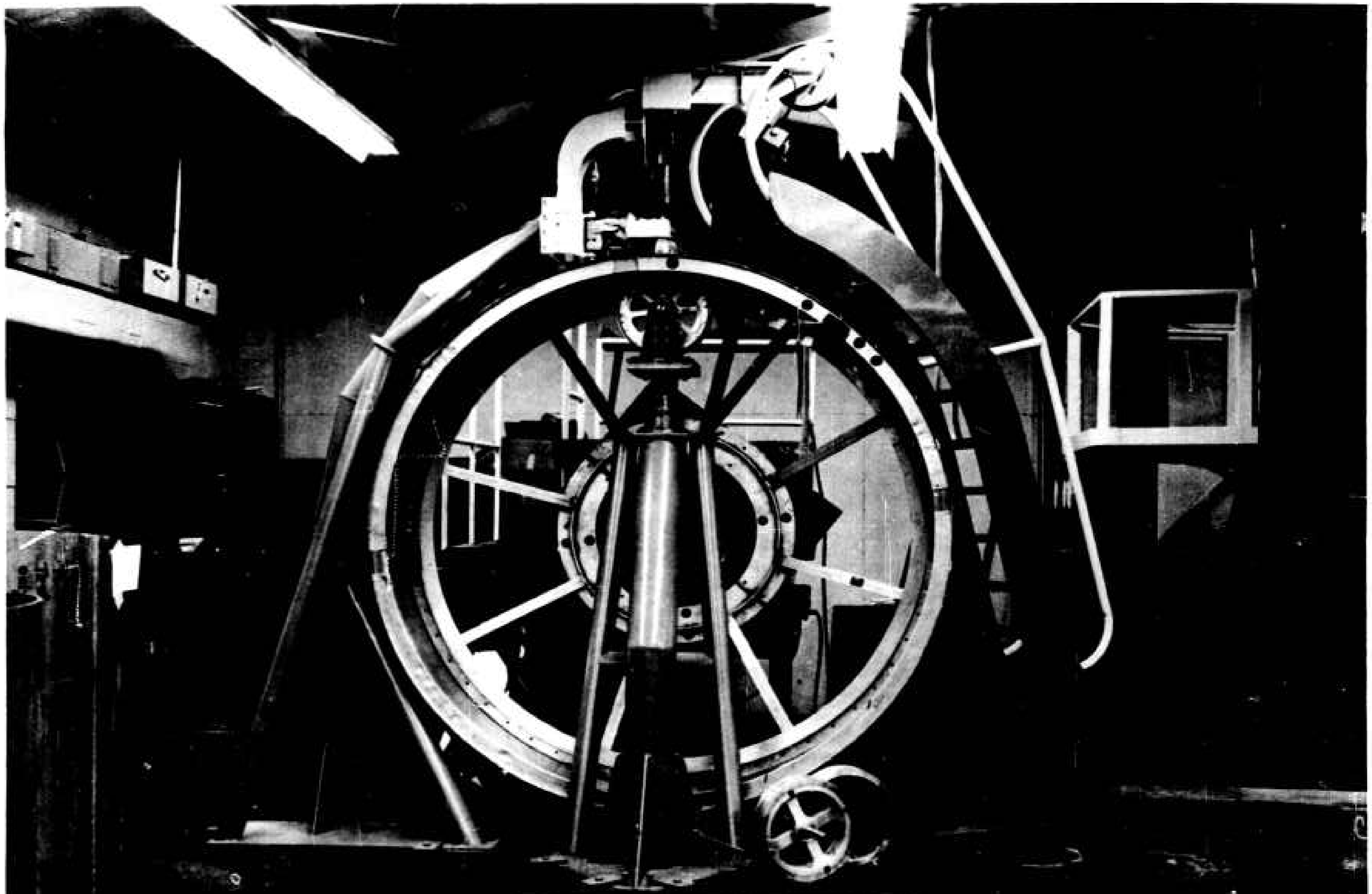


FIG. 35. ROLLING GIRTH WELD

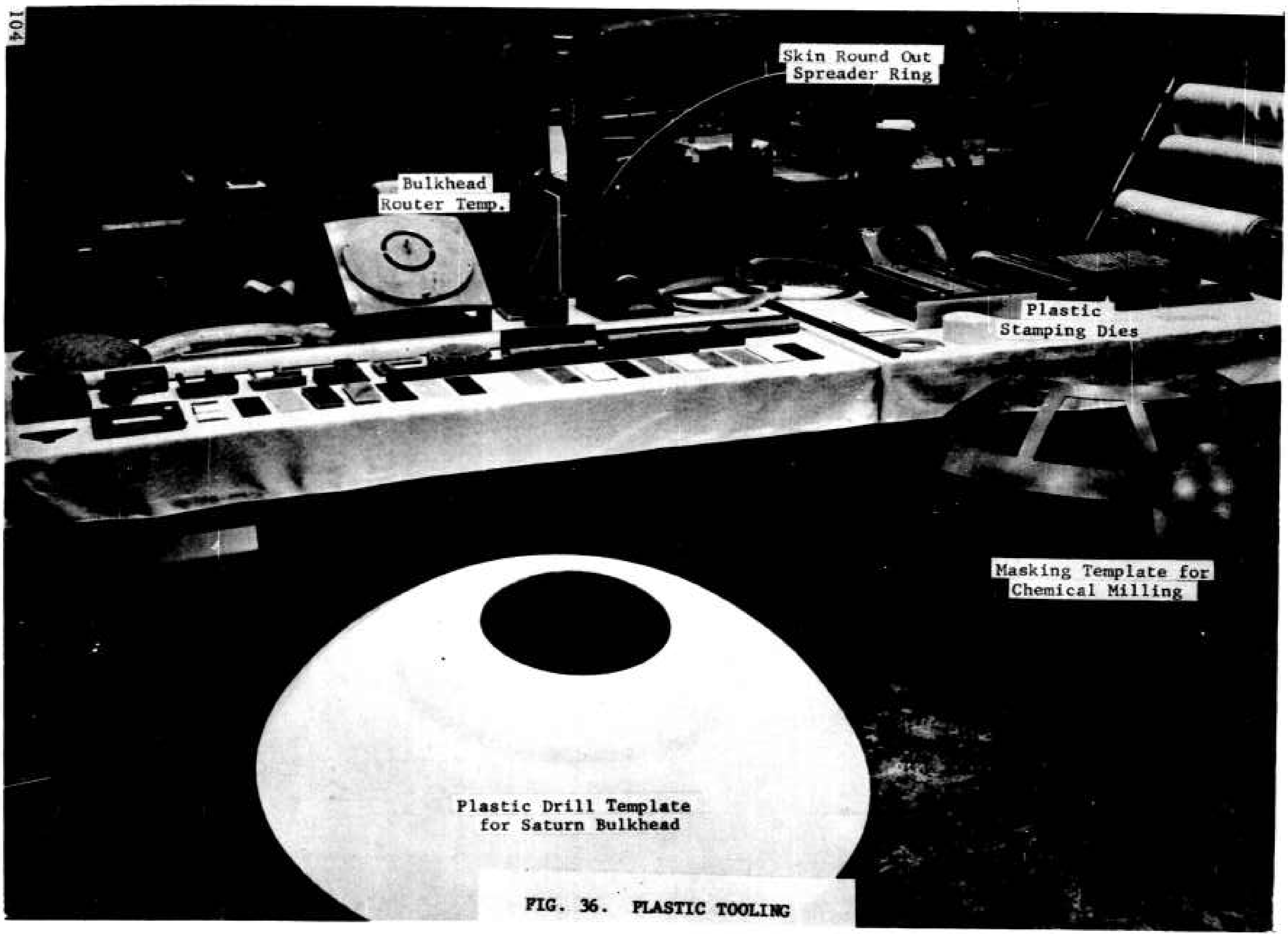


FIG. 36. PLASTIC TOOLING

FIG. 37. HYDRAULIC STRETCH PROCESS

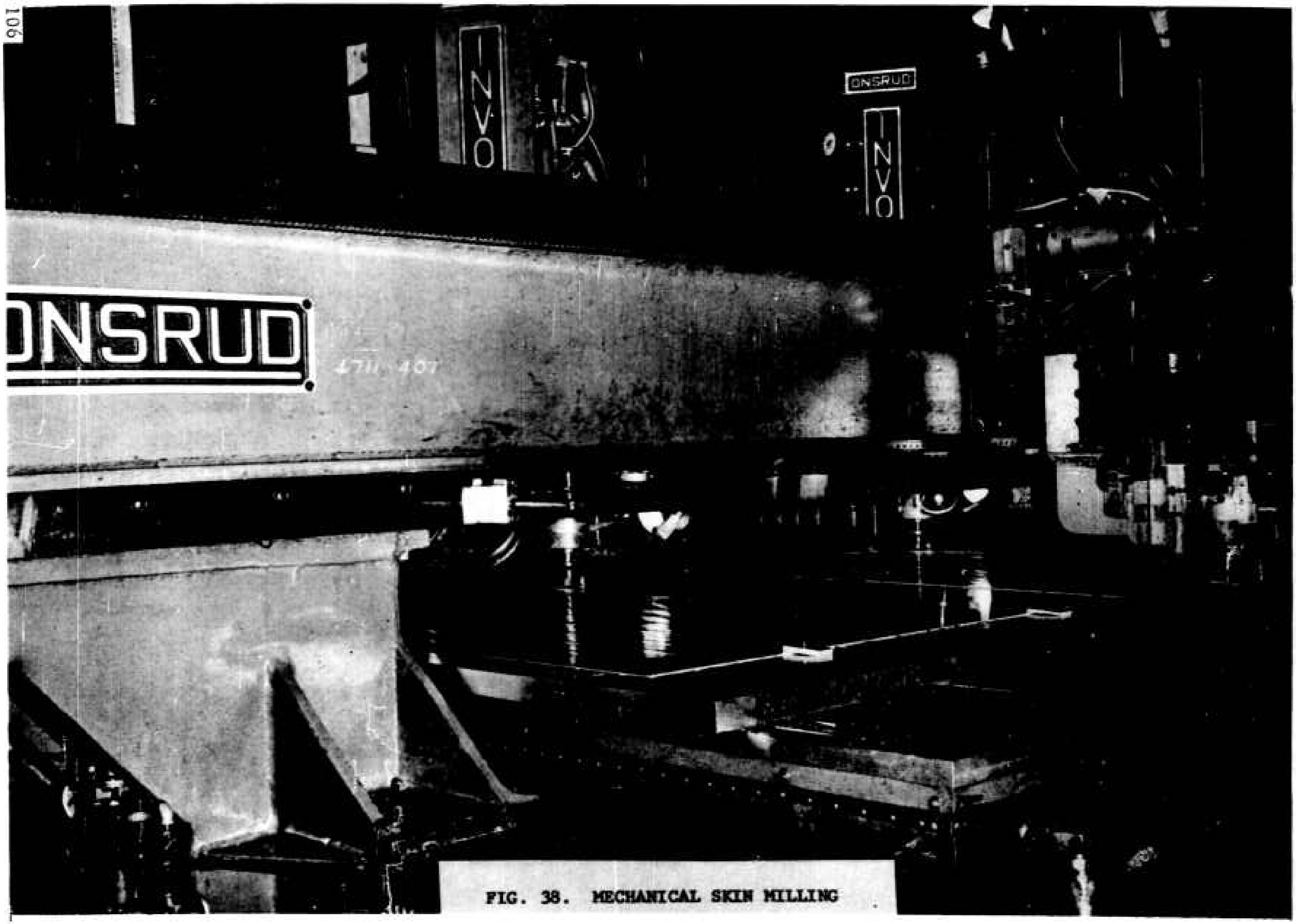


FIG. 38. MECHANICAL SKIN MILLING

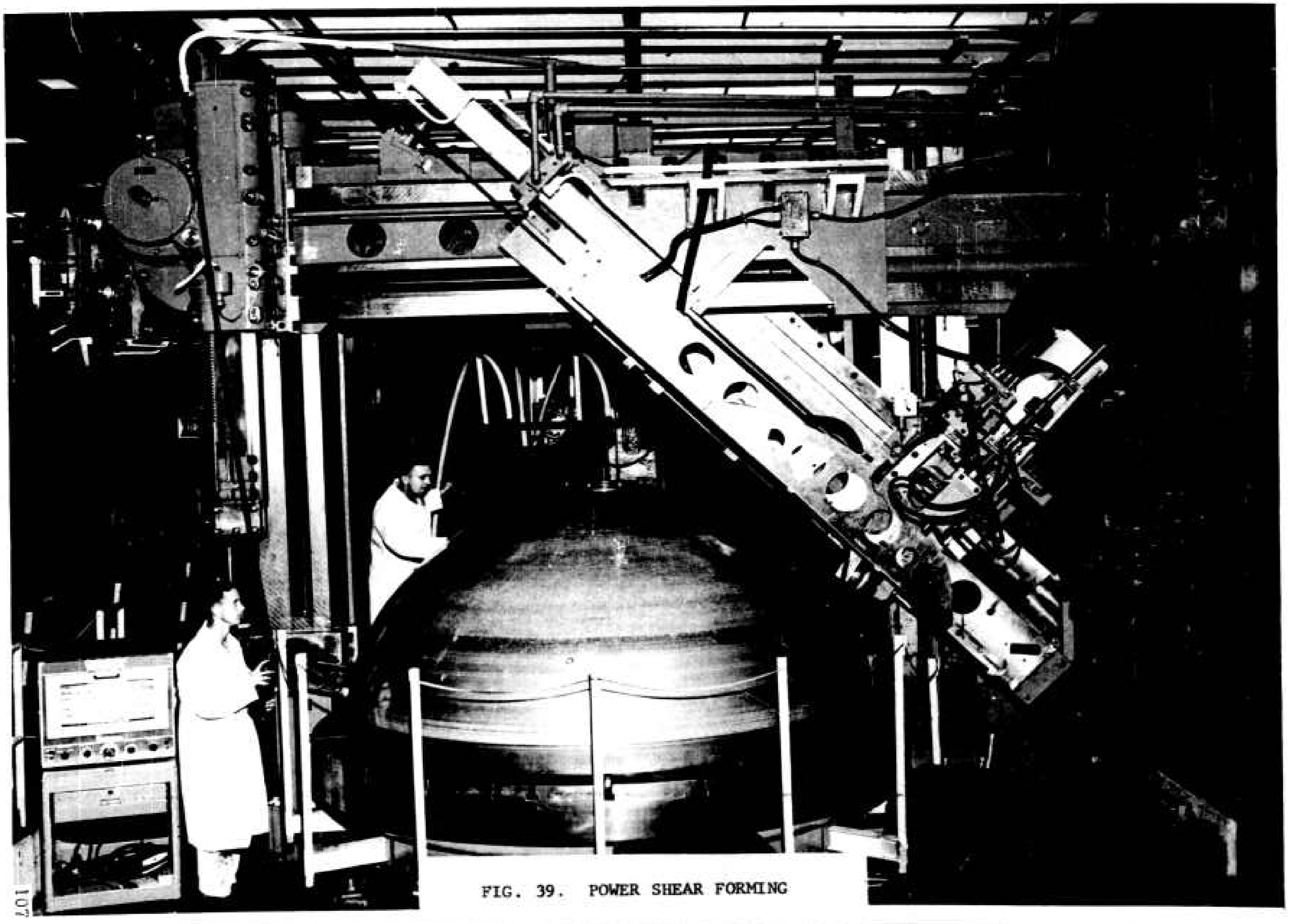


FIG. 39. POWER SHEAR FORMING

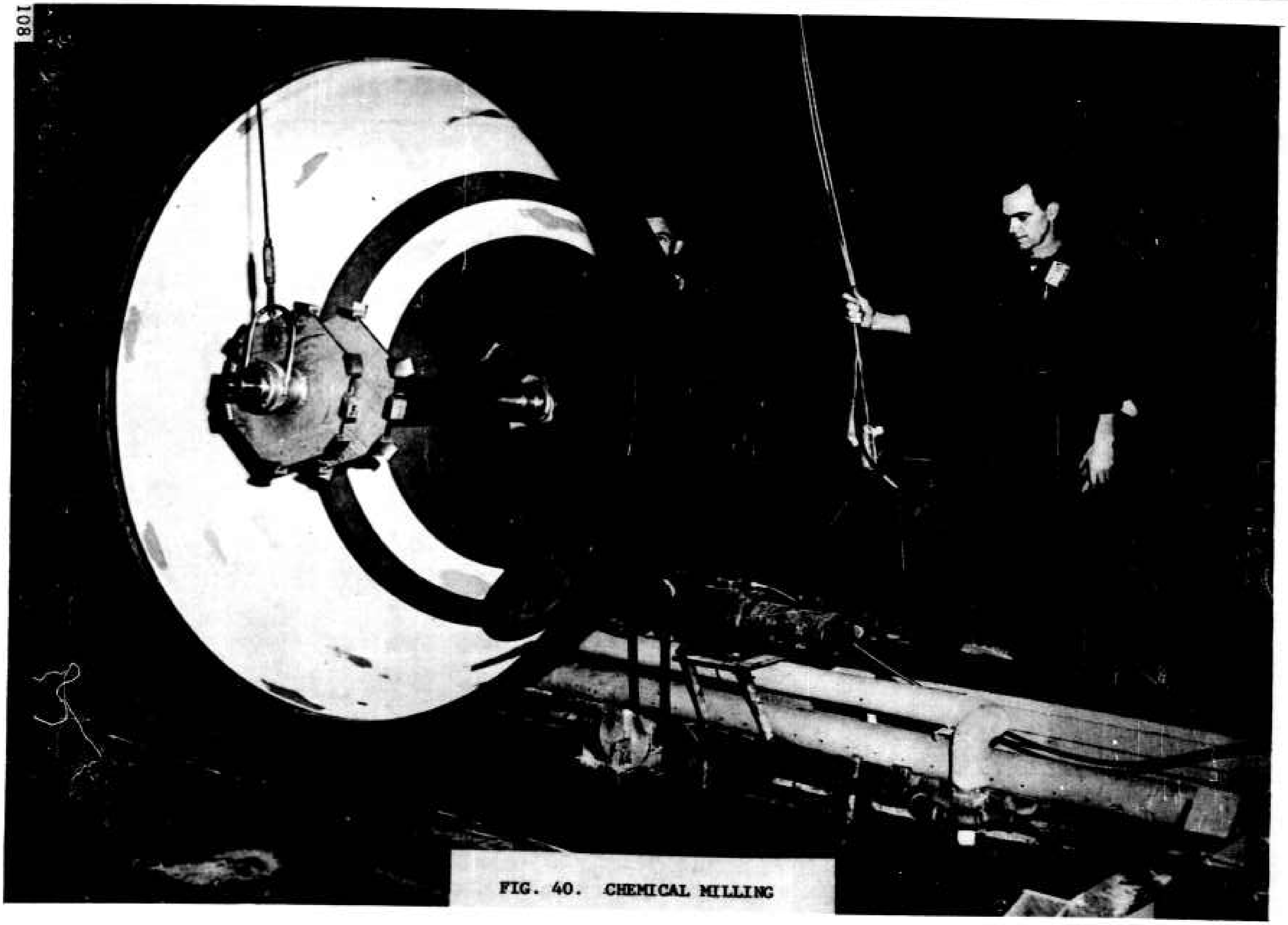


FIG. 40. CHEMICAL MILLING

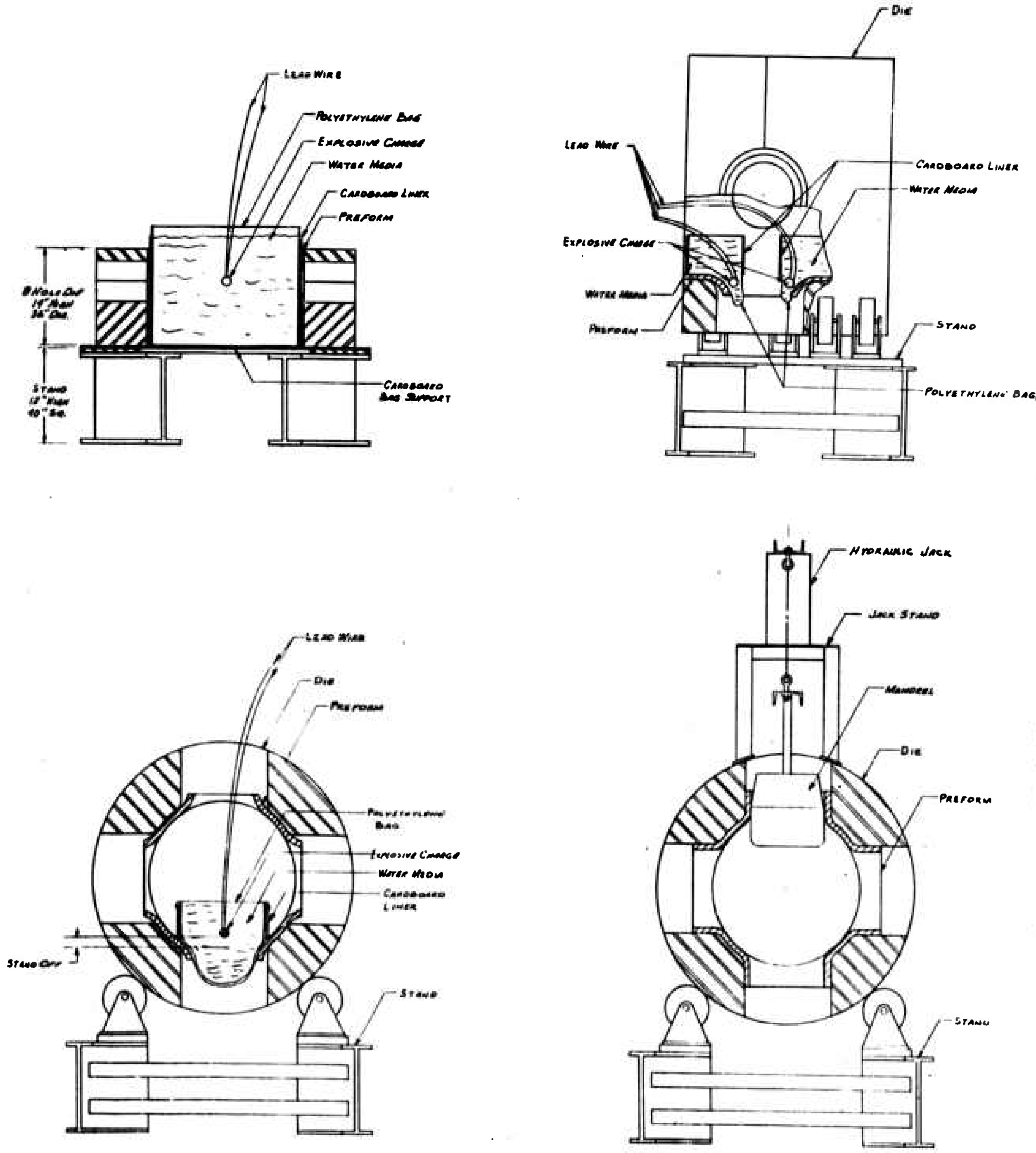


FIG. 41. EXPLOSIVE FORMING PROCESS

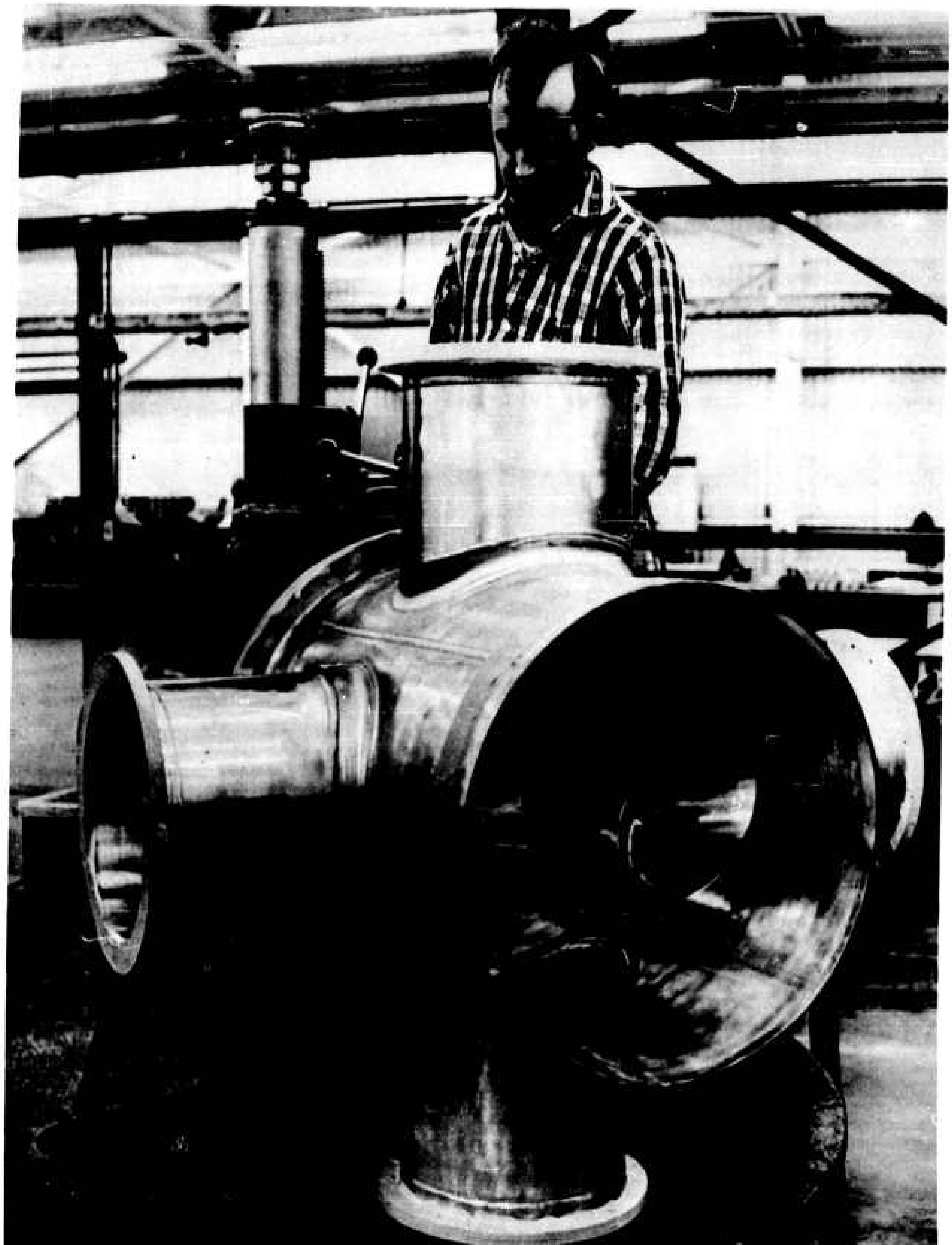


FIG. 42. EXPLOSIVELY FORMED LOX MANIFOLD

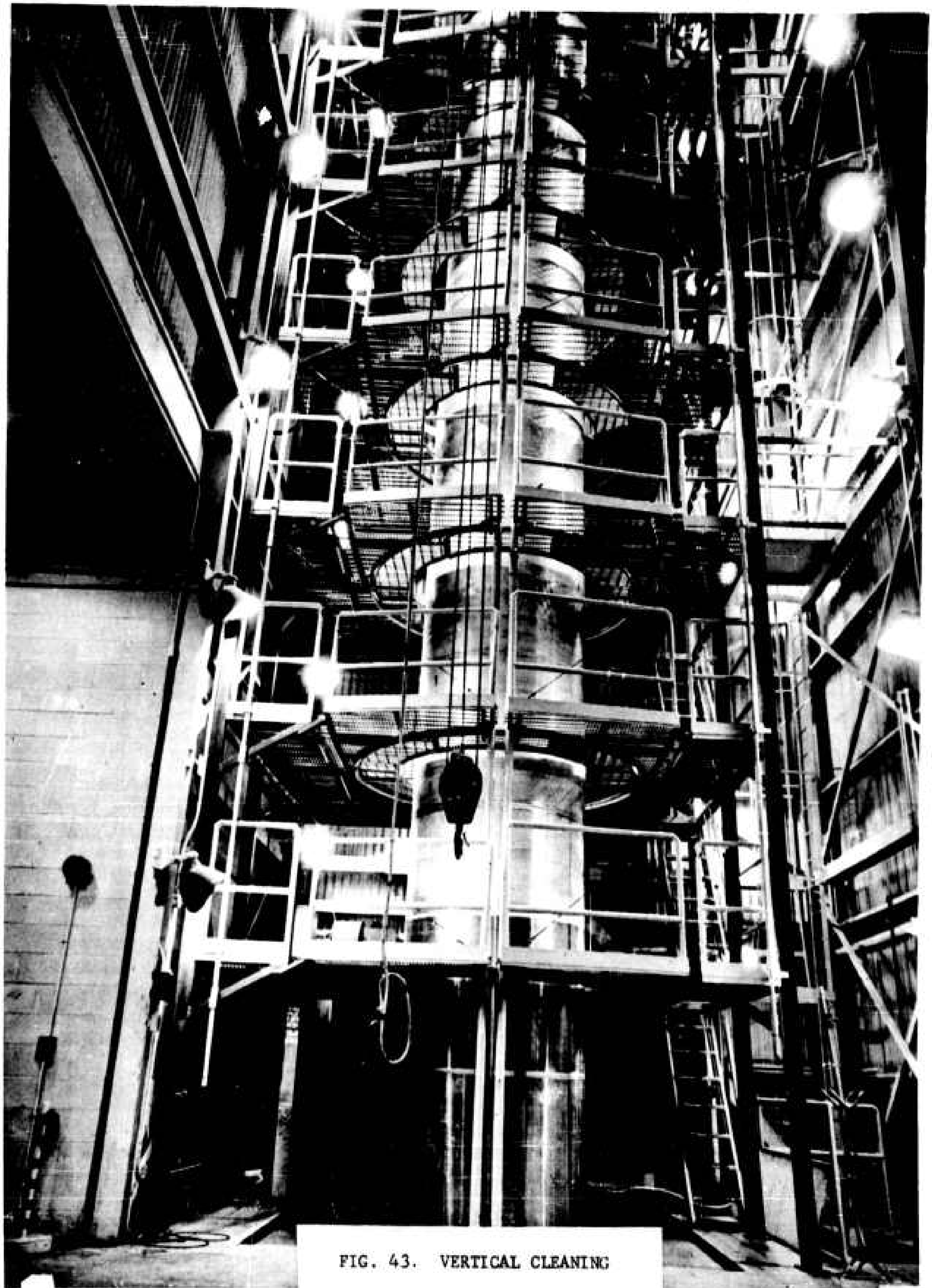


FIG. 43. VERTICAL CLEANING

operation is accomplished. The main feature of the new cleaning method is that the cleaning fluids are moved around inside the container rather than the whole container being moved as was previously done with the smaller REDSTONE and JUPITER containers. The chemicals are sprayed at a very high pressure through a rotating nozzle which is lowered up and down inside the container.

SATURN tooling buildup was completed without major difficulties. Illustrating the new and novel types of tooling required, Figure 44 shows the spider beam assembly position where final assembly and alignment of the 20-inch I beam structure is accomplished. The tooling ring, which is required to allow rotation of the booster during final assembly, is shown being attached to the spider beam. Two separate assembly fixtures, one for the inboard engine outrigger and one for the outboard engine outrigger (Fig. 45) had to be provided. They assure accurate alignment and positioning of the outriggers before final assembly to the tail barrel. The two types of outriggers are positioned and attached to the tail barrel on tail section assembly and alignment fixture (Fig. 46). The optical alignment made while the tail section is mounted in this fixture assures that all the 56 mounting and attachment surfaces of the tail are precisely located.

Figure 47 shows the booster assembly station as it stands ready to receive the major components for final assembly. Multiple level scaffolding surrounds both the front and rear areas of the station, permitting the performance of several operations simultaneously. The scaffolds support two lightweight traveling cranes at each end of the booster for handling smaller subassemblies. During assembly the booster rests on the two cradles, shown in the center of the picture, which support it at the 20-inch I beam tooling ring and the outrigger tooling ring. Rollers on each cradle permit rotation of the whole booster. The front and rear cradles are connected by two heavy truss structures. This arrangement is left intact for booster transportation. The cradles are supported by four leveling stands, one at each corner. These leveling stands provide for vertical and horizontal movement of the cradles, thus permitting leveling of the booster for alignment and positioning of instrumentation.

Provisions are being made for automatic self regulation of the missile position during assembly operations. This is rapidly becoming a requirement as conventional positioning and alignment operations with optical equipment are proving increasingly burdensome and time consuming with the growing size of boosters. This operation can be speeded up and cleared of human error with the development of electro-optical and electronic positioning and alignment devices. The development of remote and automatically positioned large scale assembly fixtures will also allow for economic, lightweight design of fixtures and will eliminate the need for heavy independent structural foundations. Presently being accomplished at ABMA is testing of gravity referenced resolvers which could be used in automatic electronic alignment of the SATURN during assembly (Fig. 48). A specially developed alignment instrumentation panel is shown in the background.

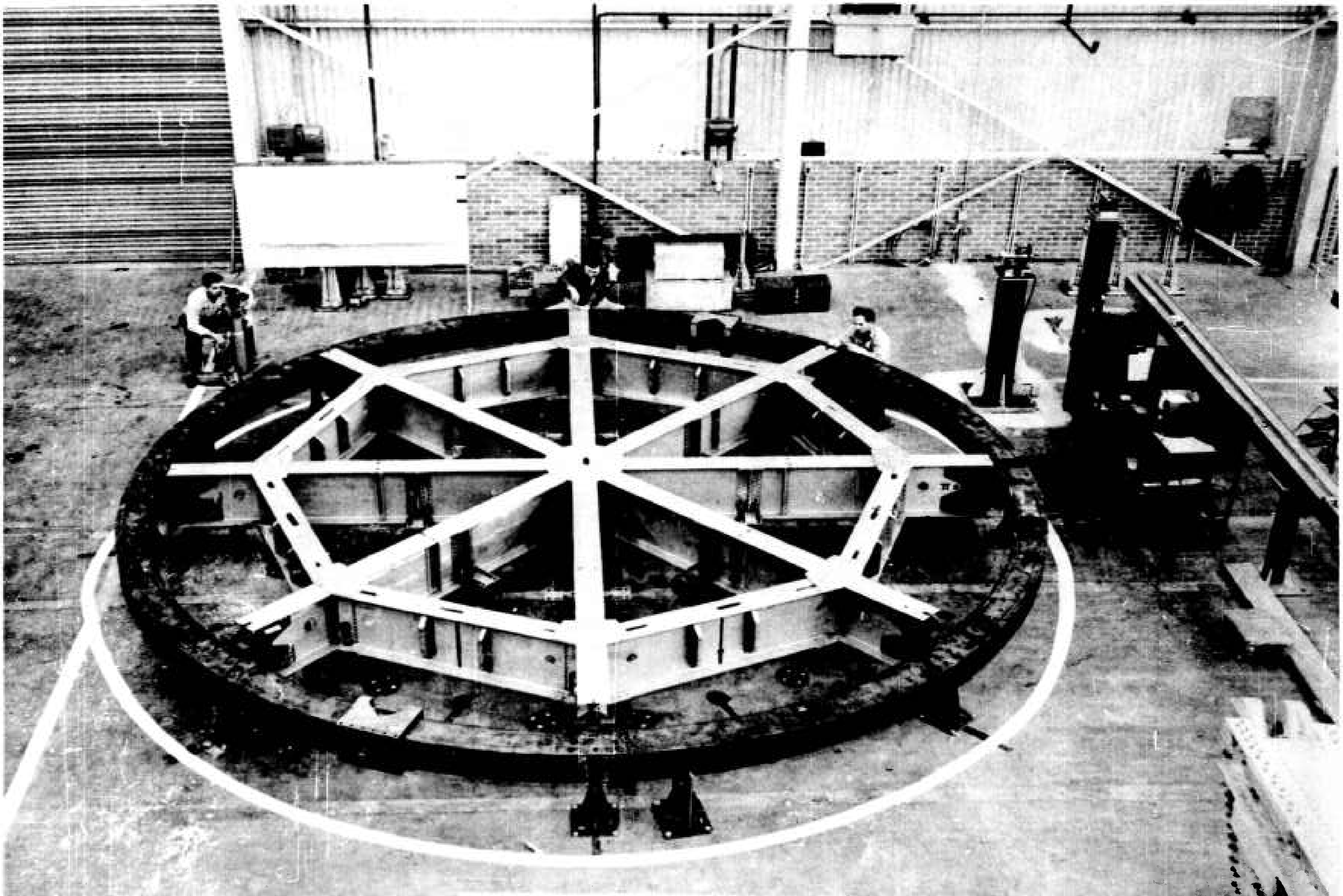


FIG. 44. SPIDER STRUCTURE

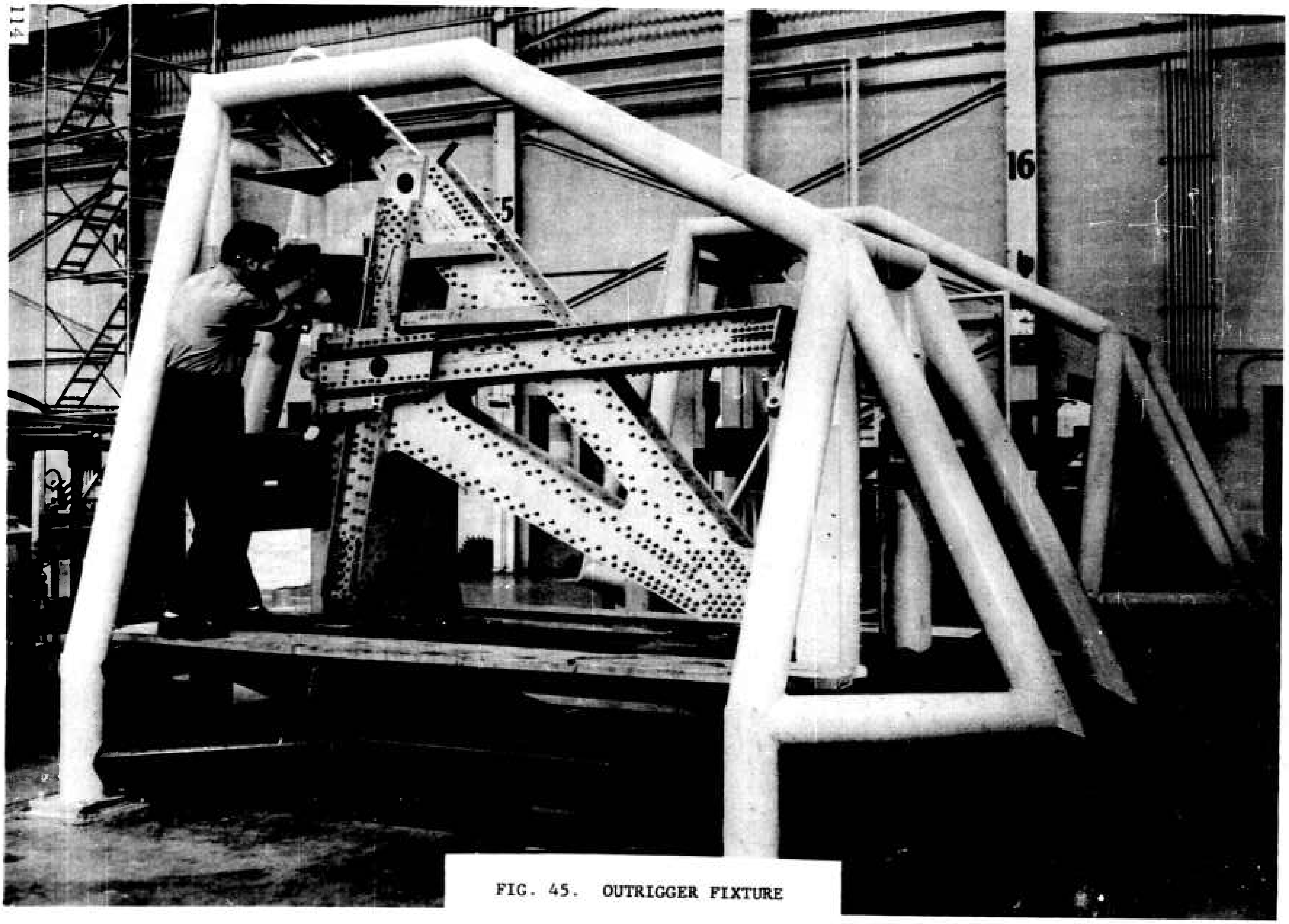


FIG. 45. OUTRIGGER FIXTURE

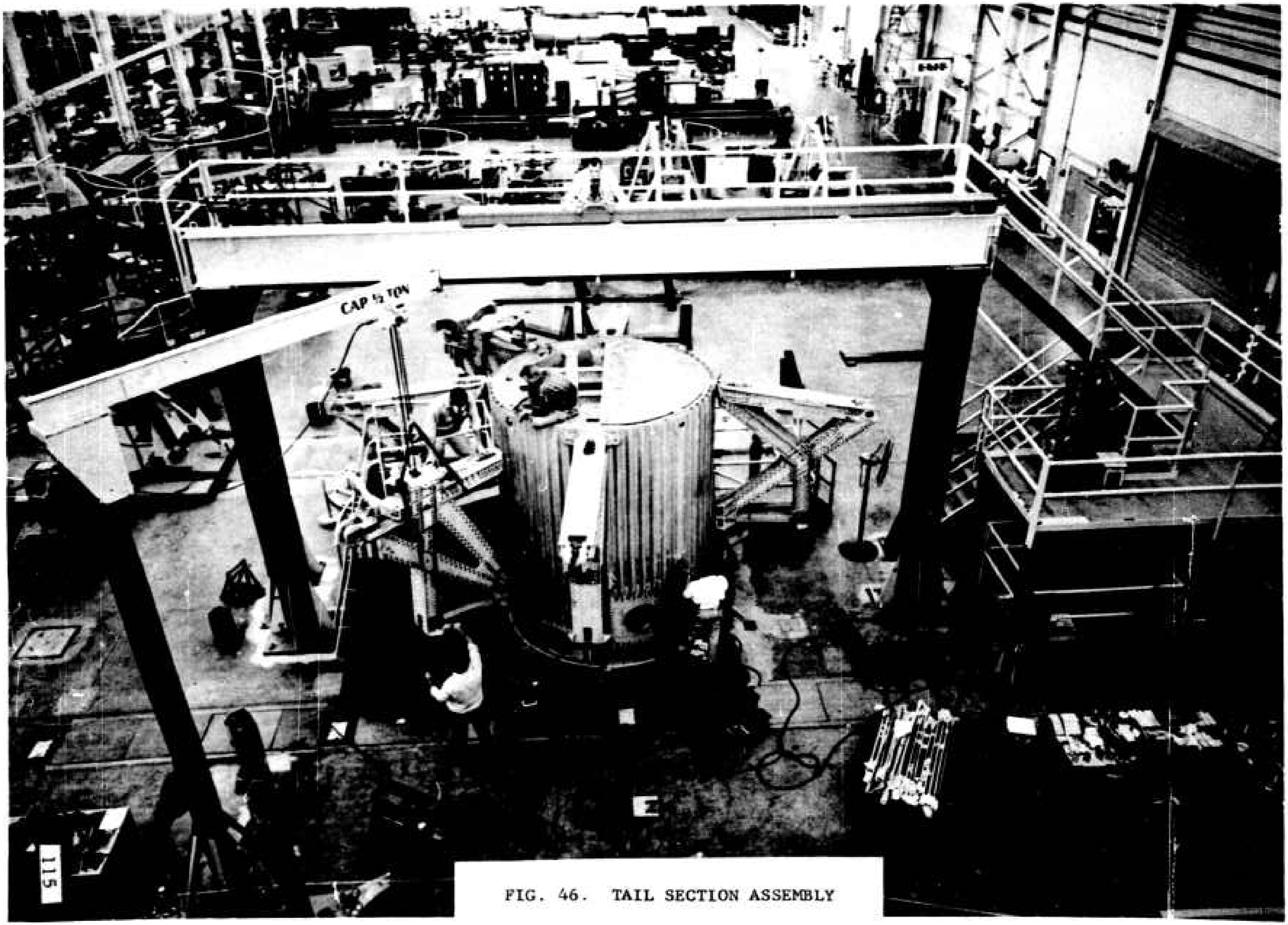


FIG. 46. TAIL SECTION ASSEMBLY

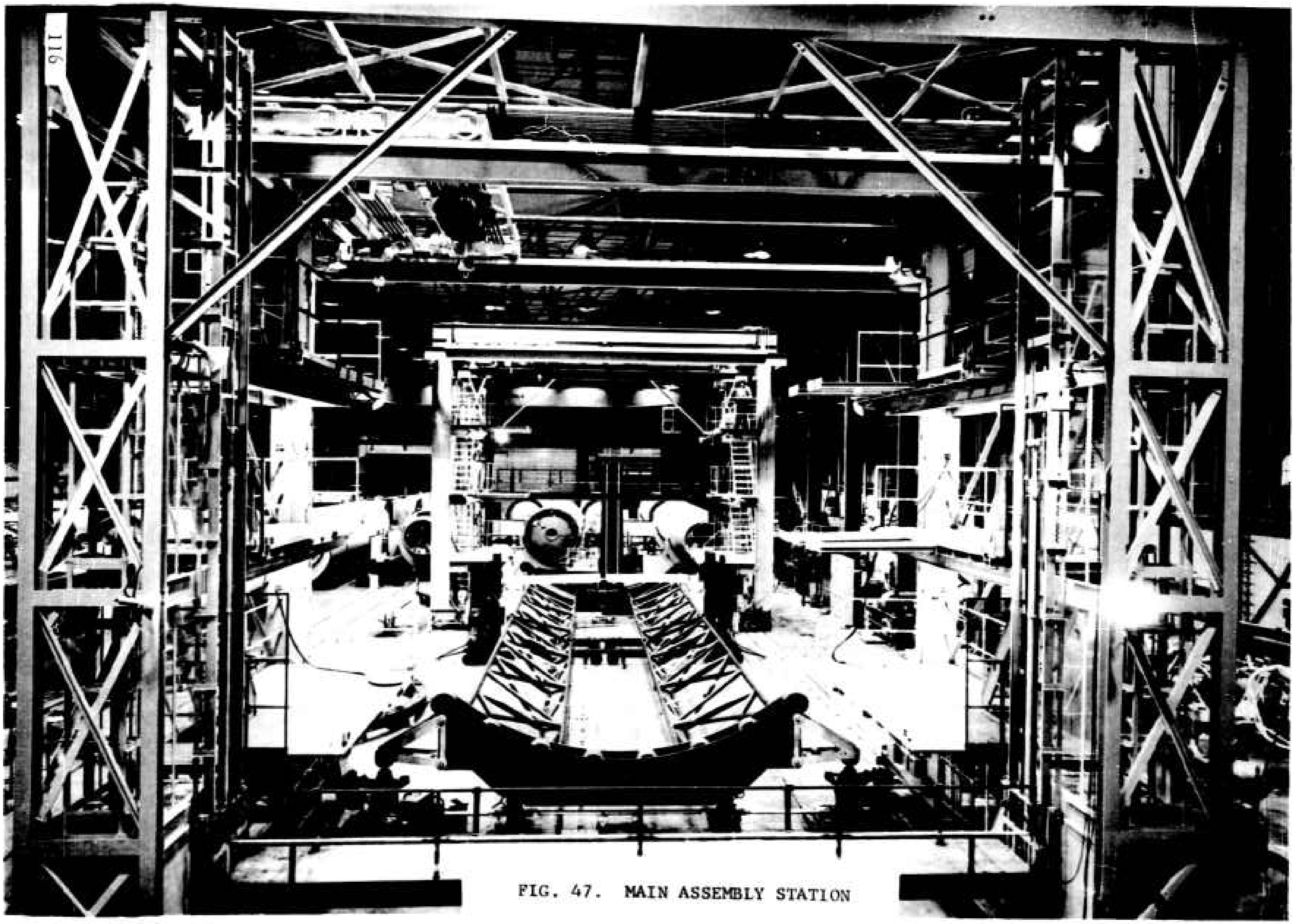


FIG. 47. MAIN ASSEMBLY STATION

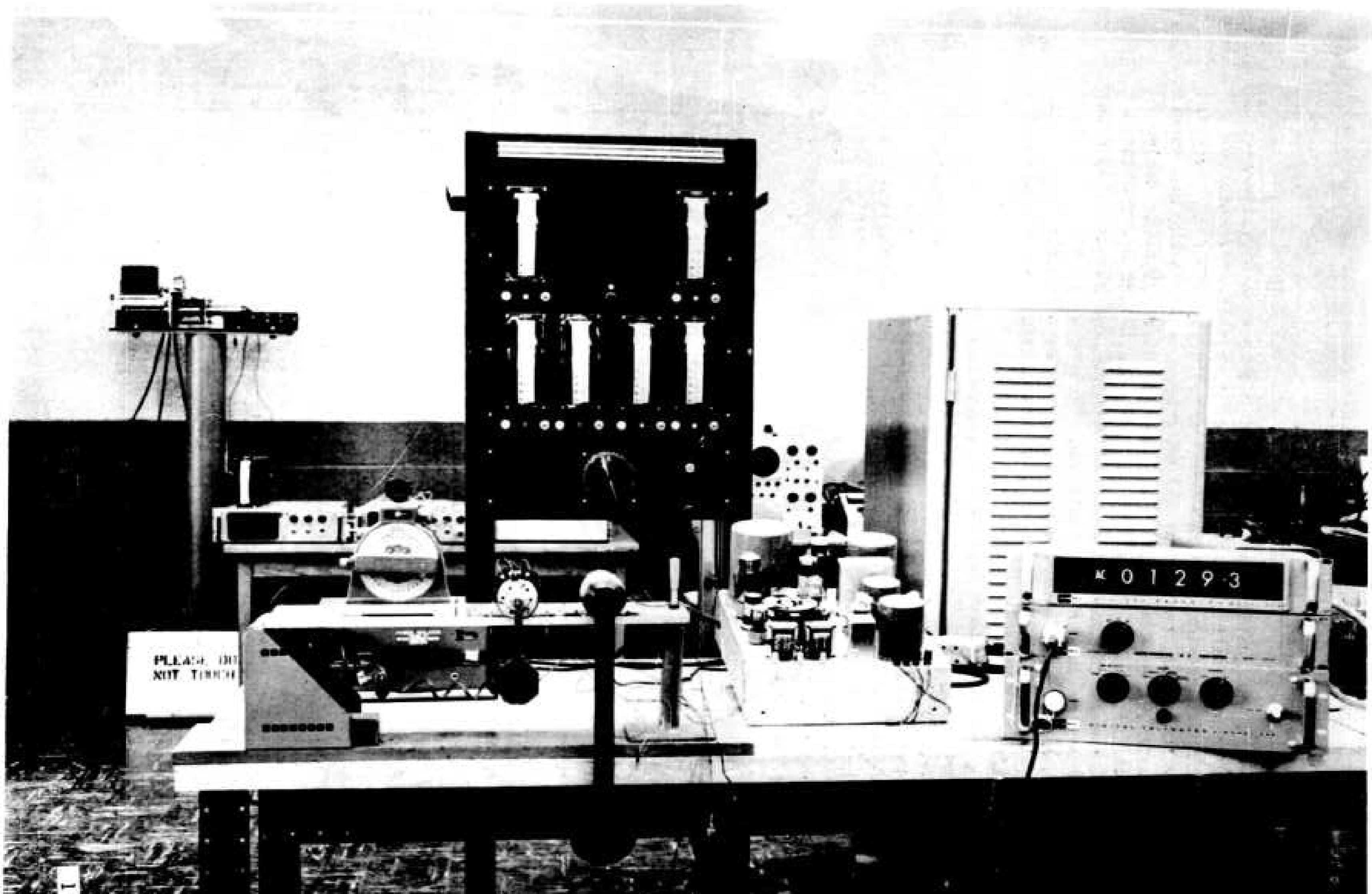


FIG. 48. ALIGNMENT PENDULUM TESTING

6. (U) System Quality Engineering

a. Reliability

A quality engineering program has been implemented for the SATURN program with general objectives of missile reliability similar to those previously established for the JUPITER. The purpose of the program is to assure that all materials, components, and systems are fabricated, inspected, and tested in accordance with the design intent, qualified parts lists, and approved inspection and testing procedures and specifications. In addition to the thorough inspection procedures set up at vendors plants and in-house fabrication assembly and test stations, a drawing review board has been established. This board reviews drawings prior to purchase or fabrication of equipment and establishes inspection procedures to insure that components can be adequately inspected upon delivery.

In addition to the standard testing of components for flight qualification, a separate program of component reliability tests has been established. The initial phase of the SATURN reliability test program will cover the testing of critical booster components. During the past six months' period primary effort has been devoted to the establishment of the reliability program scope, test procedures, and selection and procurement of hardware.

Initial tests on the selected components will determine if they perform satisfactorily within their required limits under operational environments. If possible, further tests-to-failure will be conducted to ascertain the margin of safety. In follow-on programs additional components, built into systems, will be used for reliability testing of system compatibility. During the component reliability test program, life-cycle testing will be performed to simulate re-use of a component from a recovered booster.

A round robin type of test sequence will allow a number of simulated flights to be completed on a group of component samples within a year. Confidence levels and fatigue data should be learned concurrently.

Test hardware selected with FY-60 funds includes: lox and fuel valves for fueling, draining, pressuring and venting; control components, such as the control computer and actuators; the power, control propulsion, and measuring distributors; engine components and selected launch accessories that are critical for operation of the launcher hold down and support arms.

Propulsion components for reliability testing were selected from the engine system on the basis of H-1 engine system tests that have been made at both Rocketdyne and ABMA. These tests have pointed out the weaker points in the H-1 engine system and a detailed study on mode of failure will be made. Component analysis will be conducted, as much as is practicable, as though it were a subsystem or system.

Some missile systems will logically require a component level test program prior to a subsystem level test. The guidance and control system is one of these. A series of subsystem tests conducted in this area would be of little value inasmuch as no specific knowledge could be gained from the failure of a black box if all the components could not be monitored. This cannot be done and still accurately duplicate missile configuration and use.

The major emphasis in ground support equipment for the SATURN will be on those critical items that must be extremely reliable once the launch command has been initiated, e.g., the release mechanisms for the retractable arms and the hold down arms.

b. Inspection and Analysis

The inspection portion of the SATURN quality engineering program is a continuing operation. Initial inspection of components is at the vendors plant or at the ABMA section that produces the component. This initial inspection is followed by an in-house receiving inspection of a vendors component. In the case of some very large in-house fabricated components or units, such as the tail section assembly, or very intricate units, such as the ST-90 platform, the inspection is carried on with the inspectors working side by side with the fabricator.

One of the inspection operations is the calibration of the propellant containers. Very accurate calibration of the propellant containers is required as a differential pressure head method (See Section II.C.2.c.(3)) is used to determine within 1/4% the total mass of propellants in the containers. One step of this operation is seen in Figure 49.

During the entire process of the SATURN booster assembly, inspection teams conduct checks on the correctness of the assembled components, including pressure testing of the containers and lines. After the booster is completely assembled, it undergoes a complete electro-mechanical component, subsystem, and system checkout during a 12-week period which is climaxed by a simulated flight test of the booster.

FIG. 49. LOX CENTER TANK CALIBRATION

D. Upper Stages

1. (U) General

Instructions from ARPA on 29 and 31 July 1959 reduced contractor effort on the TITAN type second stage to an almost negligible minimum. ABMA continued in-house studies on the "B" and "C" series of vehicle configurations, for which study results were forwarded to ARPA and NASA. Recommendations for the optimum upper stage configurations and an over-all SATURN vehicle development plan were submitted by the SATURN Vehicle Team (Silverstein Committee) late in the report period and accepted by the Administrator of NASA. The upper stage program is presently being reoriented accordingly.

2. (C) Second Stage

a. (U) General

Agreement was reached between ABMA and AFBMD/BMC, under the guidelines of Amendment No. 7 to ARPA Order 14-59, on the contractual relationships to be exercised between ABMA and the Martin and Aerojet Firms. Contracts were subsequently established with these companies. The level of effort was reduced shortly afterwards, however according to instructions received from ARPA, as only those studies independent of the upper stage configuration were permitted. The effort on both contracts was further reduced to one man-month per month on 31 December 1959 as a result of the recommendations, established by the Silverstein Committee study, to continue SATURN vehicle development with the C-1 configuration.

b. (C) Structures and Propulsion

Design effort was not initiated on the second stage tankage because of the changing diameter considerations. An adapter ring design was studied which could accommodate one, two, or three pairs of engines, depending on the ring diameter, and space available. Also, a modular engine packaging concept was developed which allowed for such an arrangement. The engines used in the study were designated XLR-87-AJ-3, with a thrust rating of 200,000 lbs at 16 to 1 expansion ratio.

3. (U) Third Stage

Effort in the third stage studies, as for the second stage, was reduced according to instructions from ARPA, and only technical discussions on the basic principles were continued. The newly conceived C-1 configuration continues utilization of a CENTAUR for the third stage.

SECTION III. (U) SUPPORT EQUIPMENT

A. Introduction

The progress of the ground support equipment program for the SATURN is on schedule; it was not affected by the changes considered for the upper stages during the report period.

One of the primary areas of progress has been in transportation. As a result of a study, an enclosed sea-going barge was selected over a river barge as the form of transportation from ABMA to Cape Canaveral. The necessary land transportation will be provided by a trailer utilizing part of the assembly equipment and an aircraft tug. The method of loading the barge was changed from using a stiff-leg crane to a roll-on, roll-off concept. In other areas, the simulated booster was used in transportation, handling equipment, and erection tests. Investigations have been made in the effect of lox on the tanks when it is being boarded. The design of the umbilical tower has progressed and a prototype short cable mast fabricated. The launcher and the launcher arms have been designed.

A detailed description of the ground handling equipment and a general discussion of the progress made during the report period are given on the following pages.

B. Checkout Equipment

The nonfacility type portable checkout equipment for the SATURN booster consists primarily of two units of leak and functional checkout equipment. The first unit, complex in nature, consists basically of cart-mounted pneumatic and electric checkout circuits plus the necessary interconnecting pneumatic hoses, electrical cables, and test fixtures. This unit (Fig. 50) is used by systems analysis personnel to perform a complete electro-mechanical test and simulated flight during the twelve week period after the booster has been completely assembled and before static firing at the ABMA Static Test Tower. The other unit is a simplified version and functions as part of the firing equipment at the launch site. It consists of several pneumatic high-pressure distribution systems needed for pre-launch and launch functions. These systems are referred to as valve box and control panels. They are mounted on the launch pad and in the umbilical tower rooms. In addition to the firing equipment, an engine servicing trailer, used only at the launch site, is used for last minute purge and flush operations. The checkout equipment used at the launch site can be simplified due to the reduced on-site check and the availability of the entire vehicle pneumatic and electrical network as a test media.

C. Transportation Concept and Equipment

The large size of the SATURN booster precludes all forms of long distance transportation except water.

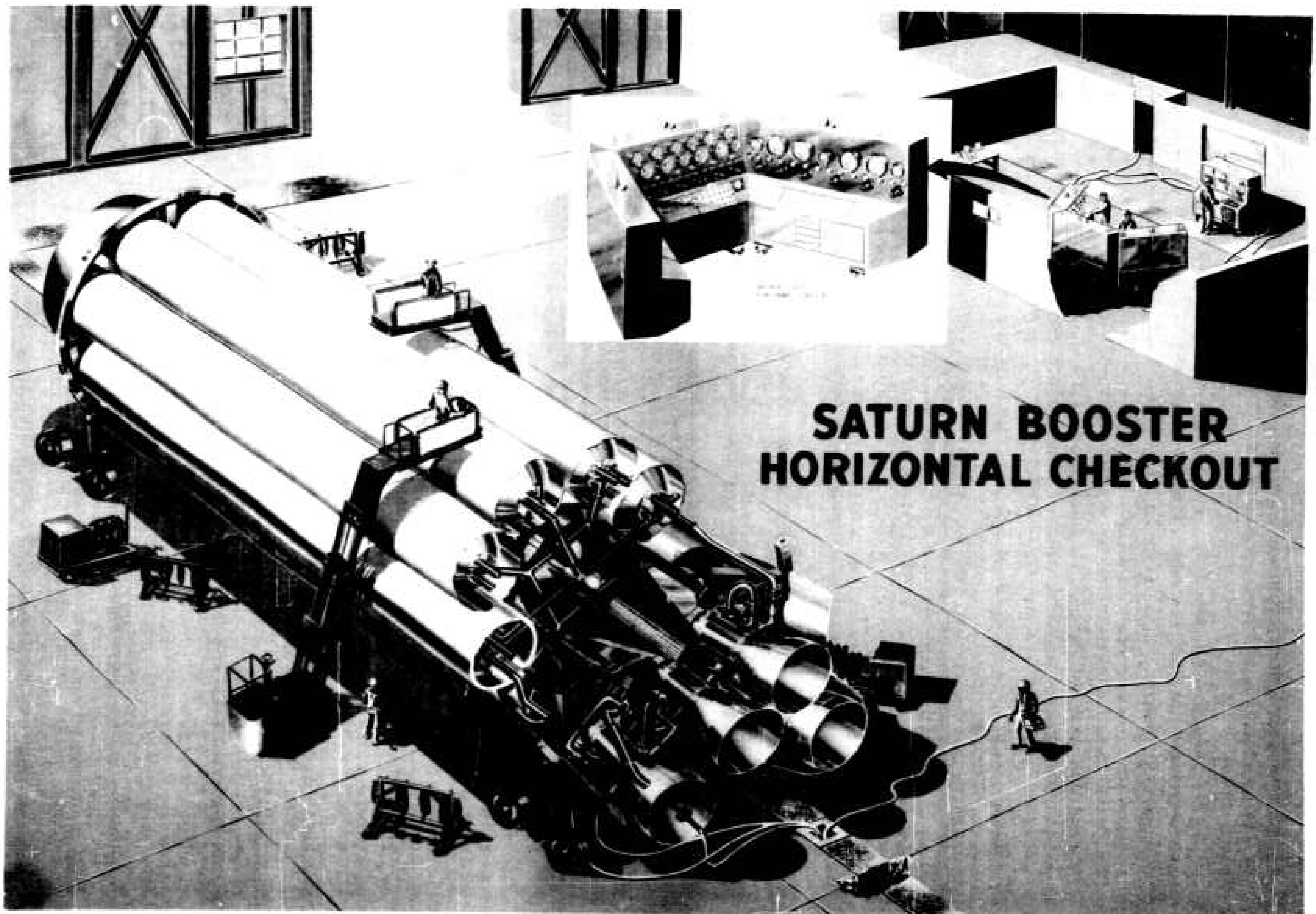


FIG. 50

A trailer-type transporter will be used for the limited land travel at ABMA and CCMTA.

The transporter assembly (Fig. 51) a unique piece of equipment, is formed by placing the assembled booster and its support cradles, connecting trusses, and assembly rings as a unit on two axle and wheel assemblies. The support cradles are secured to the axle assembly. The axle and wheel assembles incorporates two pairs of two independently braked and steered aircraft tandem wheels on an axle assembly. A towbar on the front axle assembly provides a connecting link to the prime mover, a standard aircraft tug (TT-11). The booster is carried on this vehicle from the Fabrication Laboratory, through testing and checkout, to the launch site.

After completion of all work at ABMA, the booster and transporter will be loaded, at the ABMA Tennessee River docks, on a specially designed flat-deck sea-going barge. The loading equipment at the dock (Fig. 52) is comprised of a ramp to the waterline and two electric winches mounted at the top of the ramp. These winches control the movement of the transporter during loading. The capability exists for expanding the loading equipment capabilities by constructing a 100-ton derrick. This will provide an additional unloading capability, which will be required when the recovery program becomes more extensive.

The loading equipment at CCMTA will be similar to that described above. It will be located at Site C.

The sea-going barge was selected as the water vehicle because of its simplicity and lower cost of operation. The barge compartment to house the booster will have end-opening doors and removable roof hatches.

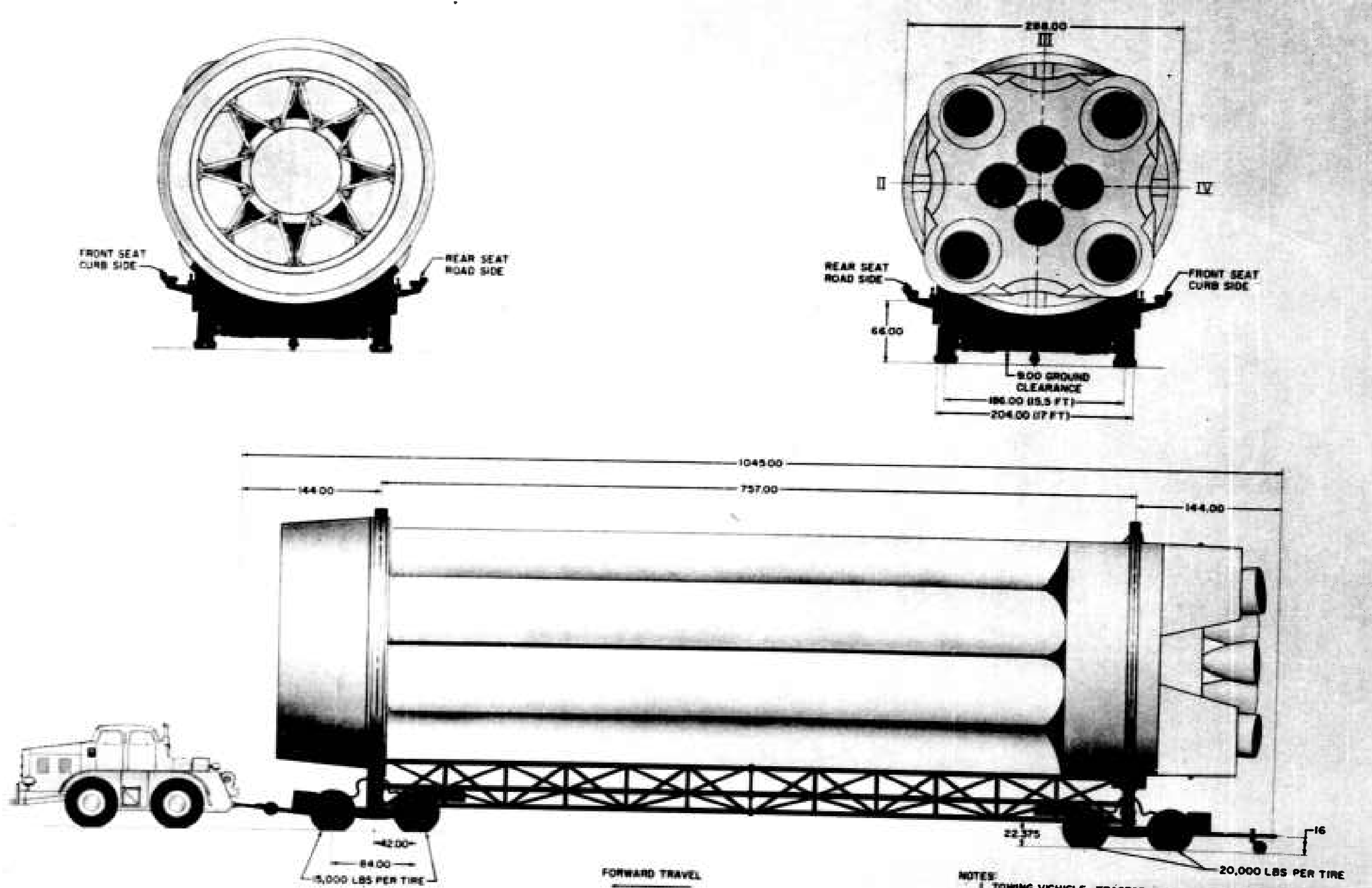
The barge will be towed from ABMA, via the Tennessee, Ohio, and Mississippi Rivers to New Orleans by a river tug (Fig. 53). From New Orleans, a sea-going tug, following the coast, will move the barge to Fort Pierce. Here, a river tug will assume the tow and move the barge to CCMTA, where it will be unloaded and taken to the launch site.

D. Booster Erection

The SATURN booster is erected on its launcher with a track-mounted gantry-type service structure and an erection equipment kit. The service structure has a bridge crane with two supporting hooks and is used to place the booster on the launcher. The erection kit (Fig. 54) supplies the necessary beams and cables for picking up the booster.

The first step of the erection procedure is to move the service structure over the launcher and to place the booster, on its transporter, parallel to the structure's base. The booster is then rotated from the transporting plane 45° to the erection plane and the top portion of the

SATURN TRANSPORTER AND TOWING VEHICLE



NOTES:
1. TOWING VICHICLE - TRACTOR, AIRCRAFT TOWING COLEMAN TUG GROSS WT. 33500 LBS.
2. TIRES - 1700 X 16 TYPE III 10 PLY NYLON
OUTSIDE DIA. 45"
WIDTH 17.34
ROLLING RADIUS @ 35% DEFLECTION 17.8"
CONTACT AREA @ RATED LOAD OF 31% DEFLECTION 275 IN²
@ 35% DEFLECTION 300 IN² MAX.
TIRE PRESSURE - REAR 80 ± 10 PSI FRONT 60 $^{+30}_{-10}$ PSI

FIG. 51

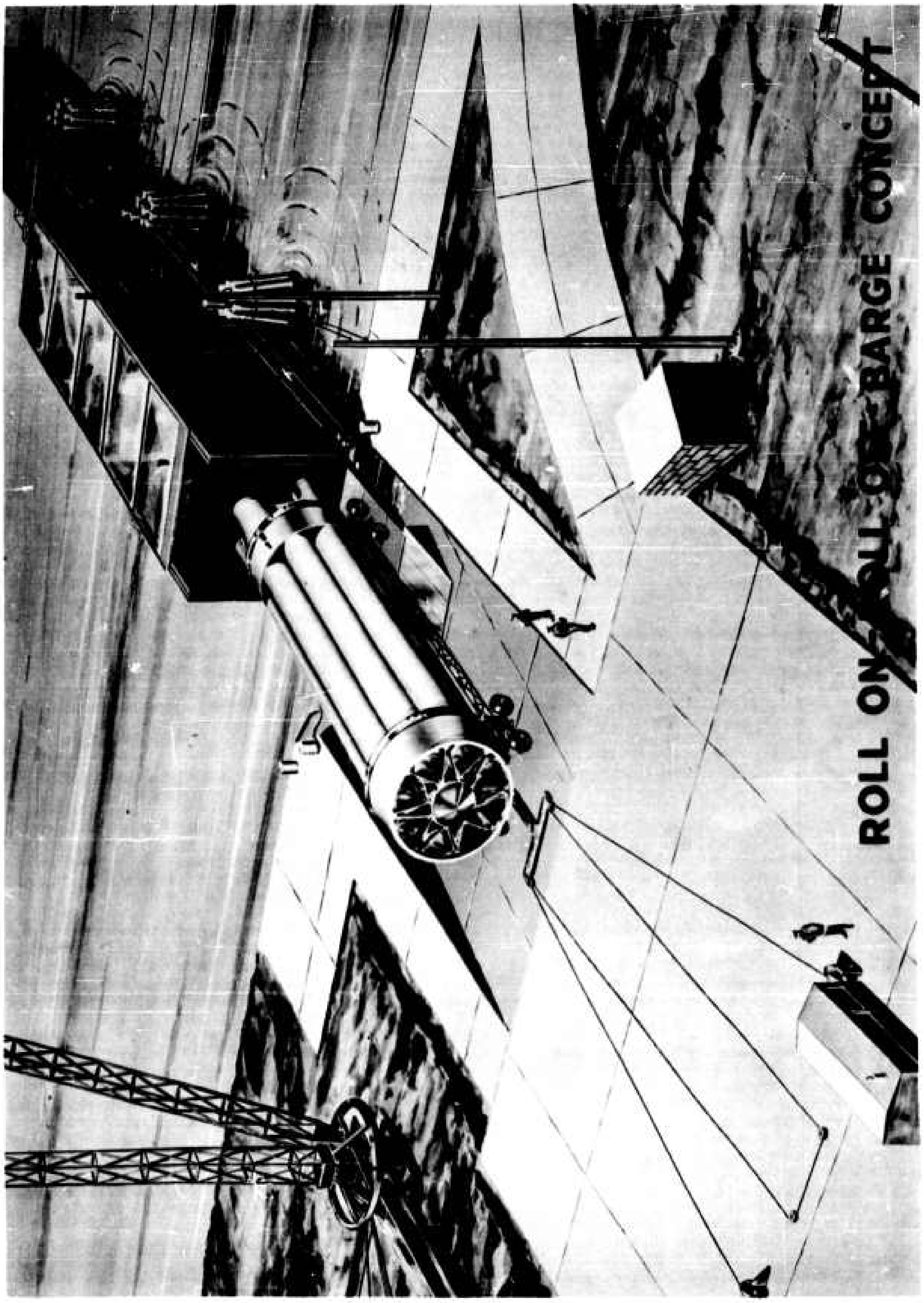
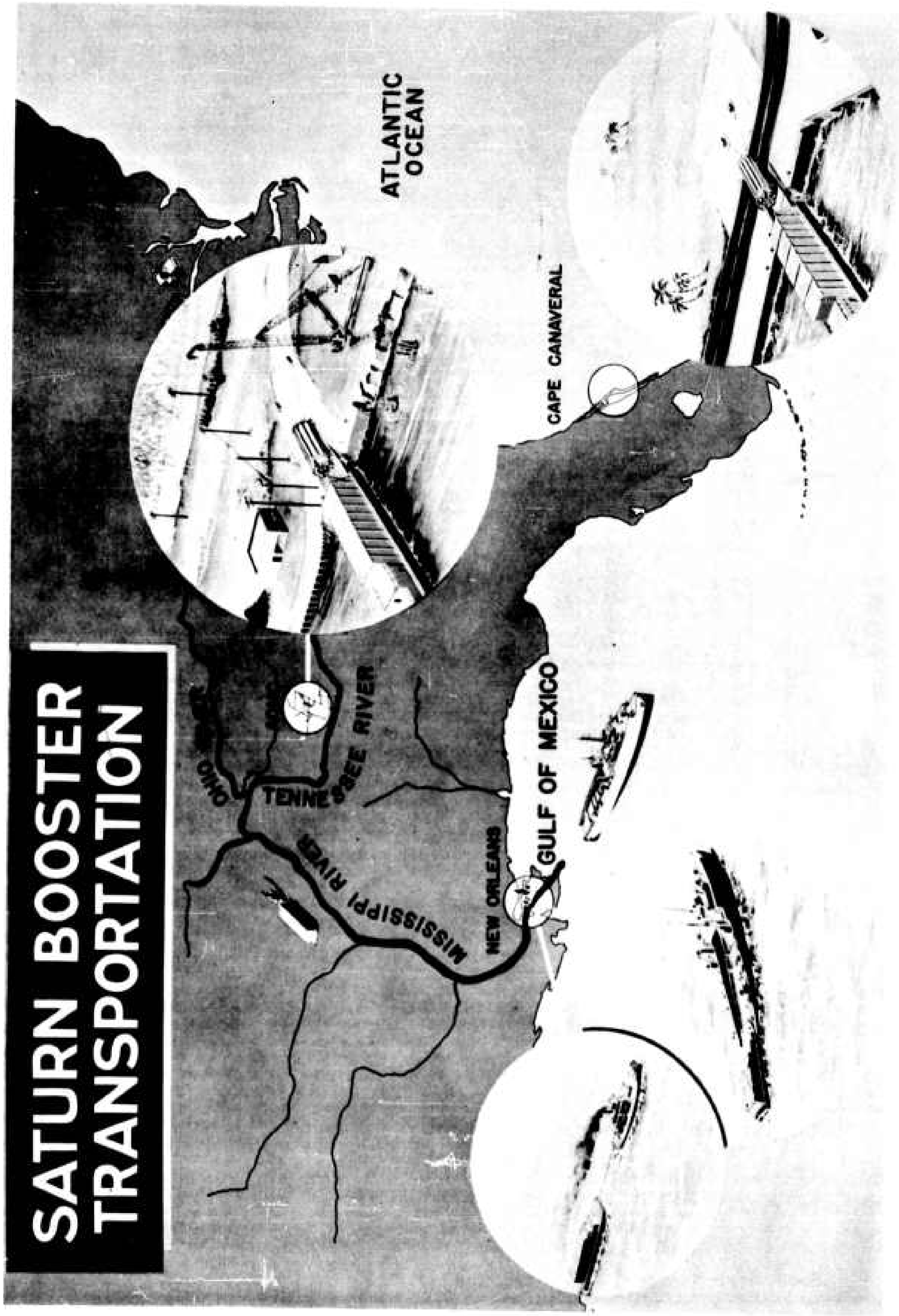


FIG. 52


SATURN BOOSTER
TRANSPORTATION
ATLANTIC
OCEAN
CAPE CANAVERAL
OHIO
TENNESSEE RIVER
MISSISSIPPI RIVER
NEW ORLEANS
GULF OF MEXICO

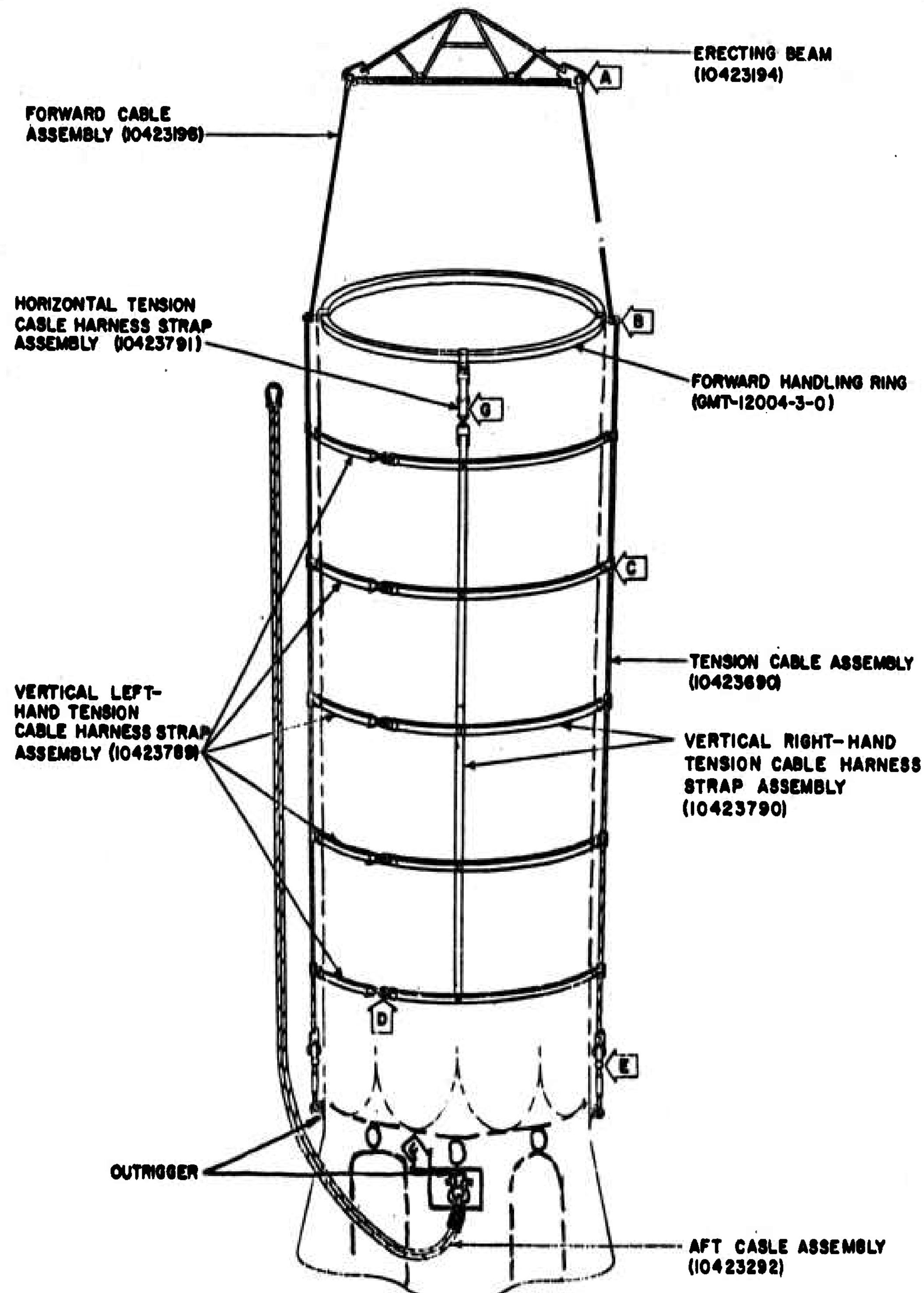


FIG. 54 ERECTING KIT DETAILS

rear assembly ring is removed from the booster. Next, to minimize eccentric erection loads, two tension cables from the erection kit are fastened between the front hoist points and the rear thrust frame. Several connecting bands across the top half of the booster minimize the catenary effects in these bands. The crane hooks are then connected to the booster pick-up points with erection slings and beams from the erection kit. The booster is lifted from the transporter and placed on four preleveled launcher support and hold-down points.

E. Launcher

The SATURN launcher (Fig. 55) is comprised of the launcher structure, a water fire emergency system, and a dry launcher deflector. The launcher structure is made of steel reinforced concrete and is 42 feet square and 27 feet in height. The launcher superstructure, 42 feet square with a centrally located hole 27 feet 4 inches in diameter, is supported at each corner by a concrete column 7 feet 4 inches square and 23 feet high. The personnel protection rail around the outside of the structure can be removed on the sides nearest the service structure. This facilitates the transfer of firing accessories to the launcher structure.

The launcher structure has eight booster support points, which are secured to the launcher according to a known firing azimuth, +10, prior to missile erection. If the firing azimuth is changed after erection, the missile will have to be taken down in order to physically relocate the eight arms to the near azimuth.

Four of the support arms, located at the outboard engine, are horizontally retractable for 62 inches. This is to provide clearance for the outboard shroud during liftoff. If any of these arms, fail to retract at liftoff, the others can be returned to their original position before engine cutoff. The four remaining arms located between the outboard engine are dual purpose; support and hold-down. Hold-down is accomplished by a preloaded toggle linkage that applies a hold-down force to the missile thrust frame. The preloading minimizes the transient loads, due to vibration, from being transmitted through the linkages, etc. Each arm has a height adjusting mechanism consisting of an actuating screw and wedge block to facilitate a ± 1 inch vertical adjustment.

The water firex emergency system is used only if the booster is not allowed to leave the launcher after the engines are fired. It provides a blanket of water, at the nozzle plane, to prevent the heat and flame of the engine exhaust from bouncing back up to the tail section. The system is a torus-shaped manifold fabricated in two halves that feeds flat spray and fog nozzles. It is located on top of the launcher around the hole. The ends of the manifold are blanked with spherical caps. Sufficient gaps exist between the ends of the halves to compensate for thermal expansion and Bourdon tube effects.

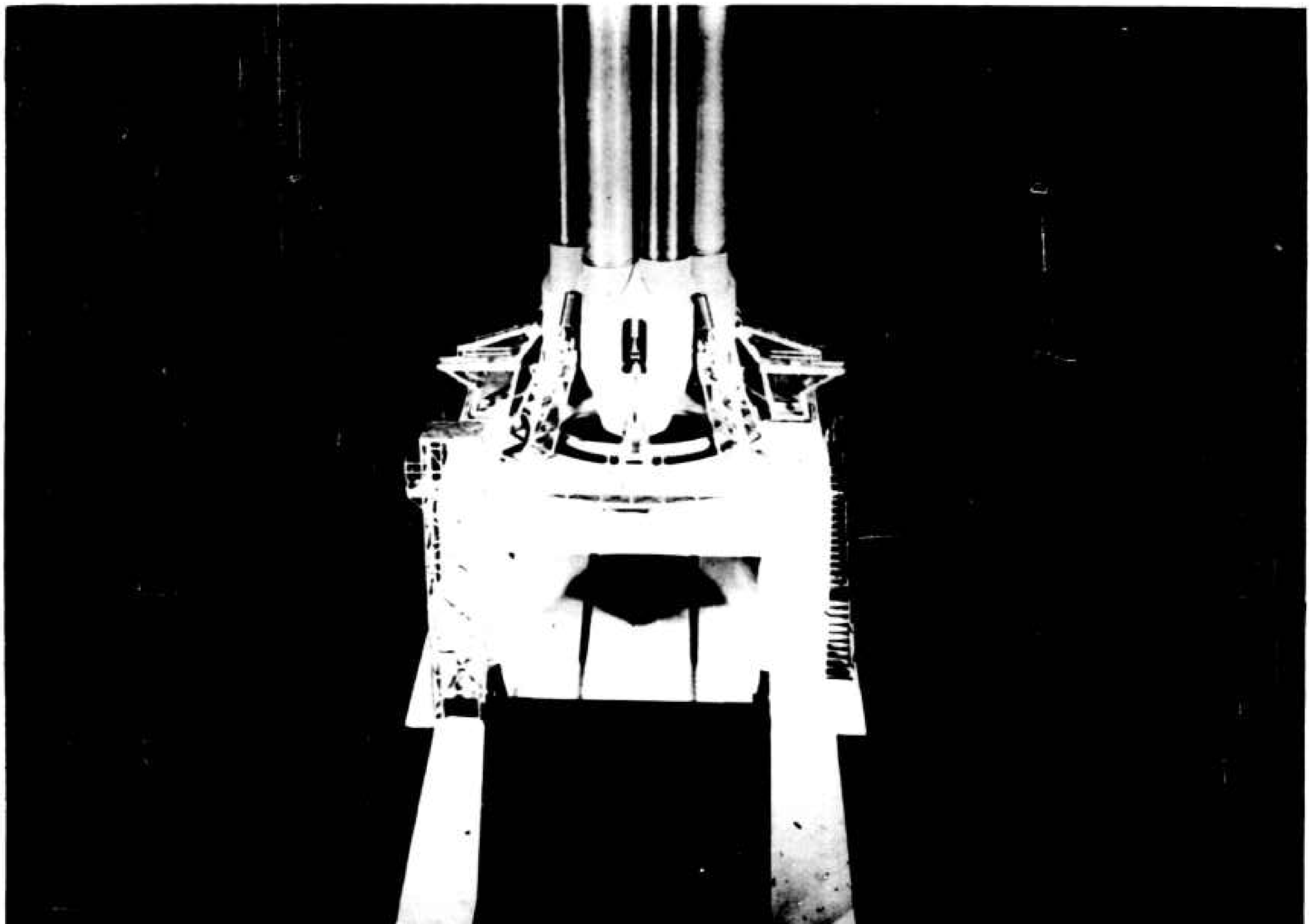


FIG. 55 SATURN LAUNCHER

The dry launcher deflection (Fig. 56) is constructed with a series of roof-type trusses at an 80° included angle. The trusses are covered with a removable one-inch thick steel skin. The deflector is carried on four retractable railing wheels. When the deflector is placed under the launcher the wheels are retracted and shear plates are lowered into provided slots. The plates secure the launcher and counter the lateral thrust components of the engine exhaust. Side shields, 41 inches high, are mounted on each side of the deflector to prevent flame spilling and backwash beyond the deflector side. The shield is hinged at the top of the deflector to facilitate the installation and removal of the deflector and the launcher.

The design of the launcher, a symmetrical two-directional configuration, was governed by the various possible orientations of the eight jets with respect to the flame deflector surface. Since the relation of the jet pattern to the deflector depends on the firing azimuth, the deflector's length and width were determined by an impingement angle of 40°, which prevents backfollow. The flame escape dierection is horizontal. The flame impingement area is strictly on the inclined surface and is well outside the radius between the inclined surface and the horizontal part; the total projected deflector area is 1080 ft^2; the projected impingement area, 90.5 ft^2.

F. Alignment on Launcher

A minimum error must exist between the stabilized platform vertical and the thrust vector when the SATURN is on the launcher. This is required because a 1-degree misalignment between the thrust vector and the platform vertical may allow the booster to strike the launcher. To minimize the error, the fixed support arms, movable one inch in elevation, are adjusted to match the difference between the thrust frame support points. The selective elevation of the support points is determined at the Fabrication Laboratory. The elevation of the support arms must be established within ± 0.032 inch.

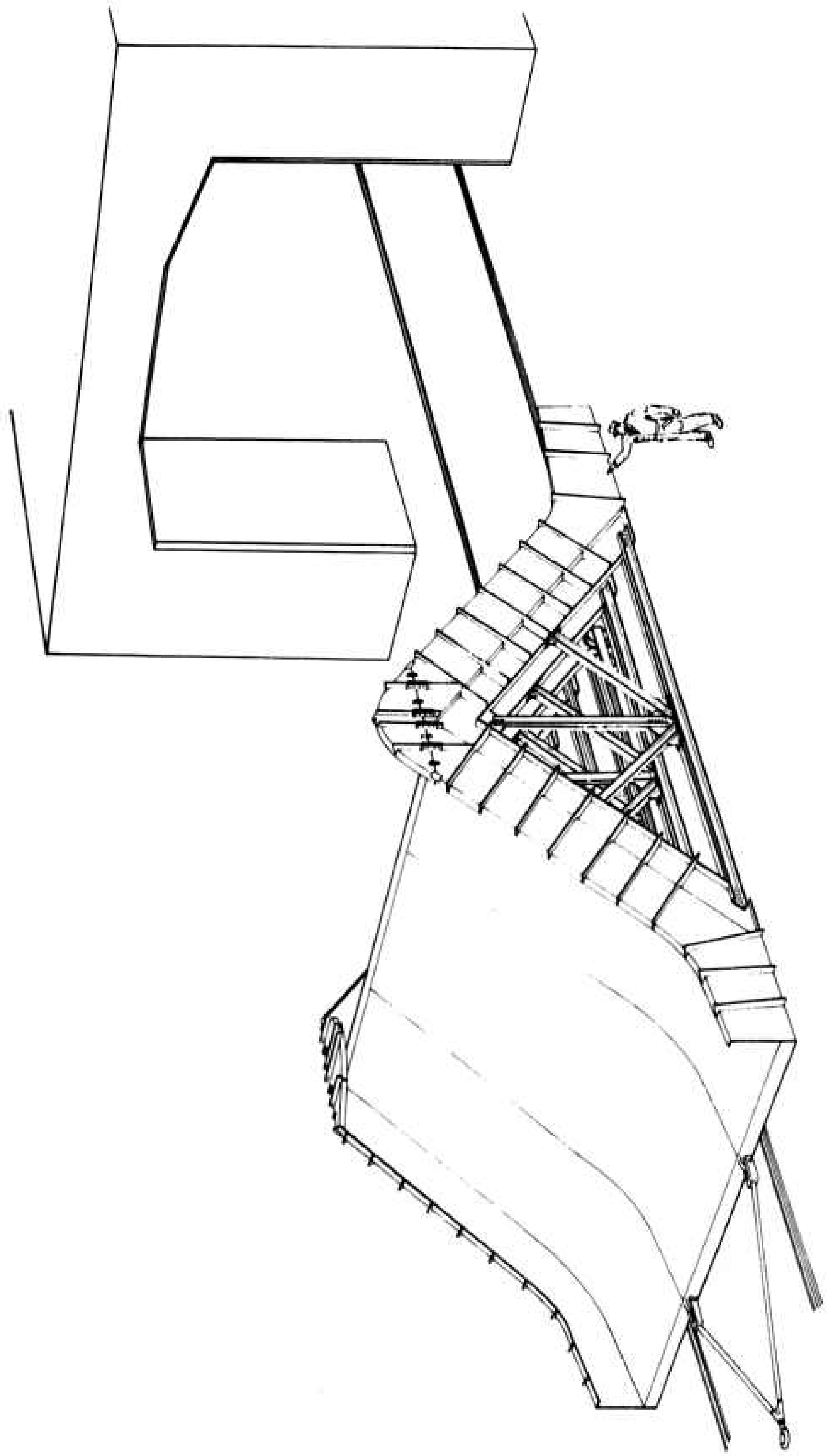
The method used to determine the elevation of the support arms, which are 12 feet above the launcher, is to suspend a target from the fixed support arms and use a level mounted on a portable instrument stand on the concrete base (Fig. 57). The procedure is as follows:

1. The instrument stand is positioned on the launcher base so that the hanging target can be seen when it is placed on each of the four fixed supports (Fig. 58).

2. The target micrometer is adjusted to its midrange.

3. The elevation movement of Fin I is adjusted to its midrange.

4. The target is suspended from the support for Fin 1 and the level adjusted until the cross hair of the level and the target coincide when the instrument is leveled.



SATURN LAUNCHER DEFLECTOR

FIG. 56

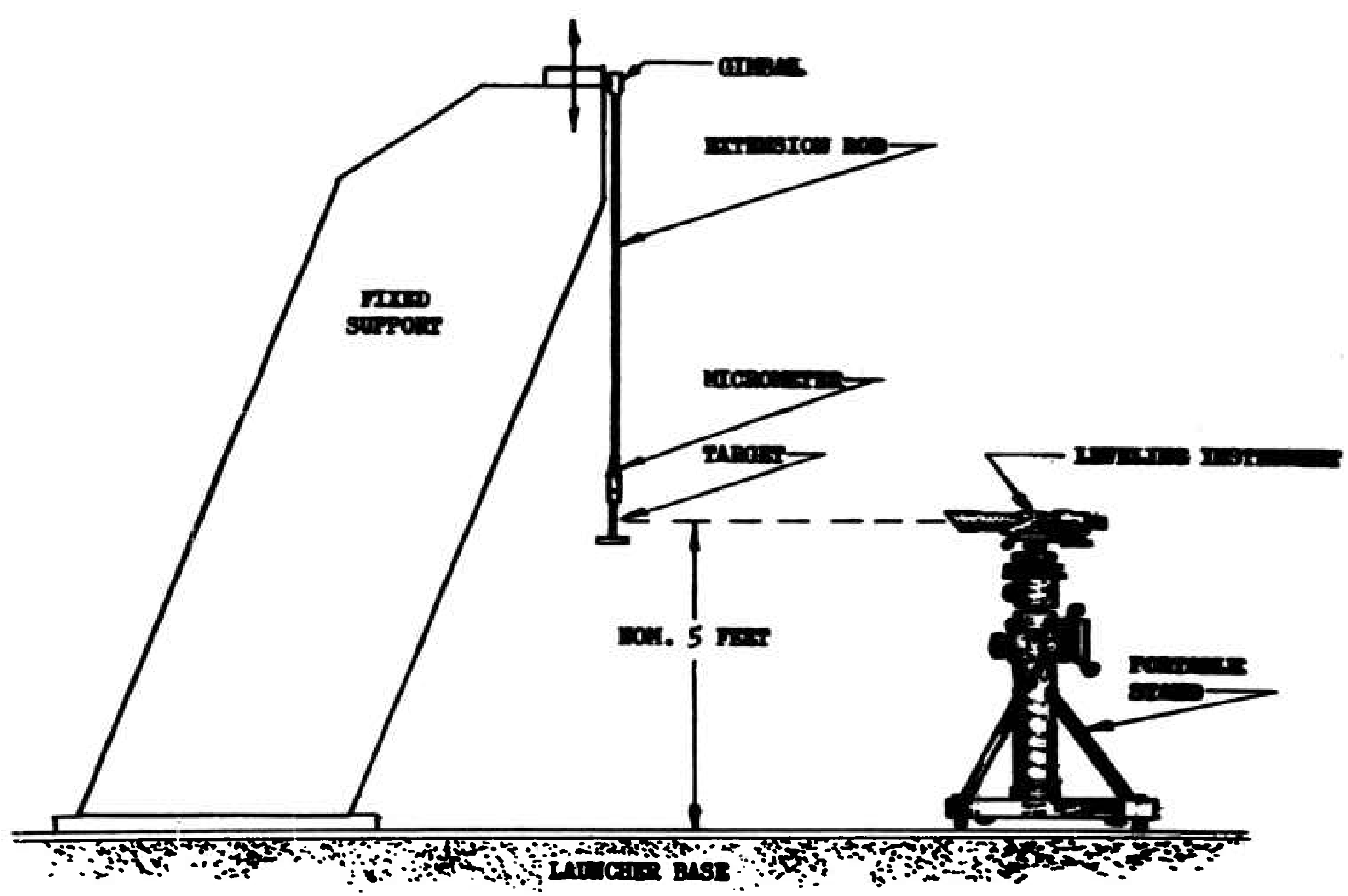


FIG. 57
ADJUSTING ELEVATION

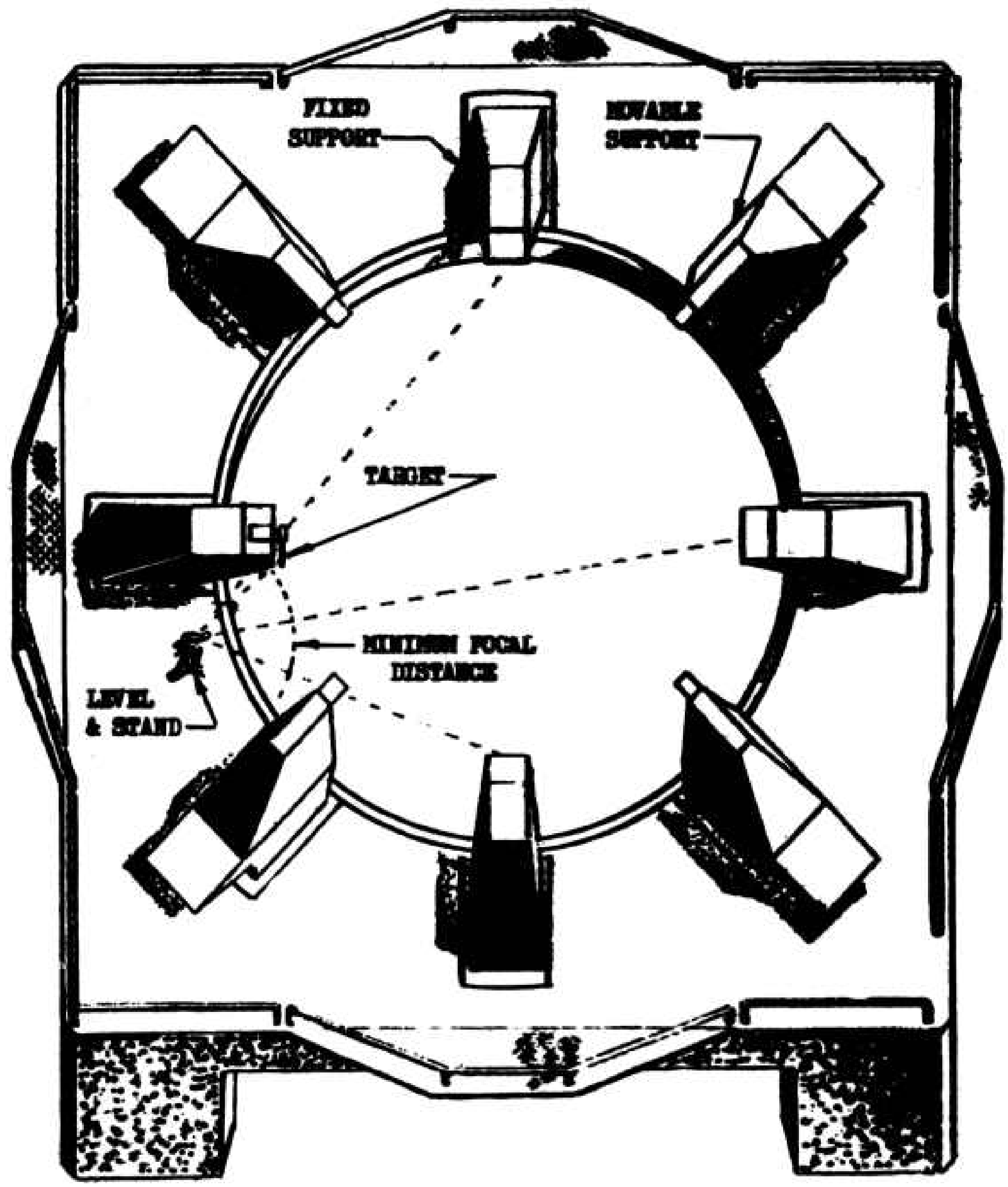


FIG. 58 LEVEL PLACEMENT FOR VIEWING THE TARGET IN ALL FIXED SUPPORTS

5. The target is now moved to the next support. If the second thrust frame support point is 1/8 inch lower than the Fin I point, then the target must be raised 1/8 inch.

6. The elevation of the support is lowered until the target coincides with the level cross hair. Since the effective length of the extension rod has been decreased 1/8 inch, the elevation of the second support will be 1/8 inch below that of the first support.

7. The procedure is repeated with the other two support arms.

After the booster has been placed in the launcher, the leveling can be checked by sighting with a theodolite, in two mutually perpendicular planes, the center line marks in the outer tanks (Fig. 59). These marks define the booster centerline within ± 5 minutes of arc. Using the tank marks have the added advantage of allowing deflections resulting from loading, wind, temperature, etc. to be measured. In order to keep the misalignment under the required 1° the centerline should be within 15 minutes of vertical. The booster can be leveled before it is fueled by adjusting the fixed arm support elevations.

To prevent the stabilized platform vertical from exceeding the limits of safety, the output of the platform can be monitored. If the platform output is beyond the limits, it can be nulled out by adjustment of the launcher-fixed supports. It is preferable to have a thrust vector misalignment then a platform output. By taking all of the tolerances into consideration, it should be possible to determine the angle between the booster thrust vector and the platform vertical within 6-1/2 minutes.

The launcher support arms allow some singular and radial adjustment. There is a $\pm 3/16$-inch tangential displacement and a $\pm 1/4$-inch radial displacement available.

G. Booster Servicing

Booster servicing encompasses the operations performed physically on the booster after it has been erected. It includes pneumatic leak and function testing, engine purging, engine removal, if required, propellant loading, and retrorocket installation.

The pneumatic leak and function test is made after booster erection to discover if there was any damage during transportation and handling. In making the test, gaseous nitrogen (GN_2) is distributed under pressure to various check-points in the system. The effects of the the GN_2 are monitored and any damage to the system is discovered. Distribution lines are also available for checking the top part of the booster.

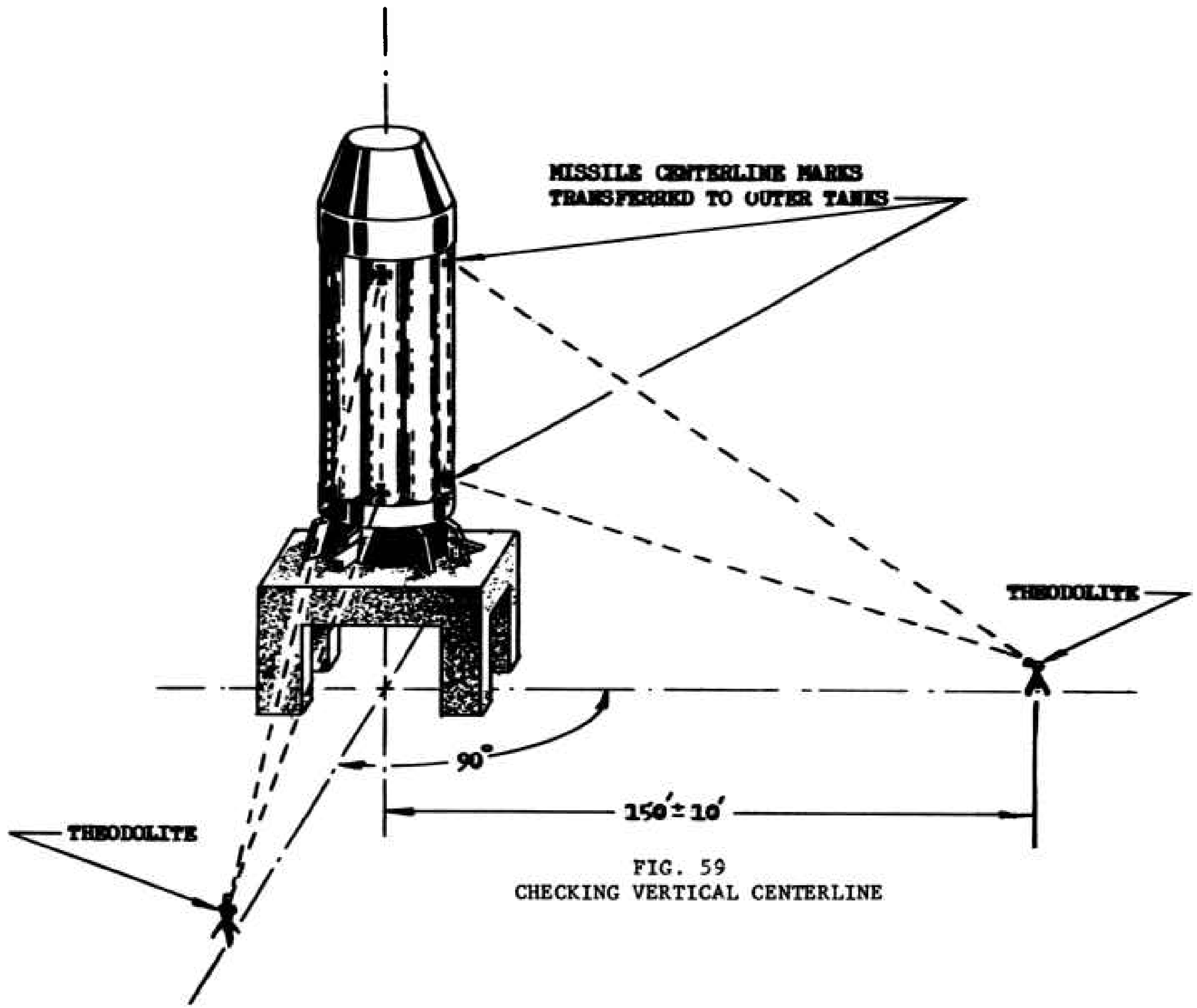


FIG. 59
CHECKING VERTICAL CENTERLINE

After this test, the engines are serviced with the engine servicing trailer. This operation includes flushing and purging the critical parts of the engine. This is done to insure the necessary absolute cleanliness of the engine.

Because of the tight working area in the booster tail section, a special handling system has been devised for removing and replacing the engines after the booster is on the launcher.

Engine removal is started by attaching a hoisting bar to the engine. The engine is then lowered by a hoist and pulley system until it is clear of the booster structure. Then it is fastened to a skid that has been raised into position by a service platform. The half of the service platform under the engine is lowered and taken from the area. The engine is now lowered, on its skid, along the face of the deflector, by means of a cable and hoist. It is removed from the base of the launcher by jib cranes mounted on the launcher. The procedure is reversed to install an engine.

The propellant is loaded on the SATURN from two separate systems: the lox storage and transfer system and the fuel storage and transfer system. The boarding of both propellants is automatic and is initiated and controlled from the blockhouse.

The lox system, (Fig. 60 and 61) which can service two pads at different time periods, consists of a 125,000-gallon insulated storage tank, a centrifugal cryogenic pump, and the necessary valves and lines to transfer the lox. The tank, a 41-foot sphere, has an evaporation loss of 0.2% per 24-hour period. The working pressure of the sphere, 40 psig, is maintained by a heat exchanger and self-pressurization. The pumps and lines for upper-stage loading are also in the system.

After it is loaded replenishing the lox is accomplished by using an additional 13000-gallon vacuum jacketed tank located in the storage area. The lox is fed from the tank, through lines, and into the booster through the lox replenishing coupling area. The purpose of the arm mounted on a three-inch universal joint, fastened to Fin III hold down arm and stationary support, is to provide lox to the booster until it leaves the pad. The arm is retracted by a pneumatic cylinder, which is activated when the booster lifts from the pad.

The fuel system (Figs. 62 and 63), which can service two pads at different times, consists of two 30,000-gallon cylindrical insulated tanks, two centrifugal pumps for fueling the booster, a recirculation pump, a filter-separation unit an educator system, and miscellaneous valves, lines and controls. The fuel level in the booster is controlled by a fuel tanking computer, and the fuel density is monitored at all times by a fuel density indicator.

LOX STORAGE AREA-SATURN

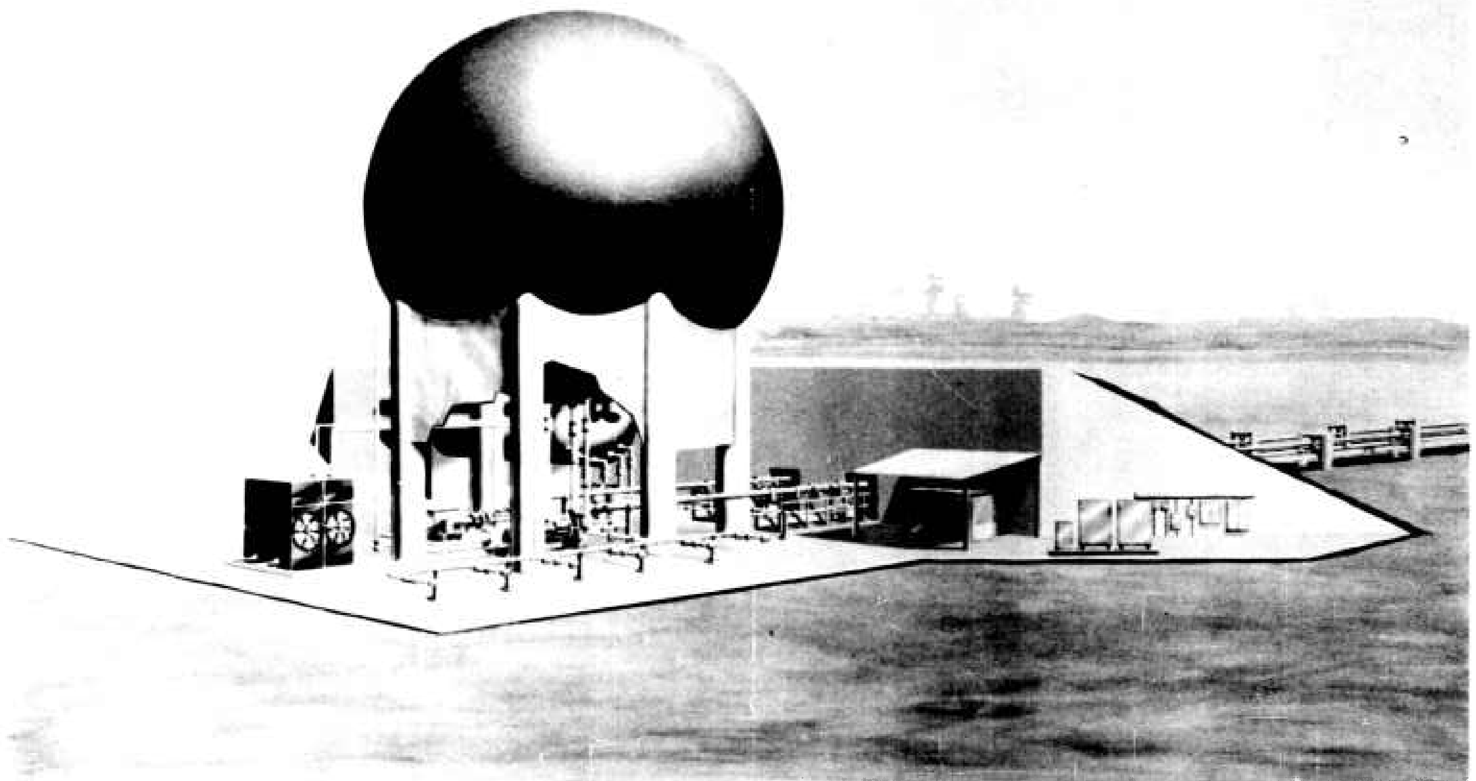


FIG. 60

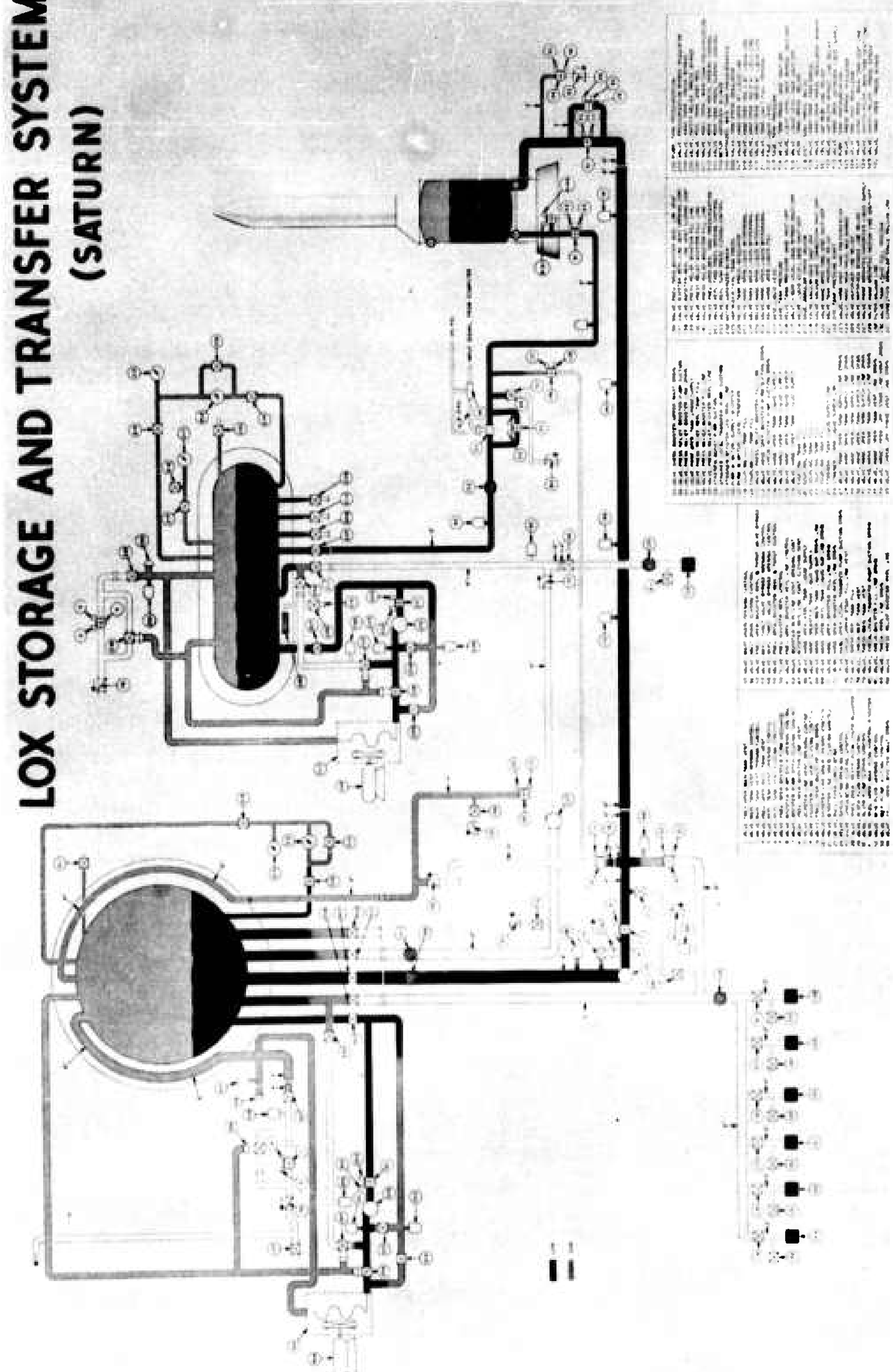


FIG. 61

FUEL STORAGE AREA - SATURN

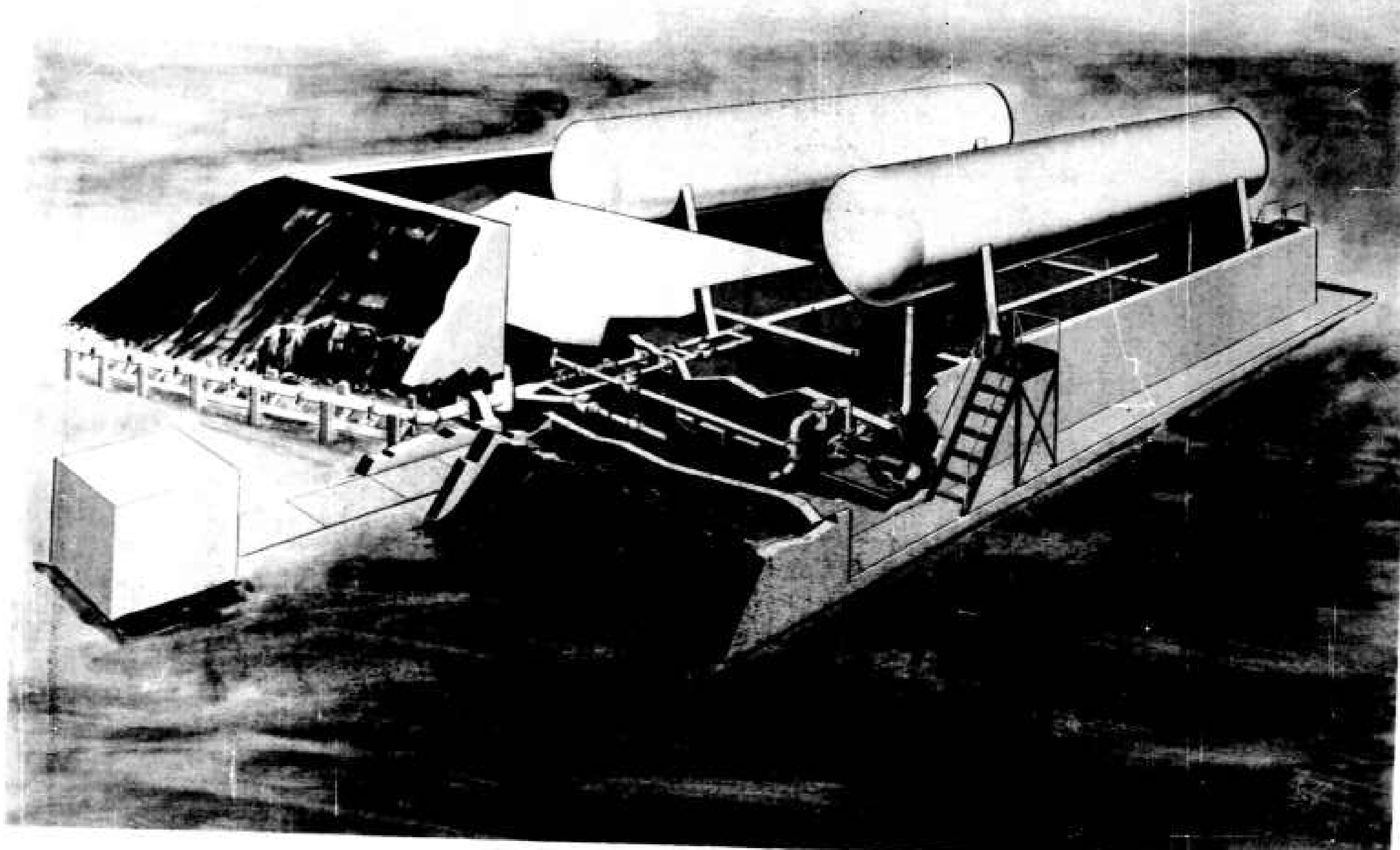


FIG. 62

FUEL SYSTEM SCHEMATIC
(SATURN)

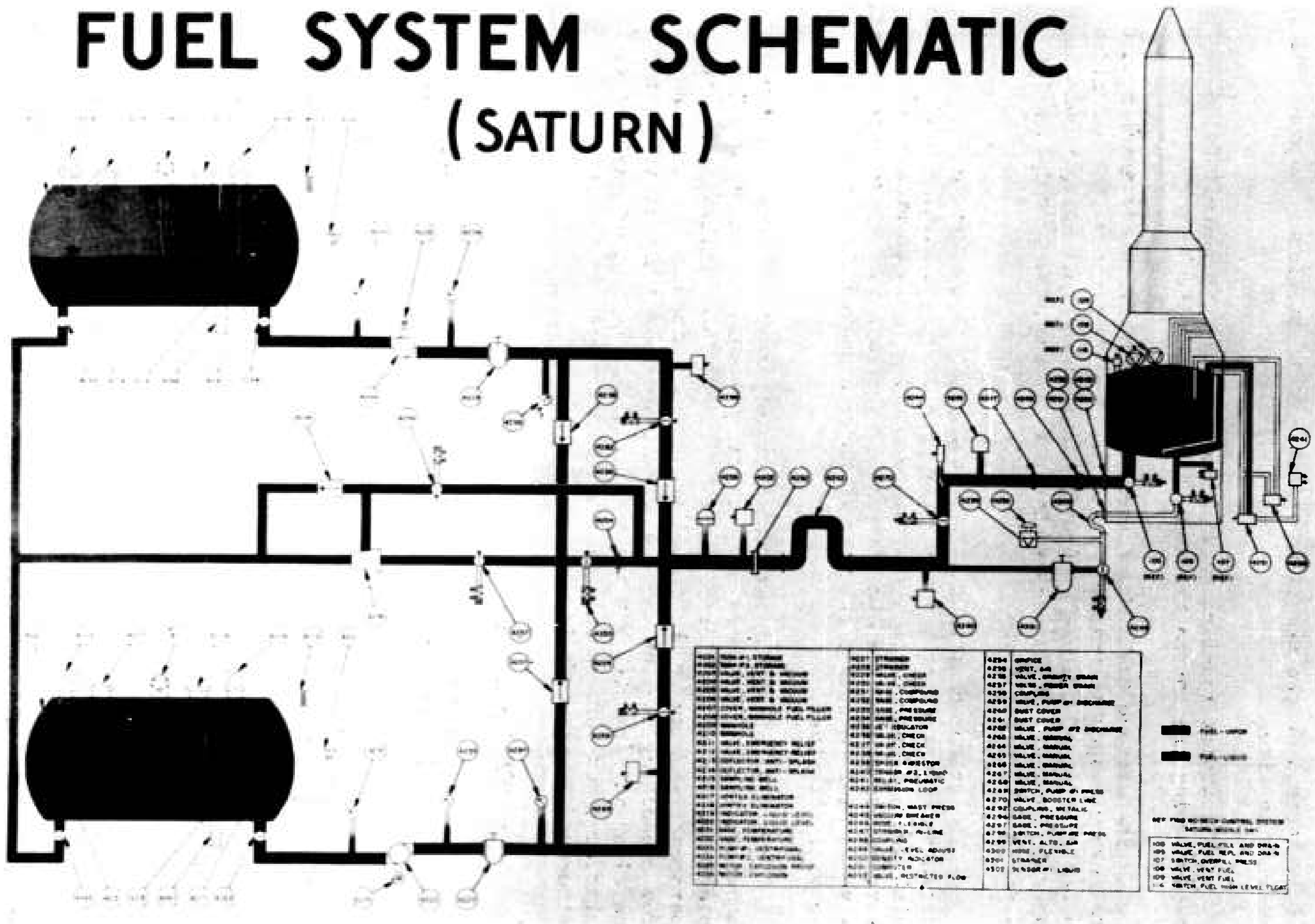


FIG. 63

The fuel tanks are over-filled so that the fuel computer can adjust the fuel level by draining it through the fuel density level control arm. This arm is identical to the lox replenishing arm except it is located on the Fin II hold-down arm and stationary support and is smaller in size.

The proper lox-fuel ratio is accomplished at liftoff by replenishing the lox to the level indicated by the fuel.

The storage tanks are filled through the filter-separation unit to insure proper filtration of the fuel and to minimize the entrained water content of the RP-1. During long storage periods, periodic operation of the filter-separation unit is required to maintain the desired fuel cleanliness.

The retrorocket installation equipment is composed on a retrorocket attaching bracket and a counter-weight lifting beam. These, with the jib boom derrick in each of the four main columns of the service structure, are used to install the rockets. These derricks will also be used to handle other heavy objects around the base of the service structure.

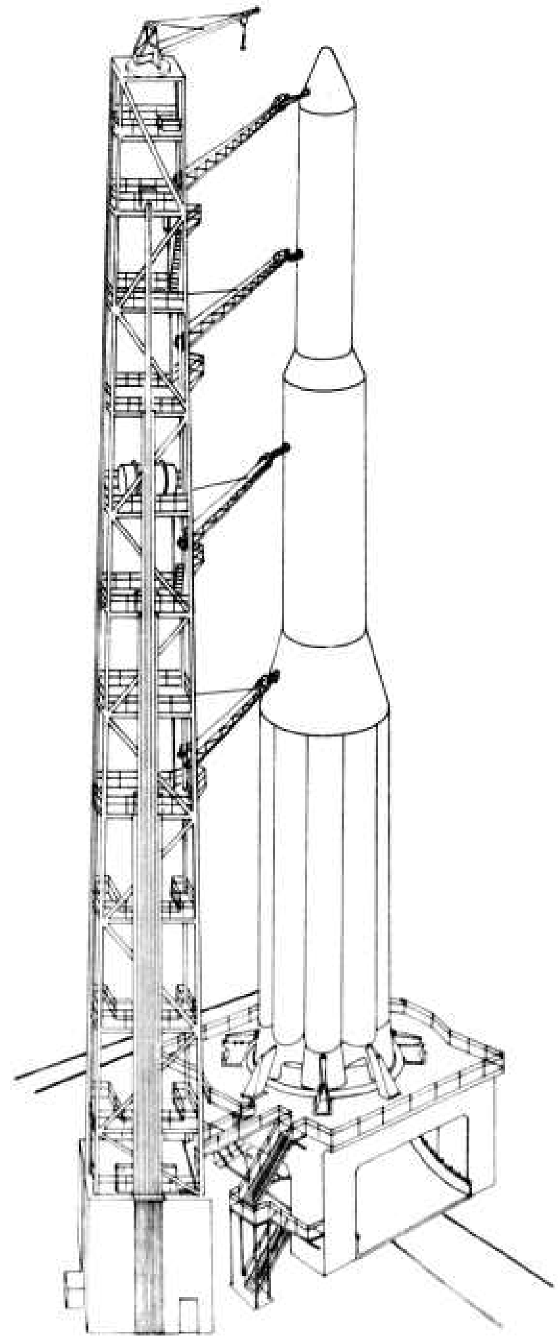
To give servicing personnel working areas and access to the booster, access platforms and servicing ladders are supplied. The platforms are on the launch pedestal to provide a working area for personnel working in the thrust frame area. Ladders are provided for connecting the lox and fuel masts, for making the final booster vertical adjustments, and for servicing the short cable mast. These platforms and ladders will be removed and stored on the service structure prior to firing.

H. Umbilical Tower

Because of funding limitations and the requirement for booster servicing only for the initial vehicles, the umbilical tower (Fig. 64) will be used to support and service the various lines and units that are required to service the SATURN prior to launch. These include electrical, pneumatic, and replenishing lines; liquid nitrogen cooling tanks; mechanical refrigeration units; and ground hydraulic units.

The tower will be built in three phases: the first will be construction of the 27-foot base; the second, construction of the steel tower to the 240-foot level, including an elevator and hoist; third, possible construction of a 30-foot extension providing an additional umbilical arm.

The base section will be a steel rigid frame with an inclosed section 24 x 24 x 27 feet. The wall will be provided with blowout panels to prevent structural damage in case of an explosion on the pad. The area will house mechanical, electrical, and fuel equipment used for



THREE STAGE SATURN
WITH UMBILICAL TOWER

FIG. 64

servicing the missile. It also will have space for the tower operating equipment required by phases 2 and 3. This section will be used, with a long cable mast, for SA-1, SA-2, and SA-3, since there is no requirement for servicing the dummy upper stages carried by these boosters. Just prior to firing the mast wall be retroacted 25^{o}-30^{o} by a ball screw mechanism and restrained to prevent mast damage in case of misfire.

The second phase of the tower will be four braced columns sloping inboard to a 10-foot square top. They will support the umbilical arms and service platforms. The tower will also have a 2000-pound elevator and a 3000-pound electric hoist at the top.

The umbilical arms are telescopic, extend horizontally towards the missile and swing 135^{o} in either direction. They are pivoted on the tower leg. The umbilical arms will automatically disengage and rotate out of the missile drift cone about 28 seconds prior to firing. The power to move the arms is supplied by a hydraulically actuated rotor unit at the pivot point.

The third phase will be a 30-foot extension of the tower, to 270-foot, if the growth of the SATURN warrants it.

I. Firing Accessories

The firing accessories are miscellaneous items needed to support the other equipment for preparing the SATURN for firing. These items include the short cable masts, the umbilical arm disconnect plate, and the lox and fuel filling masts.

The short cable masts, located at Fin II and IV provide the necessary electrical and pneumatic connections required for the eight engines until firing. The short cable masts are released simultaneously with the tail grab by two microswitches, each on a different retractable arm. The switches release pneumatic pressure which releases the masts.

The umbilical disconnect plate uses the same release mechanism employed in the short cable masts. The plate contains six electrical connectors, six 1/4-inch Wiggens connectors, one 3/4-inch Wiggens connector, one 3/4-inch GN_2 cooling connectors, and two 5-inch pre-cooling valves.

The lox and fuel filling masts (Fig. 65) are identical except the fuel mast is slightly longer and uses a sealing means that is compatible with the fuel.

The lox filling mast uses the JUPITER lox disconnect coupling and elbow but employs three new design features: (1) a rotating joint, (2) a self-sealing swivel joint, and (3) a lineal expansion and adjusting joint.

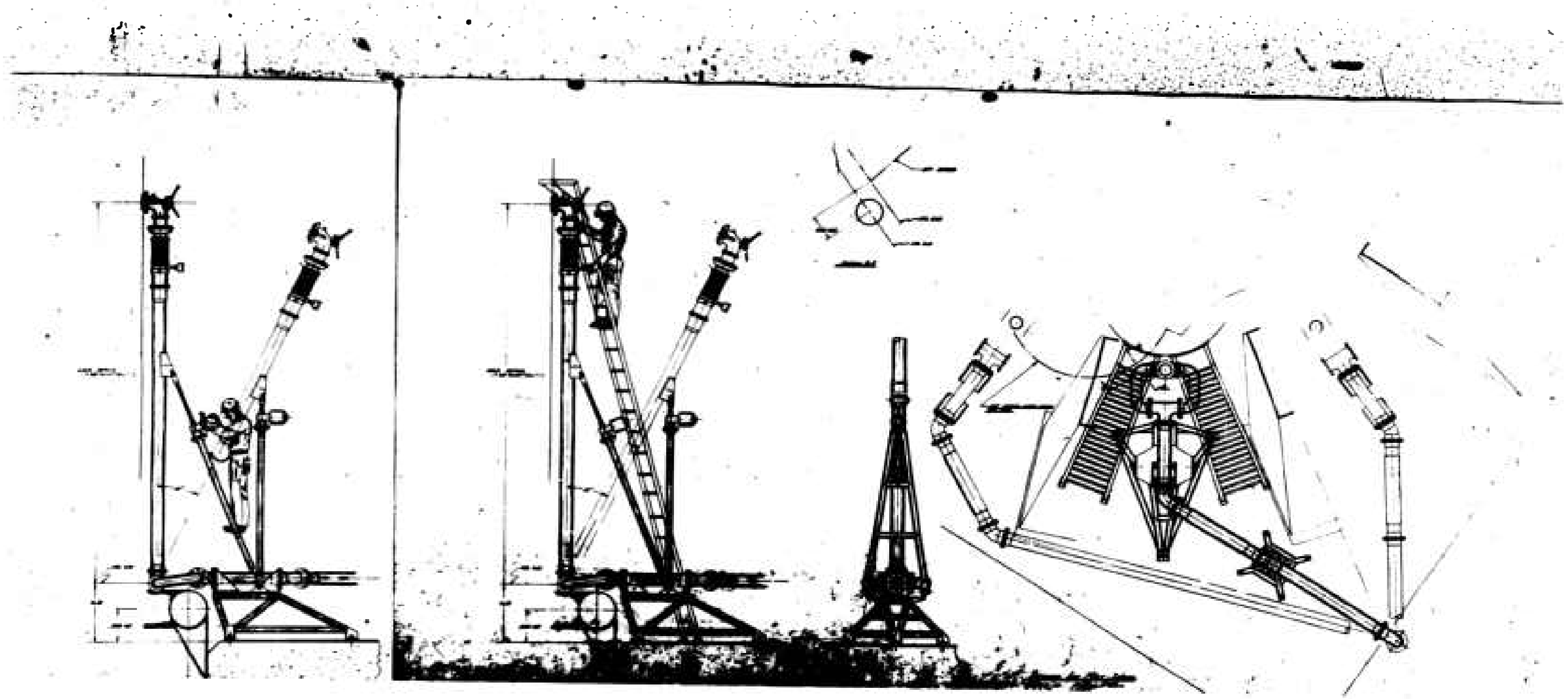


FIG. 65. LOX AND FUEL MASTS

The rotating joint will allow over 180° rotation in the lox piping set-up on the launch pedestal. Only 30°-35° rotation will be used when the mast is retracted and rotated out of the booster drift cone at firing.

The swivel joint has the conventional spherical seal but is constructed so that the presence of lox will cause a pressure differential sealing the mating surface. The joint is rigid when transferring lox, but is relatively free to swivel when lox is not being transferred.

The lineal expansion and adjusting joint uses a bellows to compensate for the expansion and contraction due to thermal differences. The bellows section is adjusted to the height required for the initial mating of the mast to the missile. A lip seal closes the joint.

SECTION IV. (U) FACILITIES

A. Introduction

The design and construction of all SATURN facilities has proceeded satisfactorily as facility criteria were not materially affected by the changes in missions and upper stage configurations. Work on the Static Test Tower East at Redstone Arsenal is nearing completion on schedule. Blockhouse construction at Cape Canaveral has proceeded to past 60% of completion and construction of the launch pad has been initiated. Meanwhile, plans were being prepared for a dynamic test facility required at ABMA which is required to accomplish preflight structural dynamics testing of the over-all vehicle configurations. Also, a proposal was forwarded for a second static test position which is required as a backup to the present facility in case of a catastrophic failure and to provide sufficient capability of doing both extensive R&D testing and handling the flight program simultaneously. A general description of the facilities was given in the first Semiannual Technical Summary Report, pages 57-70; however, more detail is included at this time.

B. Static Test Facilities

1. Static Test Tower, East

ABMA's new high thrust capacity static test tower has been provided for the SATURN Program to allow complete booster system operational checkout prior to shipment to the launch site. This procedure has enabled ABMA to correct any deficiencies at the assembly plant, increase reliability and save operational time at the launch site. The tower is designed to accept boosters with a maximum diameter of 25 feet in the 1.5 to 2-million pound thrust class (Fig. 66).

The principal vertical load carrying platform of the static test tower consists of a structural steel frame 34 x 34 x 6 feet. The load platform is supported on four "H" beam columns which serve a dual purpose by supporting all vertical dead loads of the new structure and resisting the vertical uplift during captive firings. These columns are anchored at the base through shear connectors which transfer loads into the reinforced concrete deflector pit walls that are a part of the new foundation. The foundation is supported by "H" piles which are driven to bedrock and by two 36-inch diameter reinforced concrete caissons which are also anchored in bedrock. Flame impingement on the steel flame deflector which rests on the deflector pit is calculated to resist about 50% of the vertical uplift.

The horizontal thrust component on the flame deflector is carried into the existing tower at its base to minimize the effect on the tower. The existing concrete tower is also utilized to support the structure against horizontal thrust, due to gimballing in the east-west direction.

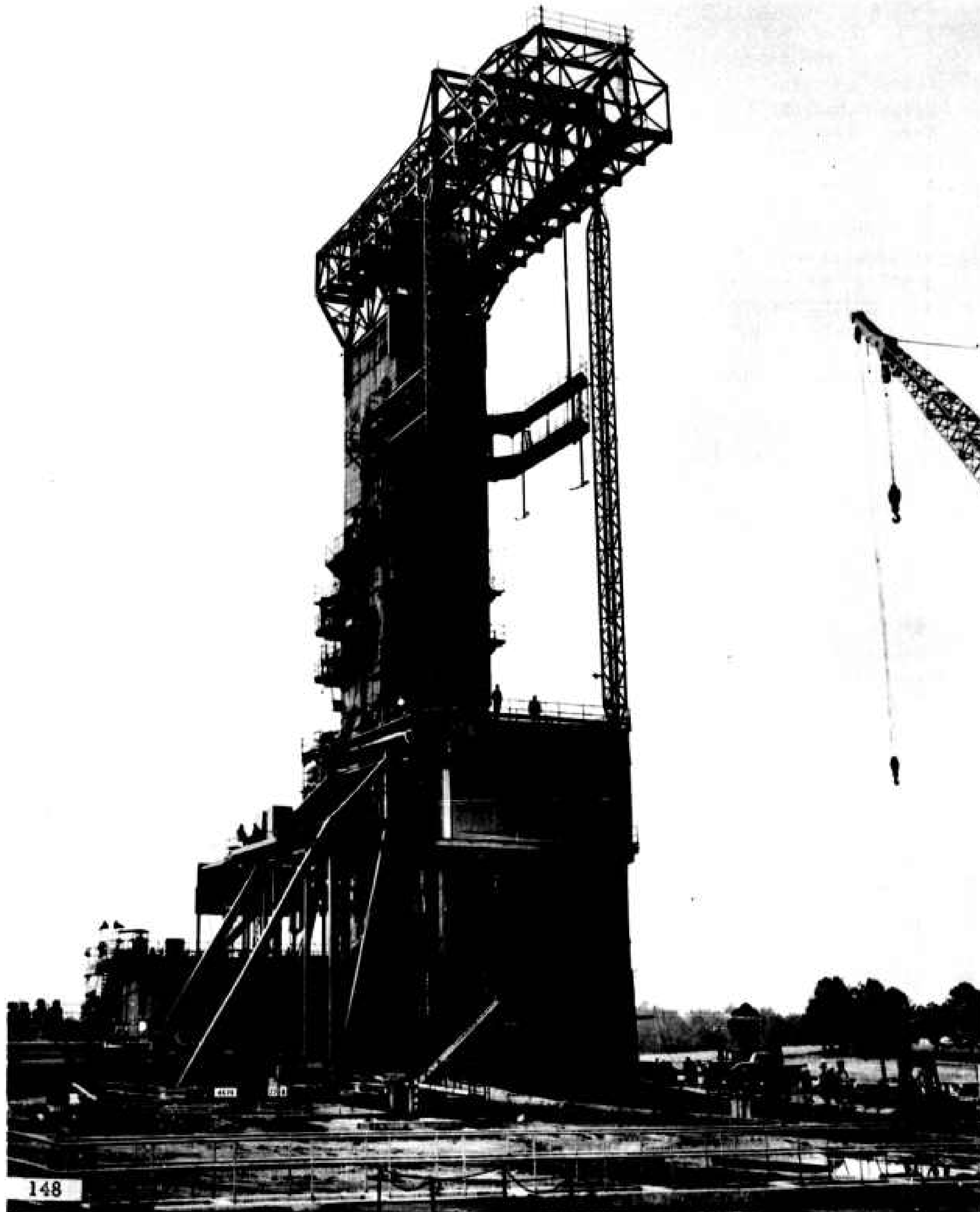


FIG. 66. STATIC TEST TOWER, EAST

Buttresses located south of the new tower were provided to resist the horizontal thrust component during gimballing in the north-south direction. The tower and buttresses are connected to the load platform by linking arms in order to permit relative deflections between buttresses and the load platform.

A 100-ton capacity bridge crane supported by the tower and two columns leading from the bridge to the main load platform is used to lift the booster into position. The booster, when installed in the tower, is connected directly to a thrust ring. The 29-foot diameter thrust ring with a 30-inch square cross-section weighs approximately 100,000 pounds. The thrust ring is horizontally stabilized by two tension-compression pipes to the buttress and two to the existing tower. The stabilizer rods are connected through load cells in order to measure in horizontal component of thrust. The thrust ring is connected through eight load cells to the main load platform for vertical support. Over 2,000,000 pounds of steel have been used in the steel tower structure, jet deflector, reinforcing steel, and "H" beam piles.

The deflector for the static test is a semi-bucket type fabricated from low carbon steel plate. It features a three-main-girder support system with a rigid framing system to maintain the shape of the flame deflector plates. Cooling water distribution system is such that cooling water jets may be made available by drilling the desired hole size at any desired location.

The deflector is fed with two supply lines with a total capacity of 40,000 gpm.

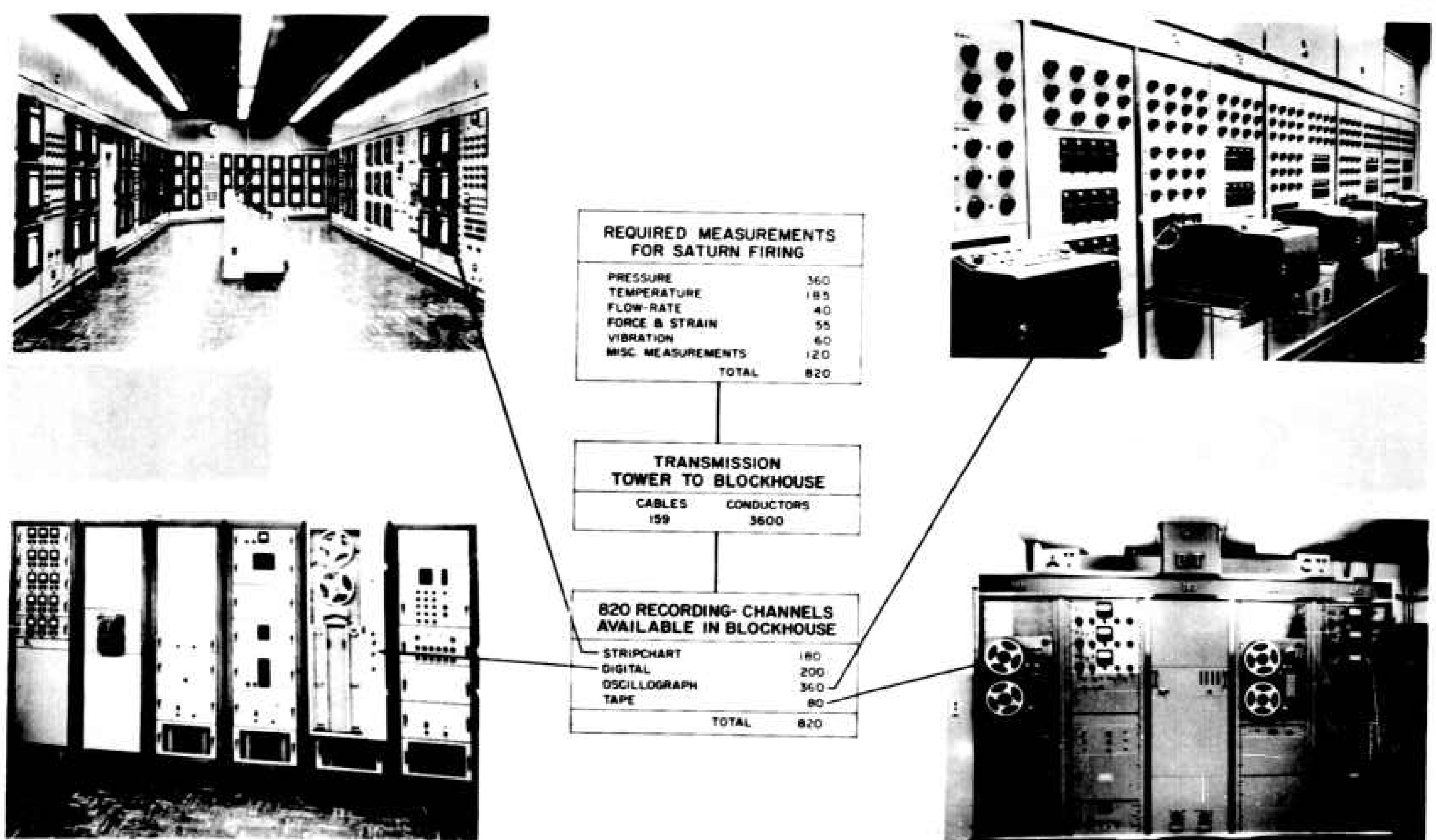
Model tests of the deflector confirmed that gas deflection from the test position is adequate with no ill effects to the deflector. The model test did reveal, however, that for clustered engines the overriding exhaust jets were forcing the gas stream down after exit.

The fire fighting system for SA-T at the static test tower is comprised of independently controlled systems for the engine nozzle area, engine compartments, suction line compartments, propellant containers, load platform area and tower wall. In case of fire, the system protecting the area in which the fire occurs would be manually activated from the control room. Up to 8000 gpm could be diverted from the deflector to the fire protection system immediately without overloading the pumping system.

2. Static Test Instrumentation

The development testing of rocket engines requires measurements with extremely high reliability, precision and accuracy. In the SATURN vehicle eight independently operating propulsion systems are employed; the instrumentation problems are also of increased numerical magnitude (Fig. 67).

FIG. 67 *Instrumentation*



The SATURN static tests will require 820 separate channels of instrumentation. Prior to buildup of instrumentation for SATURN firings there was a considerable amount of equipment existing from the REDSTONE and JUPITER programs. For example, 193 strip chart recorders to record thrust, weight, flow rate, temperature, and quick response measurements, a 100-channel MilliSADIC (for digital data acquisition) with a sampling rate of 400 samples per second, fourteen 36-channel oscillograph groups, six 7-channel magnetic tape recorders and two rough combustion cutoff devices were available. The interior of the existing blockhouse is shown in Figure 68.

Equipment which was added specifically for the SATURN program is listed below.

A 24-channel automatic thrust calibration system, a 16-channel temperature rate of rise circuitry, and a system for giving automatic warning on cutoff when a recorder reaches a prescribed upper or lower limit are used in conjunction with the strip chart recorders. A second 100-channel MilliSADIC was added and the sampling rate increased to a total of 2400 samples per second for the 200 channels. Also, 120 channels of the newest model Wiancko conditioning equipment, 174 amplifier channels, 24 bridge channels, 24 position channels, and 45 flow integrator channels were added. Eight more 36-channel oscillographs and three 14-channel tape recorders were installed. Finally, the ground-based rough combustion cutoff devices were increased from two to eight.

Of the 848 channels available, 193 are recorder channels, 200 digital, 374 oscillograph and 81 are tape channels.

3. Dynamic Test Position

ABMA, as a result of numerous studies, has recommended the construction of a dynamic test position where large, multi-stage space boosters could be assembled and subjected to various mechanical, structural and operational tests. Such a facility would allow checkout of mechanical mating features, determination of the natural bending characteristics of the complete vehicle configuration, both on the launch arms and under simulated flight conditions, determination of temperature distribution within the vehicle during standby conditions, after filled with liquid oxygen or hydrogen, and the effects of simulated vibration conditions on the vehicle's stages. In addition to this there will be the capability for other tests involving wind effects, fueling techniques, ground crew training, launcher arms checkout, and assembly methods.

The position will consist of two foundations capable of carrying the launcher arms and complete vehicle under fully loaded conditions plus a steel superstructure to provide lateral support and working platforms along any point, with a stiff-leg derrick on top to be used for handling and assembling the vehicle's stages (Fig. 69). One of the foundations will be used for tests requiring numerous work platforms and

FIG. 68. TEST CONTROL BLOCKHOUSE, INTERIOR

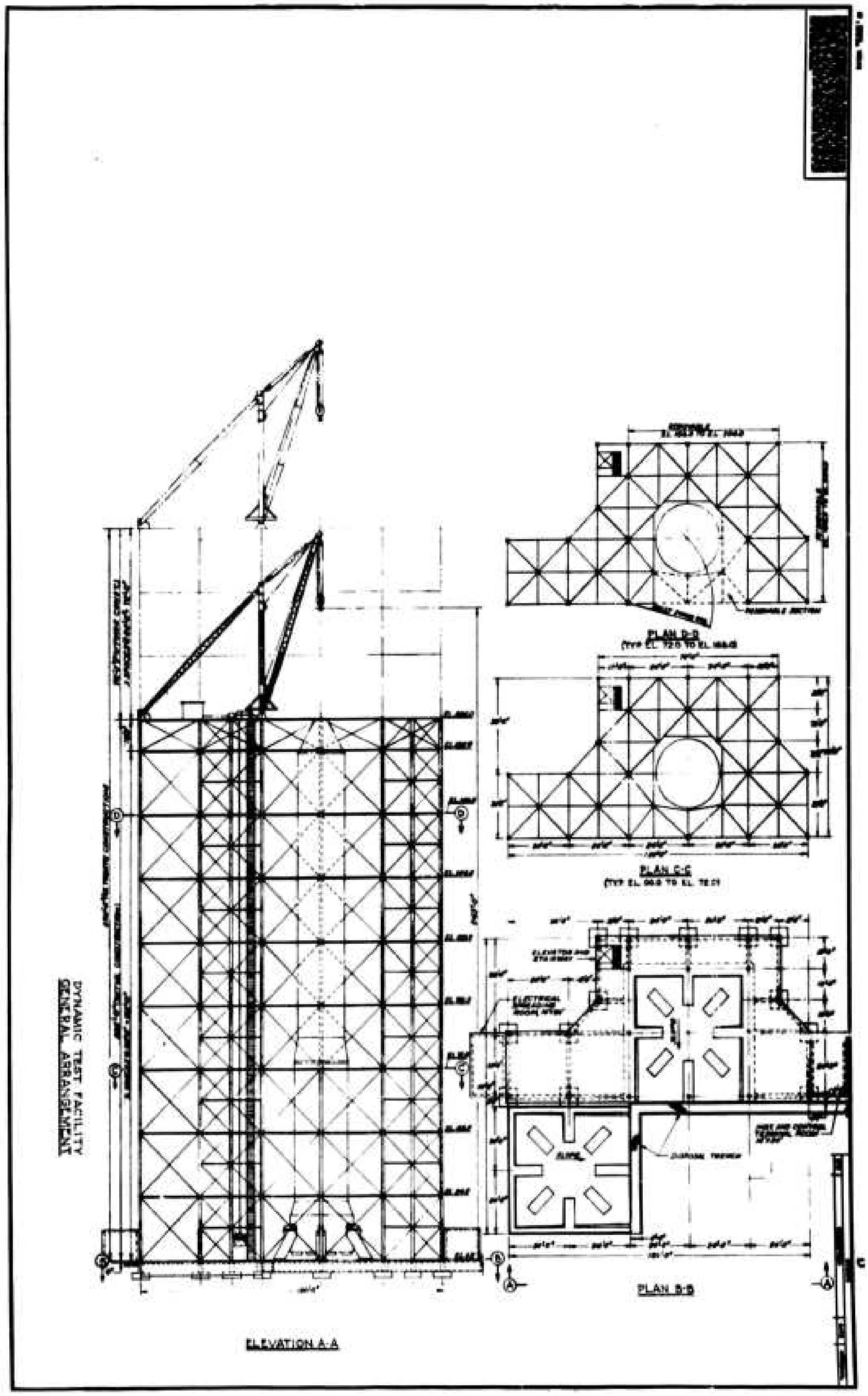


FIG. 69. DYNAMIC TEST POSITION

maximum support for the vehicle while the other will be used when minimum fixed surrounding structure is desired. Access to the upper platforms of the structure will be by means of stairs and an elevator. Means are incorporated for carrying spilled propellants to remote holding pounds. Recording of test data will be accomplished with equipment available in an existing blockhouse.

4. Static Test Tower No. 2

To supplement the only position available for static testing of high thrust space vehicle boosters, ABMA has proposed the construction of a second static test facility. The new facility will be capable of handling boosters in the thrust category envisioned for the present and future SATURN boosters and will supplement the existing stand when both extensive R&D testing and final operational checks for flight vehicles are required to be done simultaneously. Of more importance, perhaps, is that the new position will serve as a backup in case a catastrophe should be encountered on the first tower.

The tower will consist of a heavy steel structure capable of carrying the required vertical loads, both static and dynamic, plus a lighter structure which will absorb horizontal loads from engine gimballing or from winds. Service and work platforms will be supported by the tower. Leading from the tower will be a shallow, concrete-lined trench which will drain the deflector cooling water to a holding pond which, in turn, will discharge into an adjacent creek. Immediately adjacent to the tower will be a two-story concrete building used for terminating instrumentation and control cables and for housing special power generators and power switch panels. A cable tunnel will lead from the termination building to a new instrumentation and control blockhouse, which will also be connected to the existing blockhouse and instrumentation through a bank of conduits. The new blockhouse will contain all the control equipment and consoles and the recorders needed for testing the booster in the new tower. The remainder of existing recording instrumentation will be provided from the existing blockhouse. A gantry crane with its own system of tracks will be provided to handle the large boosters.

C. Launch Facilities

1. Blockhouse

In order to conserve the limited funds made available for the program, the basic design of existing ICBM blockhouses was also adopted for the SATURN. Some changes were made to the interior arrangement of the blockhouse to accommodate the unique requirements of the SATURN vehicle. The purpose of the blockhouse is to protect the firing personnel and instrumentation during launches. This is accomplished by utilizing a reinforced concrete dome molded in a semi-eliptic shape strong enough to withstand a pressure force of 320,000 psf. The thickness of the concrete dome is required to be 5 feet at the top increasing to

6 feet 4 inches around the bottom periphery to withstand this pressure. The concrete is covered with sand ranging from 7 feet in thickness at the top to 15 feet at the bottom. Retaining the sand is a 4-inch layer of sprayed concrete covering the outer surface. The blockhouse provides a total of 12,097 square feet of usable floor space. Two floors are provided. The first floor will be used to house measuring instrumentation, tracking and telemetry equipment (Fig. 70). On the second floor all the test monitoring and control equipment will be installed (Fig. 71). Separate consoles are provided for each stage. As presently envisioned, each stage will have a different test conductor who will report to the chief test conductor. A small area on the second floor will be reserved for observers who can witness test progress through television monitors. There is no direct visual observation of the pad area except through periscopes provided for the test conductors.

2. Launch Complex

The SATURN launch complex includes the launch pad, a high pressure gas system, lox storage emplacement, fuel storage emplacement, service structure parking site, launch pedestal, instrument tunnel, communications equipment building and several instrumentation sites (Fig. 72). Three primary considerations have governed the complex layout. First, minimum allowable separation between various elements of the complex is desirable from an efficiency standpoint. Second, the elements were required to be oriented in a certain manner because of neighboring facilities and launch azimuth considerations. Finally, minimum mutual interference between elements had to be incorporated wherever possible. The pad to blockhouse, pad to fuel emplacement, pad to lox emplacement and lox emplacement to fuel emplacement distances were governed by mutual agreement with safety representatives and range personnel. Where practicable, the facilities were located so as to take advantage of the shielding provided by the four corner columns of the launch pedestal. Orientation of the launch pedestal, umbilical mast and service structure rails were set to provide a launch azimuth freedom from 44° to 110°, and to provide a clear line of sight from the erected vehicle to both the central control building and the blockhouse.

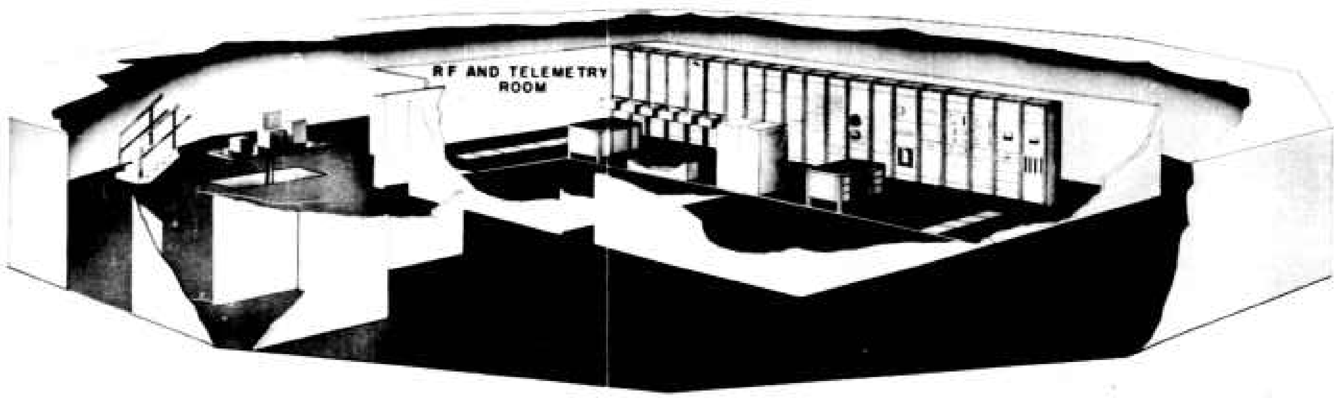
The launch pad, of a 430-foot diameter circular shape, is being constructed of reinforced concrete with the center depressed to allow installation of refractory brick in order to simplify replacing surface areas damaged by firing. Rooms are provided below the pad to house electric cableways, terminal boards, instrumentation racks and mechanical and electrical equipment required for the various stages. Also, recorders for the measuring equipment will be located underneath the pad. This relieves blockhouse space requirements and reduces lead lengths to the vehicle.

A high pressure gas facility is provided to supply sufficient air and nitrogen for the vehicle. This facility contains a bank of storage cylinders, inlet manifolds, nitrogen dryer-filter units, an electric

FIG. 70

SATURN BLOCKHOUSE

VLF 34

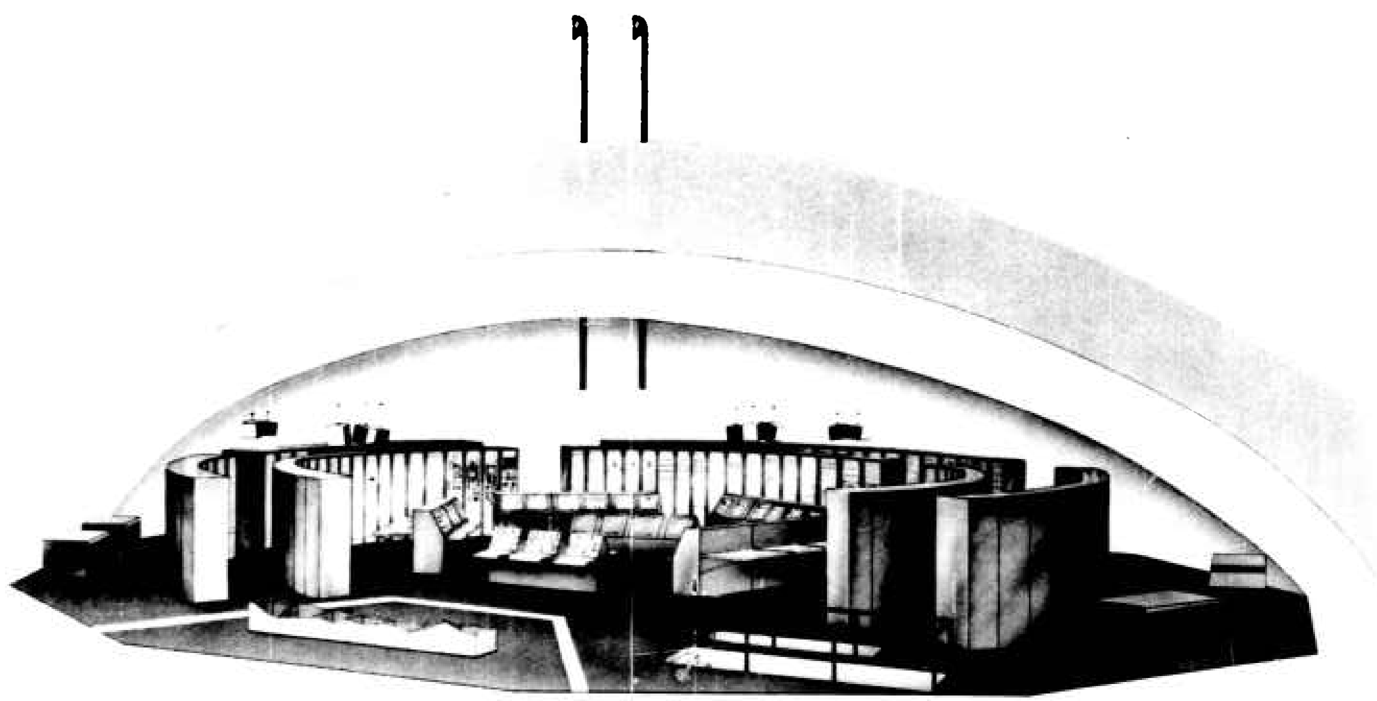


1st Floor Equipment Layout

FIG. 71

SATURN BLOCKHOUSE

VLF 34



2nd Floor Equipment Layout

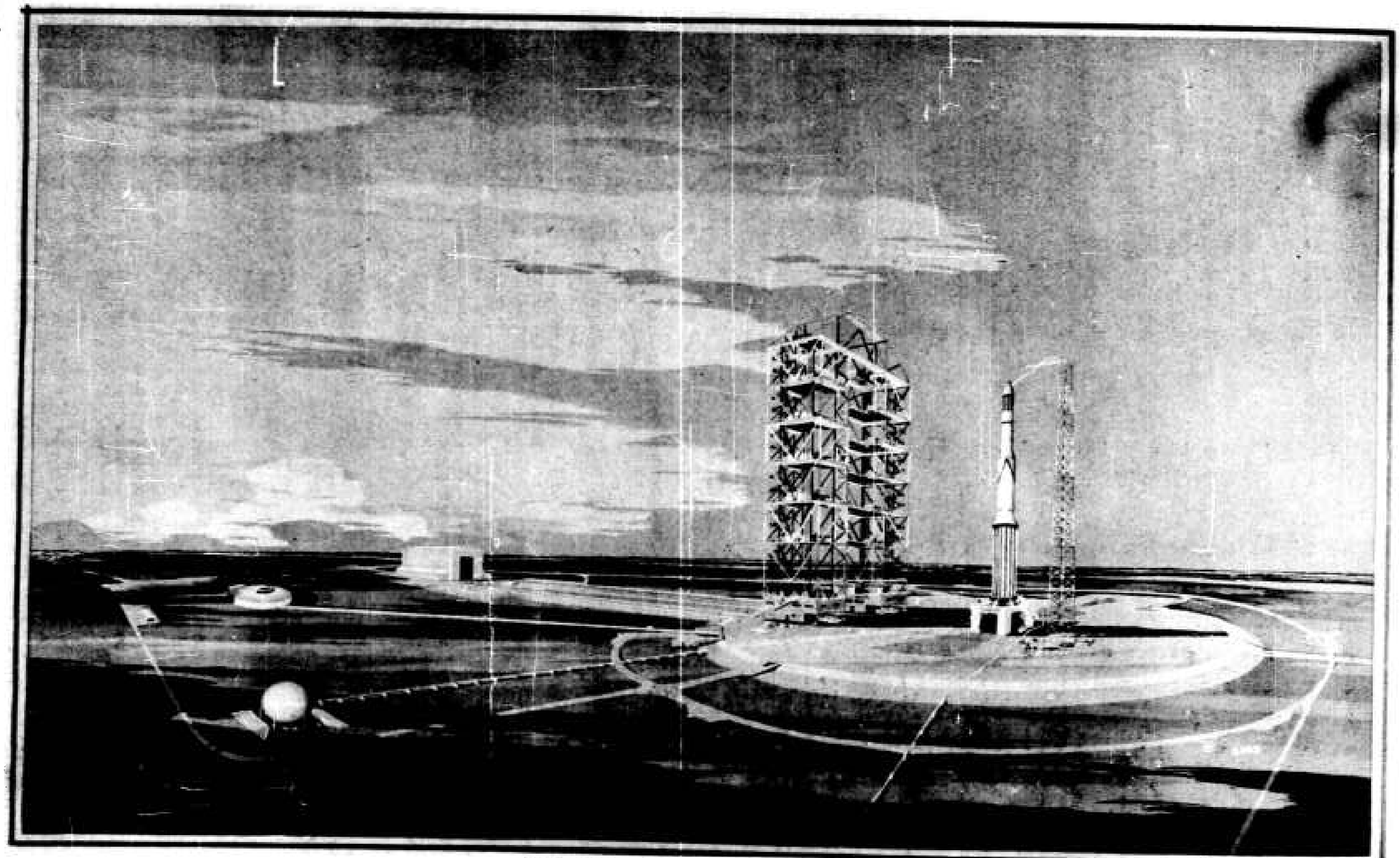


FIG. 72. LAUNCH COMPLEX LAYOUT

distribution room, and parking facilities for trailer mounted portable gas compression and evaporation systems.

Other areas, like the lox and fuel storage facilities, the launch pedestal, and the umbilical tower are described in Section III, Support Equipment.

3. Service Structure

The SATURN service structure, required to accomplish erection of the SATURN vehicle and to provide work platforms for the checkout and servicing operations, is an inverted U shape, rigid box truss frame design (Fig. 73). The center top box truss with bridge runway hoisting equipment is approximately 130 feet long, 70 feet wide and 50 feet deep. It rests on two box truss columns of 70 x 35-foot cross sections, each of which rests on a two-story box truss base section, 37 feet wide and 70 feet long. The total structure will be approximately 310 feet high, 130 feet wide and 70 feet long at the base. A bridge crane with vertically positionable runway trusses clamped to and supported by the main vertical members of the tower legs will have mounted, individually controlled, 40- and 60-ton hoists furnishing a 245-foot hook height. The forward reach of both hooks will be 28 feet with a lateral reach of 20 feet. The service structure design is having provisions incorporated for vertical extension in order to provide a possible maximum hook height of 281 feet in the future when required. The structure is equipped with seven fixed platforms within the tower legs, each of 790 square feet floor area, and six inclosed retractable platforms each of 814 square feet floor area. The platforms are designed for a capacity of 12 personnel and 600 pounds of equipment on each half. Vertical adjustment of the platforms from the 82-foot to 224-foot level will be provided. The tower will be equipped with two personnel and one freight elevator, each of 500-lb capacity. The two 25-foot high base sections of the service structure contain engineering and laboratory spaces and power equipment required for self locomotion of the tower on a dual track railway into the parking position. The service structure is mounted on four 12-wheel retractable rail trucks which are retracted after the tower is moved into position, allowing it to rest on rigid foundations. Hold-downs will be provided to act against overturning moments from wind pressures. Ground power will be used for all operations except locomotion and hazard lighting. The structure will be equipped with lighting protection and grounding systems, aircraft warning lights, and illumination for night operation. Provisions will also be included to allow rapid evacuation from all levels at checkout position in case of an emergency. The total weight of the service structure is presently estimated at 2400 tons.

4. Instrumentation

Requirements for measuring and tracking in the SATURN program are of considerably greater magnitude than existing programs,

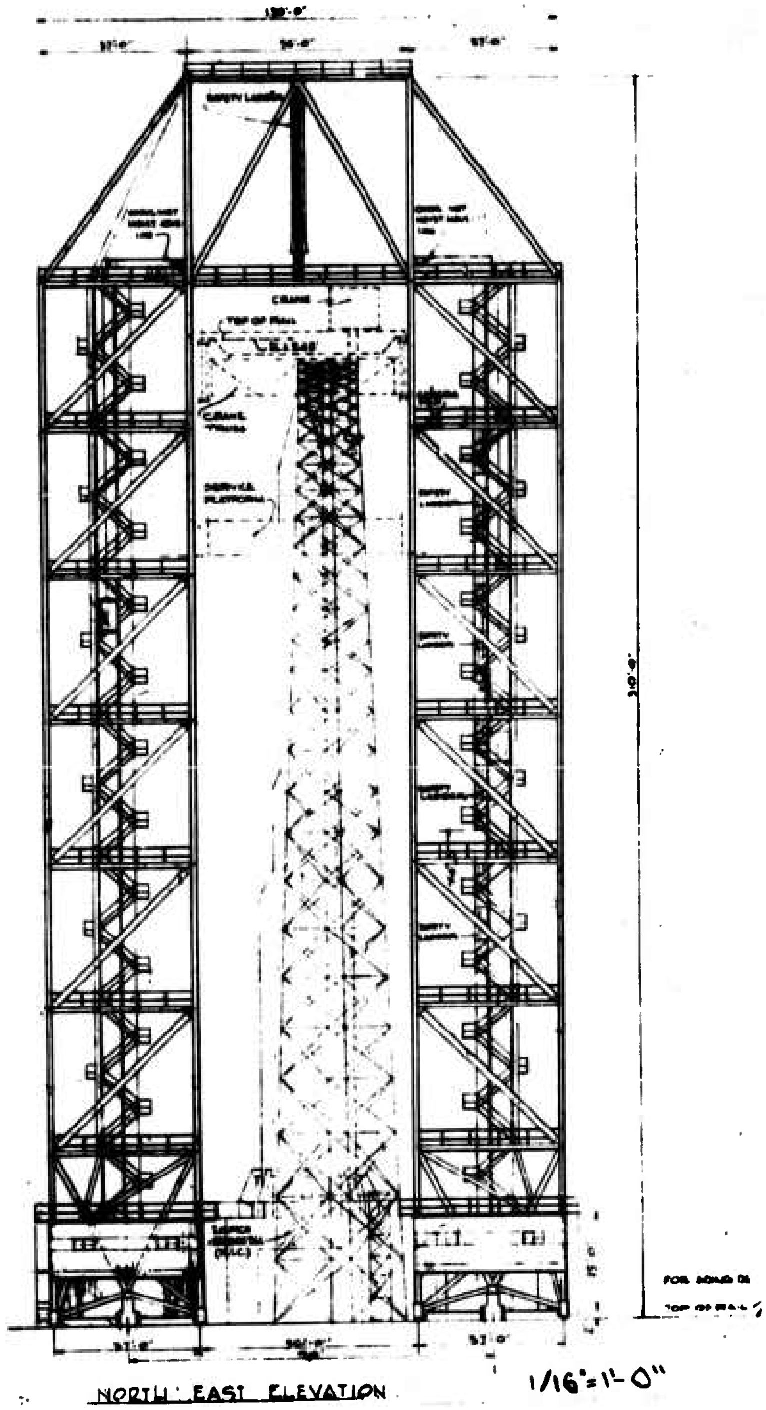


FIG. 73. SERVICE STRUCTURE

both in the scope and accuracy required. Approximately 550 measurements have been planned for the booster alone. Also, the need for tracking up to the injection point of a multi-stage vehicle greatly extends the distances over which information must be fed back to the control center, especially as the investment involved makes it mandatory that the vehicle be tracked with unprecedented accuracy. To satisfy these requirements, development has been initiated for a phase locked UDOP (ultra high frequency doppler velocity and position) system which will represent a significant advance in the state of the art. In addition to this, development of a 19-foot parabolic tracking antenna will extend tracking ranges and achieve greater accuracy. Standard equipment will be used wherever possible, of course, including recorders, plotting boards, computers for the blockhouse and discriminators, recorders and monitor receivers for the blockhouse telemetry station. The blockhouse telemetry will also include an Arnoux decommutation system and a panoramic telemetering indicator. Tape recorders will be used wherever possible for recording of high frequency response data. These recorders will be located in the pad utility room with only the remote controls in the blockhouse. Both electronic and optical systems will be required to achieve the desired accuracy necessary to secure trajectory data. In addition to UDOP, C-band radar, S-band radar and Azusa, it is planned that CZR cameras, ballistic cameras and cine theodolites will be used. In general, the most modern developments in instrumentation are being explored for possible application in the SATURN system.

SECTION V. RECOVERY

A. (S) General

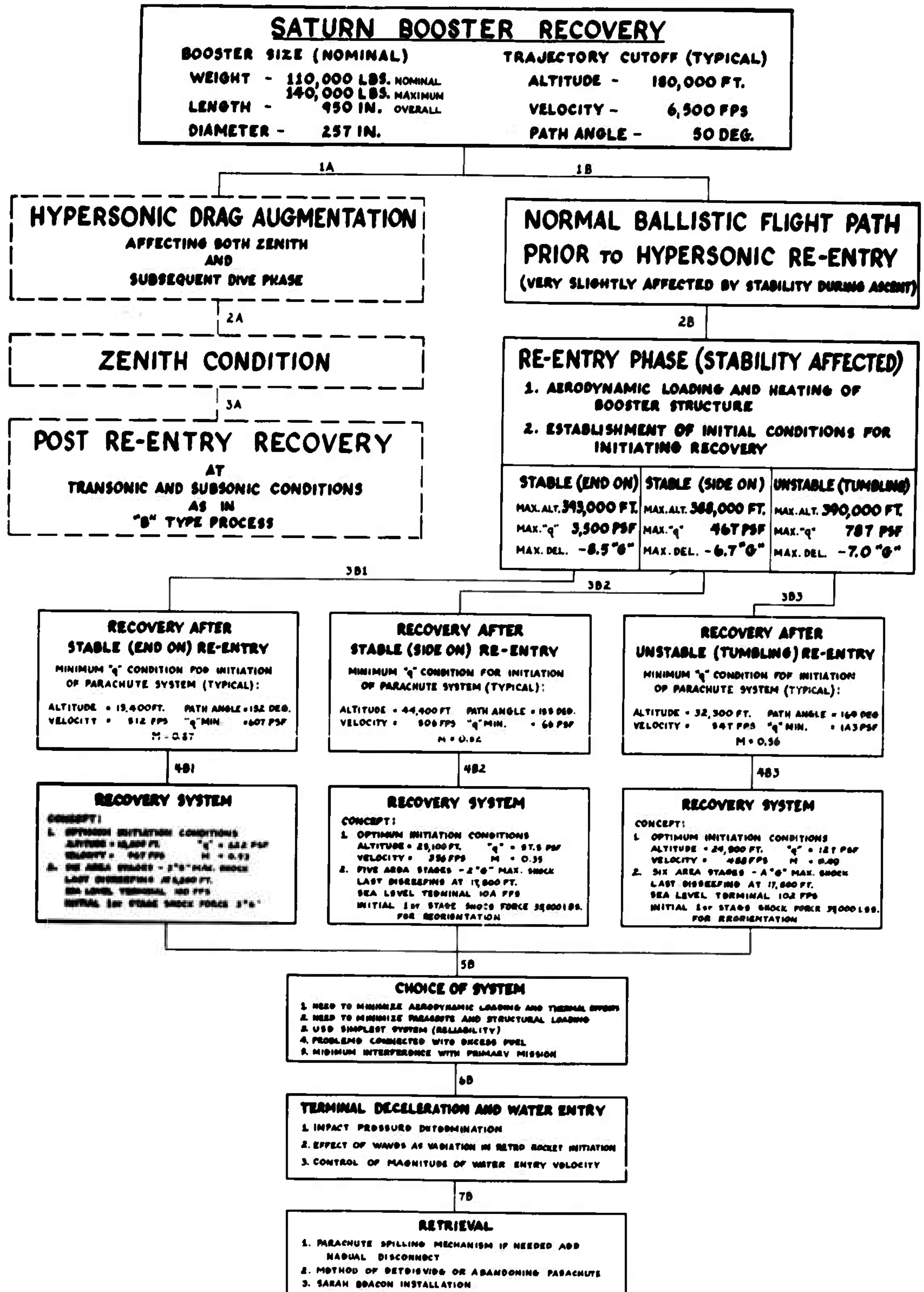
On the first SATURN flights, recovery of the booster section for laboratory examination of its structural and mechanical components is desirable. On later flights it is expected that the booster, or at least the more valuable parts of it, can be recovered for future flights. (This pertains particularly to the rocket engine since about 80% of the cost of the booster is in the power plant section at the tail.) For these reasons the recovery program was initiated.

The approach to the recovery problem is to use parachute drag devices for initial deceleration and retrorockets to produce the final deceleration prior to water entry. Initial studies of booster aerodynamics and trajectory showed that if the booster passed through re-entry with its axis parallel to the trajectory path, high aerodynamic loads, accompanied by rather severe heating, would ensue.

Two lines of approach were considered:

a. On one scheme (1A on Fig. 74) some form of hypersonic drag augmentation following cutoff could be used:

FIG. 74. SATURN BOOSTER RECOVERY DIAGRAM



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(1) The purpose would be to reduce the zenith altitude and to modify the subsequent dive phase in such a way as to reduce heating and aerodynamic loading.

(2) Hypersonic drag augmentation might take the form of dive brakes or some form of parachutes of heat resistant material.

(3) At re-entry the booster would probably be slowed down further by conventional parachutes and finally by retrorockets.

(4) The hypersonic drag augmentation might or might not be disposed of during the latter part of re-entry or immediately following re-entry.

b. In the alternate approach (1B on Fig. 74) the booster would be allowed to follow its normal ballistic flight path prior to re-entry. This path is affected very little by the attitude of the booster prior to the start of re-entry because of the low atmospheric density that prevails at the altitude of separation.

On the other hand, the re-entry phase of the flight and the aerodynamic heating and loading that prevail during re-entry will be highly dependent upon the stability of the body and upon the attitude which the body maintains if it is stable. Furthermore, the altitude at which conditions acceptable for the deployment of conventional parachutes systems are obtained will also be quite dependent upon attitude during re-entry. Some degree of stability during the subsonic post re-entry phase is necessary for the successful deployment of parachutes.

Point mass trajectories were computed, using early available drag data for the booster. In one case, it was assumed that the booster is stable with its axis parallel to the path; in a second case the booster was assumed to be stabilized with its axis normal to the path; and in a third case, an intermediate drag area, representative of end-over-end tumbling, was used. The results for the three trajectories based on the early drag data are given in 2B on Figure 74. From these results, the desirability of avoiding end-wise stable attitude during re-entry was apparent. Investigation of feasible ways of ensuring a broadside or nearly broadside re-entry was needed.

Study of the graphical plots of re-entry trajectory data as a function of altitude showed that as altitude decreased there is an altitude at which the dynamic pressure, q, reaches a minimum and then begins to increase again as altitude continues to decrease. The magnitude of the minimum q and the altitude at which it occurs gave a rather good indication of the severity of the recovery problem from the viewpoint of the parachute system designer. The minimum q versus altitude plots were rather broad and indicated that deployment of the parachutes at altitudes somewhat greater or lesser than the altitude of minimum q would not have a great effect upon deployment conditions. Parachute system designs were studied for the three trajectories: 3B1, 3B2, and 3B3 of Figure 74.

The study showed, from the standpoint of the parachute system designer, that recovery is feasible after re-entry in any attitude, and that it is much easier with broadside re-entry and that lower opening shock loads of 2g maximum can be obtained as contrasted with 3g maximum for endwise re-entry.

The choice of a re-entry attitude and a suitable parachute system depends on the following considerations, in part (5B on Fig. 74):

a. The need to minimize aerodynamic loading and thermal effects.

b. The need to minimize parachute and structural loading.

c. The desirability of maintaining the simplest (and therefore the most reliable) recovery system.

d. The effects of excess fuel upon the booster dynamics and the possible necessity to dump it.

e. The desirability of having the booster re-enter in a broadside attitude or in a flat spin.

Two possible modes of water entry (6B on Fig. 74) were suggested:

a. Tail first.

b. Broadside.

With both methods, it is envisaged that final deceleration of the booster just prior to impact with the water would be provided by solid propellant rockets.

Tail first water entry has the advantages that it is more convenient from the standpoint of arranging the parachute system, and in placement of the solid propellant retrorockets. The disadvantages are that it subjects the nonstructural shrouding around the motor area to relatively high loads if the velocity of impact is not zero, and even with zero or near-zero impact velocities the upper end of the booster might reach a fairly high velocity in the tipping over process after initial impact.

Broadside entry into the water has the advantages that the problem of tipping the booster over gently after an endwise impact would not be present and that hydrodynamic forces at impact would be distributed over a large area. The latter advantage would be diminished, however, in case of a landing on rough water. The disadvantages of broadside water entry are:

a. Difficulty in suspending the booster from parachutes in a way that would preclude high bending loads upon opening and disreefing of the parachutes. It would probably be necessary to have a rather complicated turnover operation after all parachutes were deployed and disreefed.

b. The structural configuration of the booster is such that with sideways water impact, the retrorockets would have to be grouped appropriately at the two ends. Failure of any one of the rockets would introduce a relatively large moment, tending to tilt the booster out of the horizontal position.

c. The placement of the retrorockets would be unsymmetrical with respect to the booster axis.

Analytical study of water entry has been carried on with an IBM computer and a one-tenth size model and will be discussed in more detail later.

It was expected that the ring-slot parachutes would deflate and settle into the water where they would act as a sea-anchor to stabilize the booster orientation with respect to the wind and sea, and to prevent the booster from drifting (7B on Fig. 74).

Because it was contemplated that the booster can be radar-tracked during practically all of its flight and because it should be a good radar target when floating on the surface, it seems unlikely that any special locating equipment will be required.

B. (S) <u>Trajectory Studies</u>

1. <u>Flight from Separation to Re-entry</u>

Studies were made of re-entry trajectories for both three and six degrees of freedom. A six-degree of freedom study revealed that an axial spin rate of 1 rps (which had previously been decided upon as representing the maximum allowable from structural considerations) would not provide an adequate degree of stiffness. Inspection of the results obtained (Fig. 75) reveals that the angle of attack, within the portion of the trajectory covered by the computation, oscillates between 5 and 70 degrees.

2. <u>Re-entry Dynamics</u>

The studies mentioned in para. 1., above, also show that the angle of attack increases due to the aerodynamic forces in spite of the axial spin. Comparison of six-degree-of-freedom and three-degree-of-freedom trajectories showed that essentially the same behavior results with respect to the angle of attack. The conclusion is that if spin is not imparted to the booster, the three-degree-of-freedom trajectories are adequate.

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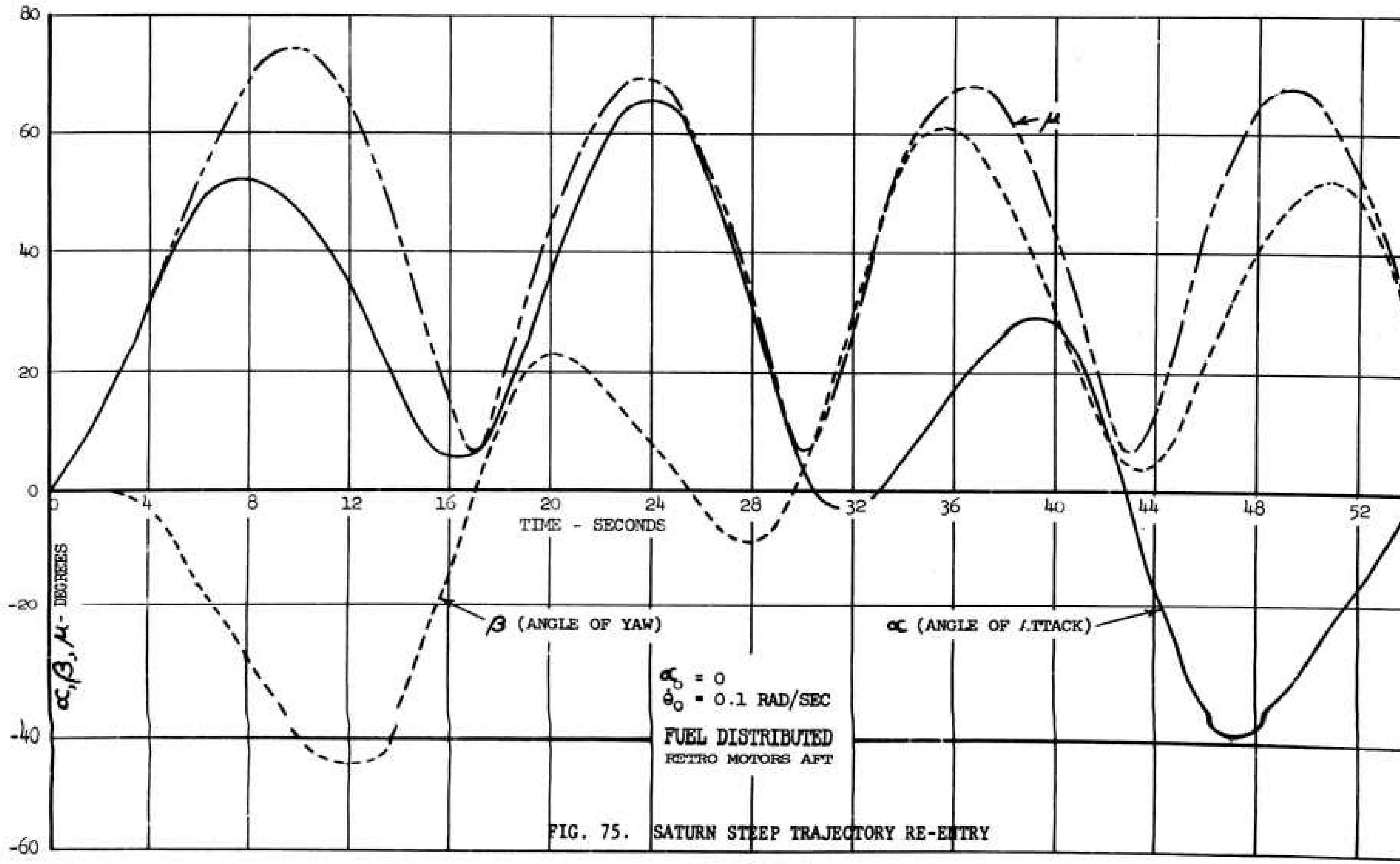

FIG. 75. SATURN STEEP TRAJECTORY RE-ENTRY

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The studies also disclosed that the booster trims to an angle of attack that places the motor end essentially forward regardless of whether the residual fuel is at the forward end of the tanks or at the aft end. Thus, even though the attitude prior to re-entry might be such as to cause the fuel to be in its most forward position, the booster would take up an attitude such as to move it aft. This indicates that a simple representation of the fuel motion is adequate for use in refined trajectory computations for the period during which reorientation is taking place.

The studies thus show that the booster will tend to oscillate about angles of attack in the neighborhood of 180 degrees. Hence, the axial force coefficient at these angles of attack becomes important. The value used in computations was estimated by assuming the value of axial force coefficient at 180 degrees is the same as that at 0 degree. Computations were made for four cases with an increase in axial force coefficient of approximately 15% over that at 0 degree angle of attack.

3. <u>Re-entry Heating</u>

Calculations were made for three re-entry trajectories. Two were tail-first re-entries (shallow and steep) and one broadside re-entry.

Although considerable oscillations would exist, the tail-first trajectory calculations considered zero angle-of-attack flat plate heating. Turbulent boundary layer flow was assumed to be present with a boundary layer run of 7.5 feet. Free-stream properties were used along with the reference temperature method of Eckert.

For the broadside, or high drag re-entry, the heating is of rather low level. Analysis of a 3-foot radius cylinder subjected to stagnation line heating revealed a temperature rise of only 50°R over the assumed re-entry temperature of 600°R (140°F). In this case (apart from the drag effects) angle of attack would reduce the stagnation line heat input. For comparison, temperature increases of 76°R and 118°R were shown for the shallow and steep trajectories, respectively, over the assumed 600°R re-entry temperature. Actual angle-of-attack flow conditions can be expected to increase considerably these heat inputs.

These simply done thin-skin analyses show these trajectories have considerably more drag (and, consequently, less heating) than the earlier ABMA trajectories. No attempt was made to utilize a more detailed convection-conduction network for digital calculations in which such refinements could be included as two-dimensional conduction, inner and outer radiation, inner convection or conduction (e.g., to fuel in the tanks) and angle-of-attack effects on the convective heat input.

4. <u>Aerodynamic Loading During Dive Phase</u>

In the studies in para. 2., above, two trajectories (for three-stage ATLAS version, and for an altitude of 326,614 feet) representing the steepest likely trajectory were compared with two other trajectories (for a two-stage ATLAS version, and for the same altitude) representing the flattest likely trajectory. The comparison showed that the maximum force level is a function of the phasing of angular position and angular rate at the time of the rise in dynamic pressure.

Further study was indicated and was made on the basis of new inertia properties, cutoff conditions, and drag coefficients for the booster at 180-degree angle of attack which resulted from ABMA wind tunnel tests. The new calculations indicated considerably reduced aerodynamic loads as compared with the previous calculations due mainly to the reduced severity of the cutoff conditions. The results of the calculations are shown graphically in Figure 76. This figure indicates that maximum dynamic pressure for an axial re-entry is on the order 1336 psf. This occurs at approximately 58,240 feet altitude and a velocity of 3300 fps. A further reduction in this maximum loading can be anticipated due to booster oscillation about the trim position.

Dynamic effects on the trajectory were investigated by means of three-degree-of-freedom trajectories. The initial conditions for these trajectories were 199,115 feet altitude, 4810 fps velocity, and 29 degrees below horizontal path angle. One calculation was made to represent a severe case in terms of angle-of-attack oscillations during re-entry. For this case assumptions were made for initial angle of attack as zero and pitch rate as 0.1 radian per second. Evaluation of the results showed markedly decreased maximum dynamic pressure over that associated with axial re-entry. The dynamic pressure of 769 psf would occur in this case at about 68,000 feet and at a velocity of about 3300 fps; maximum normal force of 9.9 g's and maximum axial force 3.6 g's. The angle of attack would oscillate through a wide range about 180 degrees. The range of oscillation at the end of the trajectory would be ± 60 degrees.

A second trajectory was calculated to investigate phasing effects on re-entry loading, using the same initial conditions as above, except that no angular rate was assumed and an angle of attack of 1 degree was taken. Again the angle of attack was found to oscillate between wide limits, with the range through the time of maximum dynamic pressure as ± 72 degrees. This wider range of oscillation tends to limit maximum dynamic pressure loading which would be 630 psf at 71,000 feet and a velocity of 3088 fps. Maximum axial loading would be 2.93 g's and maximum normal loading 9.91 g's.

The above trajectory calculations confirmed that the maximum dynamic pressure is associated with the 180-degree angle of attack, i.e., axial re-entry. Dynamic effects associated with rotation to the static position and with the initial separation angular impulse tend to decrease this loading to levels as low as 630 psf.

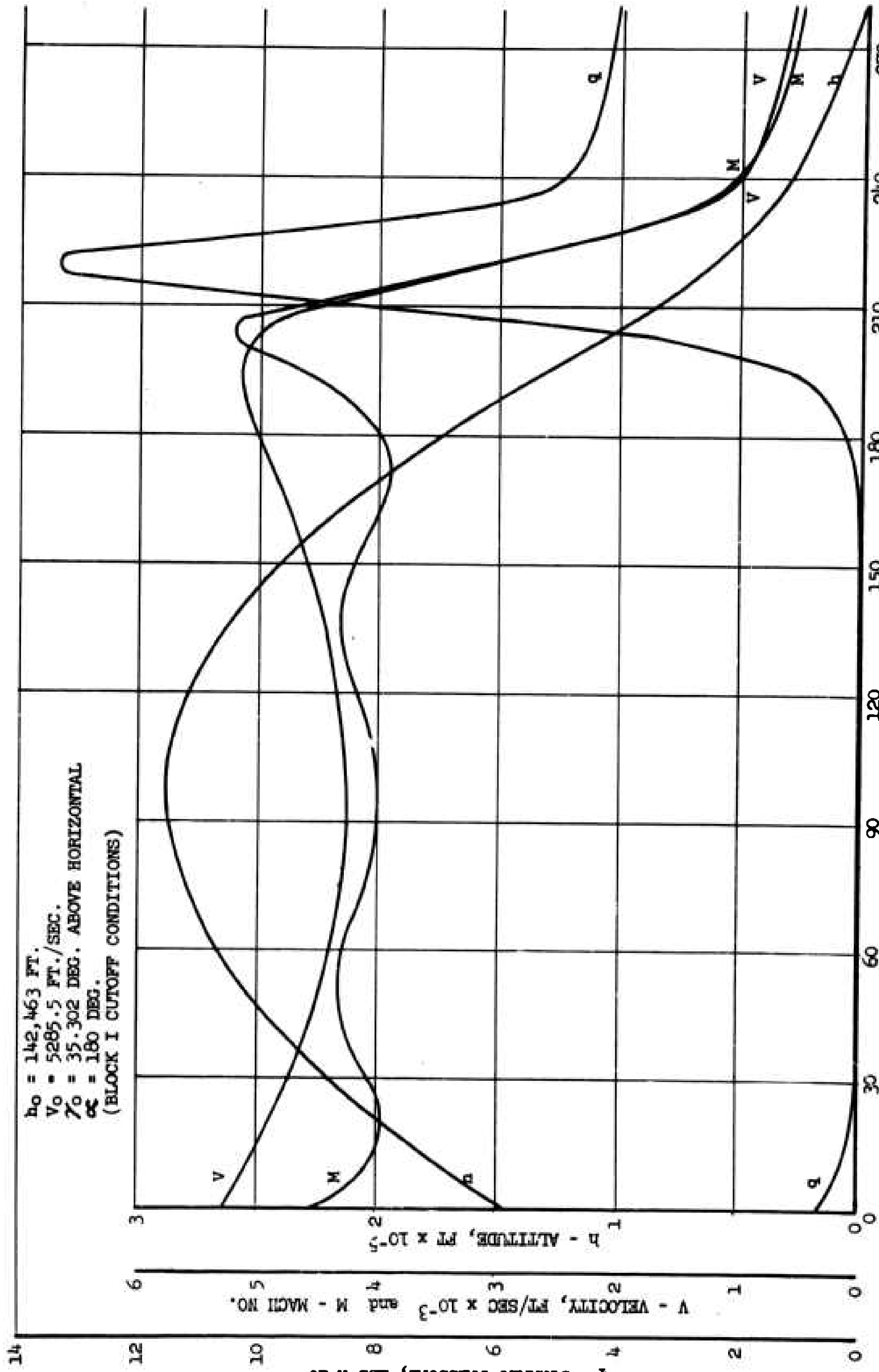

FIG. 76. SATURN BOOSTER TRAJECTORY AFTER CUTOFF

An investigation also has been made to determine if winds will significantly alter the loading and dynamic behavior of the booster during re-entry. The wind profile employed by ABMA in JUPITER dispersion calculations was used along with the initial conditions of zero angle of attack, 0.1 radian per second angular rate, and with fuel and retromotors aft. It was found that negligible effects on loading are produced.

5. Booster Reorientation During Parachute Deployment

Calculations have been made for what are considered to be rather severe, i.e., conservative, conditions of parachute deployment. The conditions are:

q	-- 430
Altitude	-- 21,000 feet
Velocity	-- 853 fps
Booster angle of attack	-- 112°
Parachute reefed area	-- 734 square feet

The calculations indicate that the booster is oriented to 180 degrees approximately 2 seconds after parachute deployment and that maximum load induced on the booster is 225,000 pounds. The peak parachute induced moment about the booster center of gravity (c.g.) is on the order of 8,000,000 ft. lb. Lower loads would be anticipated if the booster could be trimmed around 180 degrees angle of attack, which is possible, or if the reefing area is reduced. Drag coefficients at 180 degrees angle of attack were subsequently included with new inertia properties and cutoff conditions for recalculation of booster trajectories described above.

C. (S) Booster Trimming by Drag Device

A possible means of stabilizing the booster to obtain a broadside re-entry is through use of a drag device. Such a device could be deployed prior to the rise in dynamic pressure along the trajectory and thus bring the booster to a broadside attitude before the period of maximum loading and maximum heating. Calculations were made with different drag area, booster bridle, and riser line parameters but the results of the studies have not warranted adoption of a drag device.

D. (S) Re-entry Using a Lift Device

An investigation has been made of the feasibility of using a lifting type device during re-entry. This would be in the nature of a Flexikite of large area. The study was based on a fixed lift-to-drag ratio throughout the re-entry trajectory; no attempt was made to investigate re-entry with a variable lift device because of the complexity of such an arrangement in practice.

For a shallow re-entry path angle, the scheme may be usable but for the re-entry conditions anticipated the use of the kite lifting device appears to be of questionable value and no further investigation is planned.

E. (S) Deceleration System

1. General

The recovery system presently employs a 57-foot diameter FIST ribbon parachute with one stage of reefing, a cluster of three 108-foot diameter FIST ribbon parachutes, and eight retrorockets.

Early in the report period, wind tunnel results were obtained concerning the stability and drag characteristics of booster configuration BS_5 for high supersonic Mach numbers. (The designation BS_5 refers to the configuration which has the recovery package at the nose but has no shroud.) From the data, it was found that the movement of the center of pressure with angle of attack was such that there was a possibility of the booster trimming out at a relatively low angle of attack. Since this condition is very undesirable from the standpoint of recovery operations, it was decided that an estimate of the booster characteristics at the approximate recovery Mach number of 0.80 should be made. From these results, the expected attitude of the booster at that time could be determined. The results showed that booster configuration BS_5 would tend to trim out at angles of attack near 90 degrees and would therefore be in a favorable attitude for recovery initiation.

2. Parachute System

At the close of the previous report period, a revised estimate of booster drag variation with Mach number had necessitated a new point-mass trajectory which altered the initial conditions of recovery initiation. Increased re-entry velocity, dynamic pressure, Mach number, etc., had led to initiating recovery by parachute at 21,381 feet and a velocity of 853 fps. At these conditions the "q" is 439 psf. When subjecting the booster to a maximum deceleration of 2.7 g's, the corresponding drag areas were 734 sq. ft. and 1405 sq. ft. for decelerating the booster to a velocity of 342 fps at an altitude of 9880 feet. A 57-foot diameter FIST ribbon parachute with one stage of reefing was selected for initial deceleration.

Further deceleration will be accomplished by increasing the drag area in three steps. A total effective drag area of 12,360 sq. ft. is required to yield a sea level rate of descent of 90 fps. The two reefing stages were necessary in order to hold the g level of deceleration to 2.7. A cluster efficiency of 90% was assumed for the fully opened parachutes only.

An estimated force profile in terms of descent time is given in Figure 77, which shows that the maximum acceleration does not exceed 2.6 for the first four stages. A summary of the drag areas, altitudes,

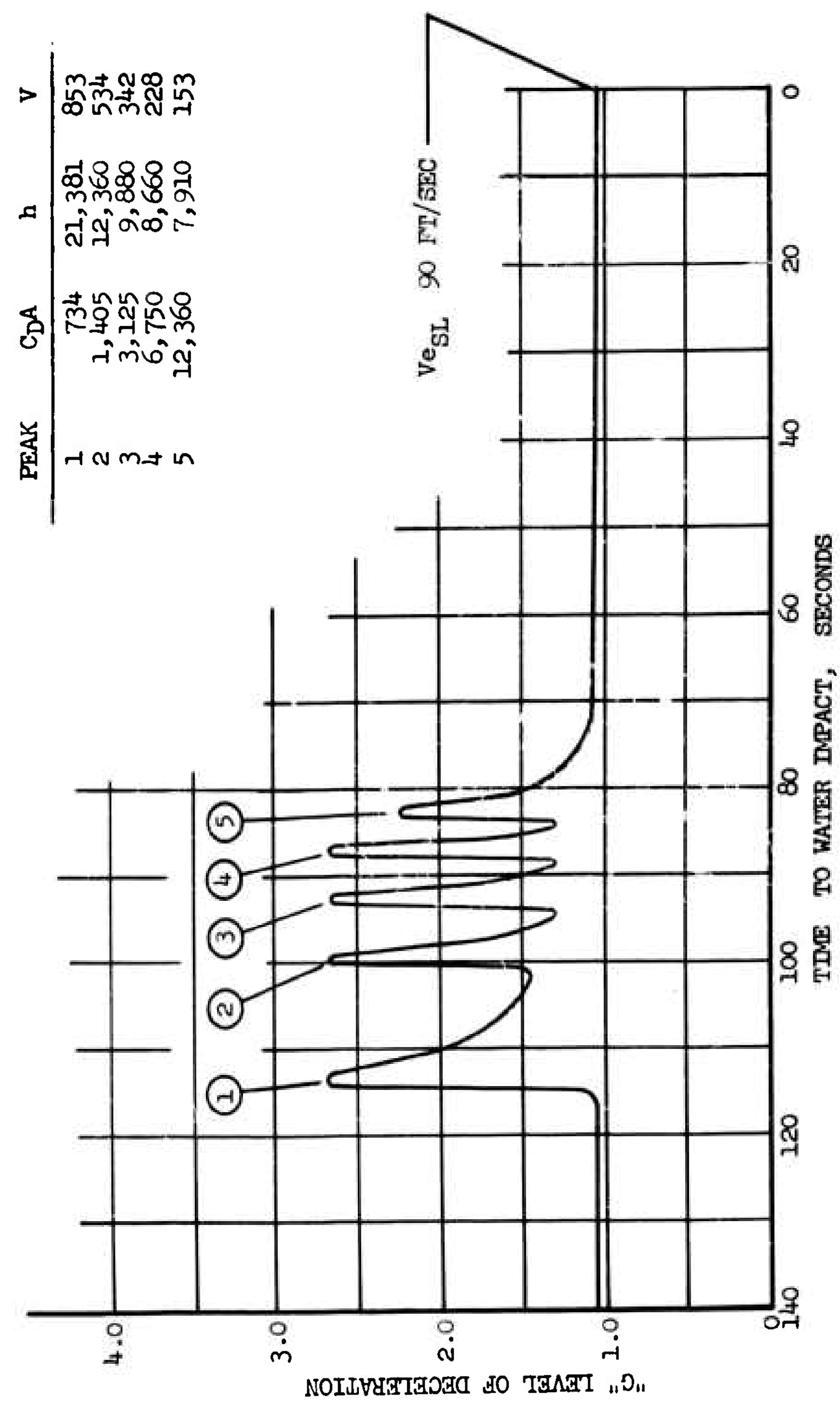

FIG. 77. SATURN BOOSTER DECELERATION LEVEL DURING PARACHUTE RECOVERY

velocities, dynamic pressures, and times required to decelerate is given in Table 8.

At the date of this report the complete parachute design specifications have been generated and work on gore patterns is underway. No delay is anticipated for materials but an order for one type of nylon ribbon recently developed for WADC has been placed since this item is presently available in limited quantities. Assembly will utilize conventional fabrication techniques whenever possible.

Since these parachutes are relatively large in size, the deployment bag will simply be a liner for the parachute container. The parachutes will be packed directly into their respective containers thereby deleting the intermediate step of packing into a deployment bag.

3. <u>Retrorocket System</u>

Choice of a particular rocket motor or motors to be used on the SATURN booster recovery system was influenced by many factors.

Preliminary studies showed that the weight of the over-all recovery system consisting of parachutes and retrorockets tended to be a minimum when there were sufficient parachute areas to give a velocity between 70 and 110 fps at the time of retrorocket initiation.

It was desirable to have the second stage parachute diameter no greater than about 108 feet in order to stay within the established state of the art. To avoid excessive complexity of the parachute system, the cluster should have no more than three chutes.

With these conditions, it appeared feasible to initiate the retrorockets at a velocity of about 90 fps.

The original study showed the retrorocket weight is at a minimum if the total thrust of the rockets is greater than about twice the recovery weight of the booster. Little is gained by making the thrust more than three or four times the booster weight. Since higher thrust rocket systems require shorter burning times, it was decided to consider total thrusts in the range of two to three times the booster weight. This called for burning times of 1.5 to 2.5 seconds. The timing of retrorocket initiation is not so critical in this range as for short duration rockets. Furthermore, there were several possible rockets available with such burning times.

Another consideration in the selection was geometry of the booster structure because the thrust of the rockets must be taken at strong points of this structure. To avoid difficulties due to impingement of the rocket blast on the tank walls and to keep the center of gravity of the booster aft, it was preferable to mount the retrorockets on the lower thrust frames of the booster rather than on the top structure. Thus, the

Table [illegible]

TRAJECTORY CONDITIONS

PARACHUTE

PARACHUTE NOMENCLATURE	PARACHUTE OPERATION	DRAG AREA		ALTITUDE (ft)	VELOCITY (ft/sec)	q (lb/ft²)	g LEVEL (max)	TIME DELAY (sec)
		NUMBER	C_DS (ft²)					
FIRST STAGE:								
57-foot [illegible] diameter Rat [illegible] Ribbon [illegible] with [illegible] stage of [illegible]	Deployment	---	---	21,38[illegible]	[illegible]	439	---	---
	Deceleration	1	734	---	---	---	2.6	12.4
	Disreef	---	---	1[illegible],180	534	226	---	---
	Deceleration	[illegible]	1,405	---	---	---	2.6	8.7
MAIN CLUSTER:								
Three 108 foot diameter FIST Ribbon Parachutes with [illegible] stages of reefing	Deployment	---	---	9,600	150	109	---	---
	Deceleration	[illegible]	2,390	---	---	---	2.6	7.9
	Disreef No. 1	---	---	7,620	---	47.7	---	---
	Deceleration	4	6,690	---	---	---	2.6	3.4
	Disreef No. 2	---	---	6,870	150	21.8	---	---
	Deceleration	5	12,360	---	---	---	2.25	4.9
	Terminal	---	---	6,320	101	---	---	---

number of rocket motors was established as being such that a symmetrical distribution can be mounted on the eight frames of the booster. Therefore, the number of rockets would be some multiple of two between two and 16, assuming that no more than two motors would be mounted on any one frame.

It was preferable to tilt the rockets in such a way that their thrust axes pass through the c.g. of the booster in the condition in which it hangs on the parachutes, thus limiting any overturning moment that would exist if one rocket failed to fire. Unfortunately this was not feasible because of the large tilt angle required. As a compromise, a tilt angle of 18 degrees was decided upon, subject to change to adjust the vertical component of thrust, or to avoid possible impingement of the rocket blast on the lower shroud.

Economics and time dictated that existing rocket motors be used. The use of thoroughly tested motors also increases the reliability factor. The decision was to use eight Aerojet-General 2KS-36250 rockets to decelerate the booster from 90 fps to zero fps at water entry.

Because the booster weight can be expected to be approximate until nearly time of the first launching, studies were made to determine which combination of the rocket motors could be used to decelerate booster weights of from 110,000 to 130,000 pounds to zero velocity. It was found that the eight 2KS-36250 motors ignited in a six-rockets-then-two combination would better decelerate the heavier booster and a four-rockets-then-four combination would be more satisfactory for the lighter weight.

A study was also made of the errors that can be encountered during the retrorocket trajectory:

a. Error in drop line due to pendulum type motion or waves causing an error in initiation altitude.

b. Error in timing mechanism which would cause an error in the firing time of the second set of rockets.

c. Error in expected weight of booster due to excess fuel which would cause an error in terminal velocity, height of initiation, and required time delay.

d. Thrust error.

The analyses also included the error encountered in the velocity at water impact for various rocket combinations and different weights. It also includes an investigation of obtaining zero velocity at burnout at other than the water level. Indications were that for lighter booster weights, a four-four rocket combination with a desired zero velocity approximately two feet below the water would render smallest water level impact velocity for the expected error values in time delay and altitude of initiation. The best conditions for the six-two system were found to be hitting the water at the burnout of the first set of six rockets.

At this time the retrorocket error study is nearly completed and the six-two stagger-firing sequence appears to be the most desirable.

F. (S) <u>Design of Recovery Package Structure</u>

A comparison study of conical, beam and cylindrical load-carrying structures established that the conical configuration had the most desirable features, namely, maximum strength to weight ratio, most economical utilization of space, and uniformity of load distribution.

A design study was carried out to evaluate methods of effecting first stage parachute separation and main stage parachute deployment. Considerations in the study were:

a. Use of vee-band clamps, frangible bands, or explosive bolts as separation device.

b. Design of a separating device with a view toward probable reuse of recovery structure.

Various methods of first stage parachute ejection studied were:

a. Single catapult at bottom of first stage parachute container.

b. Four catapults equally spaced about circumference of first stage parachute container.

c. A long single catapult with first stage parachute container axis parallel to the catapult axes.

d. Mortar-type ejection from parachute container.

Until recently it had been planned to deploy the 57-foot first stage parachute by ejecting it from the vehicle by means of a Frankford Arsenal catapult system. In order to utilize existing equipment and avoid the necessity of new development, it was proposed that four standard seat ejection catapults be arranged so they would be initiated from a common gas initiator and would act in unison to propel an inverted metal can, containing the parachute, from the vehicle.

Late in the report period an alternate deployment scheme was reconsidered. This alternate scheme calls for the use of an oversized pilot parachute which would be ejected by a mortar. Initial estimates indicate that this would be a FIST ribbon parachute about 12-14 feet in diameter. The riser lines will remain attached to the recovery package until the pilot parachute has deployed and oriented the booster into an attitude near 180-degree angle of attack so that the first stage parachute will not be dragged over any of the structure which surrounds the recovery package. This consideration determines the parachute that is required. After a suitable time interval for the reorientation process, the harness by which the pilot parachute is attached to the structure will be cut so the pilot parachute will leave, pulling out the first stage parachute.

Because the pilot parachute is small in comparison with the first stage parachute, an adequately strong mortar can be used without paying a significant weight penalty. For this reason and because of the greater simplicity of hardware involved in the mortar system, the plan for mortar ejection of the pilot parachute has been chosen.

The reasons the design was reconsidered and changed are:

a. The simultaneous employment of several catapults to launch a heavy load had not, so far as is known, been attempted previously.

b. Tests to establish the satisfactory operation of the multiple catapult system would be expensive and time consuming.

c. If the tests that were planned had revealed difficulties, changing the system or carrying on further developmental tests would have caused a serious disruption of the project time schedule.

d. The alternate system, now planned, makes possible a significant simplification of the recovery package structure with a corresponding saving in the cost of fabricating the delivery hardware.

G. (S) Water Entry

Water entry trajectories have been computed for four cases. The initial conditions for these calculations and the pertinent results are summarized in Table 9.

The data indicate that maximum booster submergence is on the order of 16 feet and is relatively unaffected by wind and vertical velocity at entry. The time to tip to horizontal varies inversely with the wind velocity. Maximum side loading occurs as the booster is in a near horizontal attitude with aft end partly submerged. The lowest point of the nozzle ranges from 3.75 to 6.25 feet submerged. The side loads on the booster can be as high as 570,000 pounds. This may prove to be a serious loading condition since the load must be borne by the lower-most tanks on the outside of the booster.

A one-tenth scale (by dimension) model (Fig. 78), scaled according to Froude Number ($N_f = L_g/v^2$) was built with a weight:

$$\frac{111{,}700}{1/\lambda^3 = 1000} = 111.7 \text{ pounds}$$

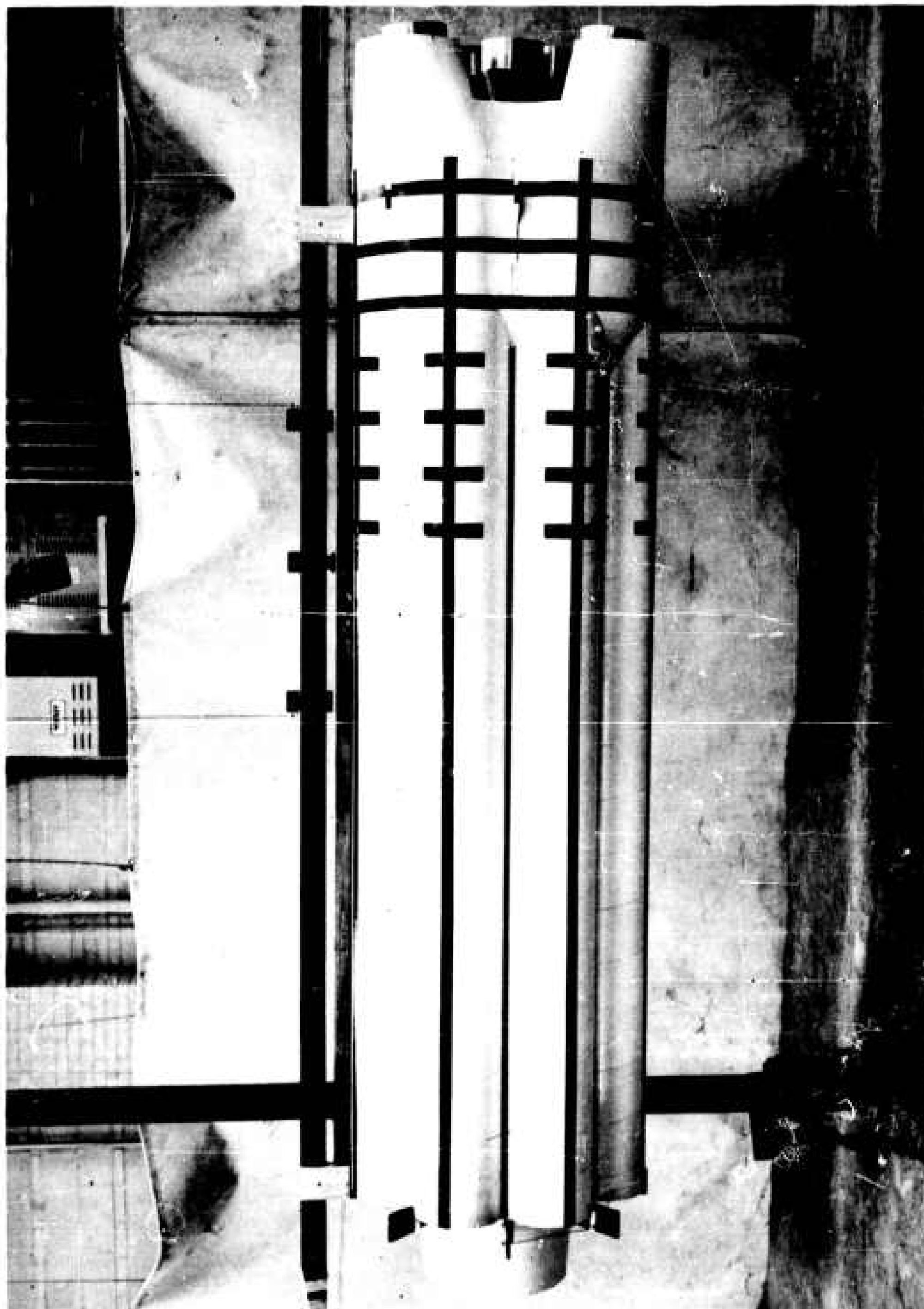

FIG. 78. ONE-TENTH SCALE SATURN BOOSTER WATER ENTRY MODEL - SIDE VIEW

Table 9

SUMMARY OF WATER ENTRY PARAMETERS

	CASE I	CASE II	CASE III	CASE IV
Booster vertical velocity at water contact (fps)	17.5	17.5	17.5	9.4
Horizontal wind velocity (fps)	25	12.5	0	25
Booster angular rotation rate at water contact (deg/sec)	0	0	-0.573	0
Maximum immersion of lowest point (ft)	16.6	16.0	15.6	16.25
Maximum booster angular velocity (deg/sec)	-39	-55	-41	-41
Time from entry to tipover (sec)	3.9	4.9	6.9	3.85
Maximum applied parachute force (lb)	28,000	54,000	53,500	35,500
Maximum applied parachute moment (ft-lb)	1,370,000	2,568,000	2,480,000	1,690,000
Maximum total load on bottom (lb)	196,000	197,500	198,000	185,000
Maximum total load on a side (lb)	569,269	401,326	450,328	460,128
Depth of booster when horizontal	6.25	3.75	4.6	5.8

and a pitch moment of inertia of

$$\frac{2,430,238}{1/\lambda^5 = 100,000} = 24.3 \text{ slug ft}^2$$

The model was instrumented and used to supplement computer studies in determining the forces which result from water impact and tipover. Accelerometers were mounted at the center of gravity to measure lateral and axial "g" forces. Accelerometers mounted on top of the booster measured lateral "g" forces during tipover. Strain gages mounted on two of the model rocket engines measured water impact axial forces and moments.

Thirty-one drops of the scale model were made into the water. Figures 79 and 80 show the model in floating position. The data are being analyzed toward the end result of providing drag and buoyant force coefficients.

H. (U) Booster Recovery after Water Entry

The plan for booster retrieval as described and illustrated in the previous semi-annual report has remained virtually unchanged. Details have been worked out during the period of this report.

Retrieval of the SATURN booster from the ocean is to be accomplished by the use of a fleet of surface vessels including an LSD (Landing Ship Dock), and escorting boats and aircraft.

The recovery of the booster after it enters the water will consist of four phases:

a. Location and damage surveillance.

b. Recovery of the booster from the ocean.

c. Decontamination and preservation.

d. Return shipment of the recovered components.

Immediately after impact, the aircraft and boats will seek out the booster and keep it under surveillance until the LSD arrives at the impact site.

With the aft deck of the LSD awash, the booster will be floated into position on fixed supports in the LSD well. The well will then be pumped dry, leaving the booster in a supported position for decontamination and preservation.

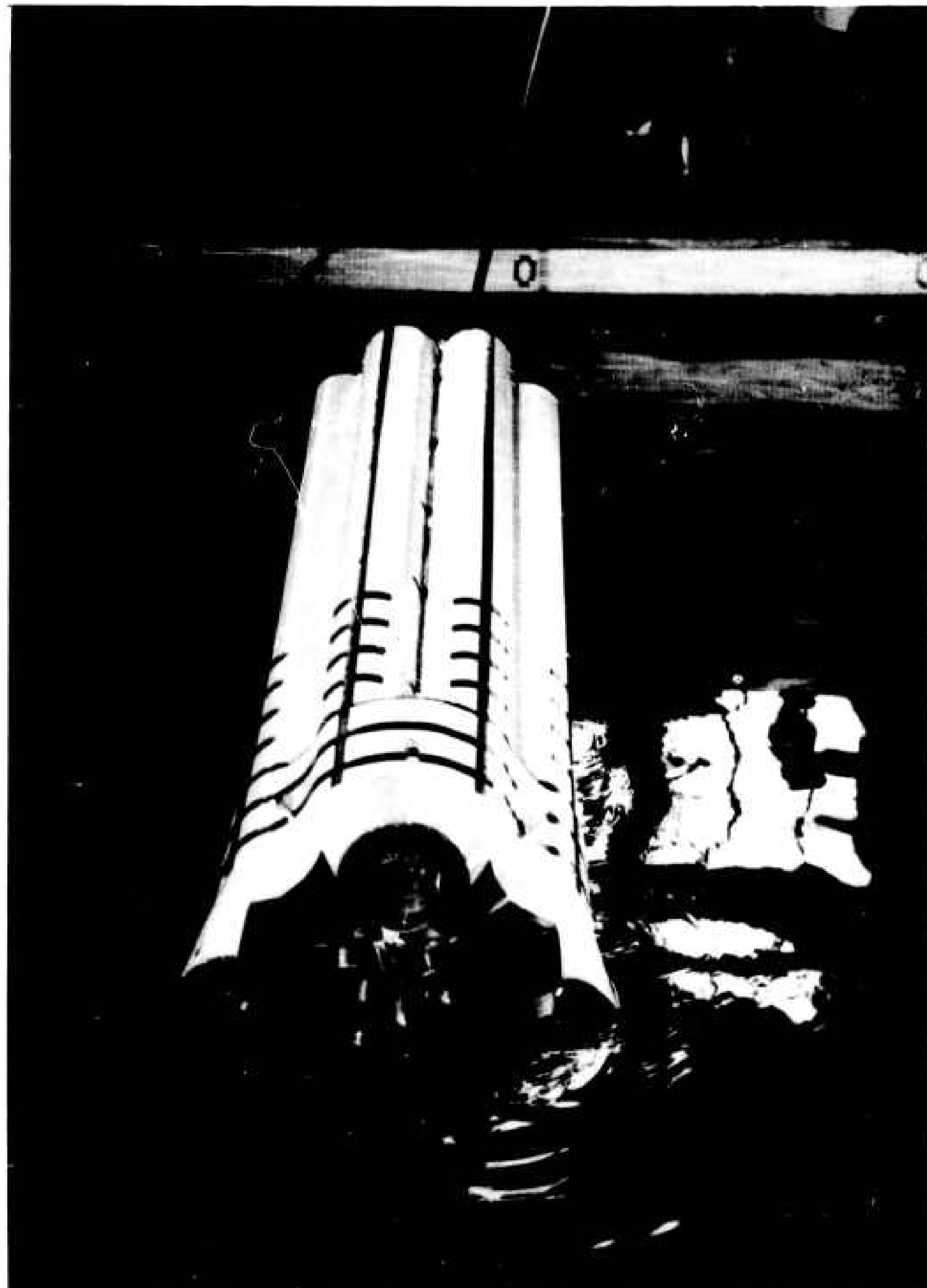

FIG. 79. ONE-TENTH SCALE MODEL IN THE WATER - END VIEW

FIG. 80. ONE-TENTH SCALE MODEL IN THE WATER—SIDE VIEW

Decontamination will consist of a dry air or LN_2 purge of the lox system, an over-all washdown with hot fresh water, and disassembly and cleaning of initial and special components. After decontamination the booster and components will be thoroughly dried by purging with hot dry air and preserved by the application of desiccator breather assemblies.

The LSD will then return the booster to the appropriate dock. There, depending on the damage conditions, the booster will be unloaded onto a transporter or onto the supports that were used in the LSD and will then be reloaded into a barge for return to Huntsville, Alabama. Upon arrival at Huntsville it will be unloaded by roll-off onto a transporter, or if too heavily damaged, by disassembly in the barge and lifting out the parts with a 20-ton crane.

The booster recovery operational concept set forth will require the following handling gear and associated equipment:

a. Tow lines and yokes for towing the recovered booster into the well of the LSD.

b. Booster saddles for supporting the booster in the well of the LSD.

c. Decontaminating and salt water cleaning equipment for cleaning the booster in the well of the LSD.

d. Preservation and packaging equipment to protect the cleaned booster and salvaged items.

e. One set of hoisting gear for lift. This hoisting gear is not to be confused with the erection gear. Its only function is to effect horizontal lift of the SATURN booster out of the LSD, with or without saddles as the case may be, and placing it into the barge for return to Huntsville.

All of the above recovery gear will be shipped on the barge from Huntsville, Ala. with the booster and removed to the LSD prior to discharging of the barge at Fort Pierce, Fla. (See Report DIR-TR-2-59 for the previous report period for the booster transportation plan.) The items will remain on the LSD for use during the recovery of the same booster.

SECTION VI. SPECIAL STUDIES

A. (U) Introduction

From the time of initiation of the SATURN program until a recent orientation of missions by the NASA, one of the main missions envisioned for the vehicle was that of placing a 24-hour satellite into orbit around the earth. In connection with this assigned mission and other basic studies being accomplished at that time by ABMA, considerable effort was expended to advance the art of guidance theory and control techniques. Also, optimization of flight mechanics to achieve a precisely controlled 24-hour orbit was studied. To reflect the state-of-the-art achieved in these developments, brief descriptions of the work are included in the following pages. Also, the general results of a noise radiation study accomplished by ABMA to determine effects from high thrust clustered engines are discussed.

B. (C) 24-Hour Orbit Trajectory

1. Introduction

To accomplish transfer to the 24-hour orbit, an ascent trajectory was chosen which would carry out most of the maneuvers at a high altitude. Also, the apogee was placed at a node to obtain maximum performance. This can be done independently of the launch azimuth so that a 90° due east value could be assumed in order to derive full benefit from the earth's rotation and smallest "dog leg" angle required for equatorialization. A second-stage cutoff altitude of 150 km seemed to be on the borderline for requiring special heat protection. Also, since engine-out capability would become difficult to retain as the trajectory is flattened, the cutoff altitude was restricted to 150 km. The trajectory was then optimized under the above restrictions. A path angle in the neighborhood of 59° against the local vertical at cutoff of the first stage was determined to be optimum (Fig. 81).

2. Injection Phase

a. Introduction

The maneuvers beginning with the injection into the geo-stationary orbit were grouped into a single phase for convenience referred to here as the "injection phase." Subsequent maneuvers for fine corrections are then thought of as belonging to a separate phase which may be called a "vernier phase."

The scheme described is exemplified for injection into a geo-stationary orbit of longitude 45° west. Subsequent alteration of this position to 38° west offers no real problem, since the injection data differ only slightly from those given here.

b. Parking Orbit

Preliminary to the injection into the geo-stationary orbit, the vehicle is injected into a 200-km altitude circular parking orbit. For obtaining this parking orbit, a due east launching azimuth is employed in order to minimize the inclination of the parking orbit

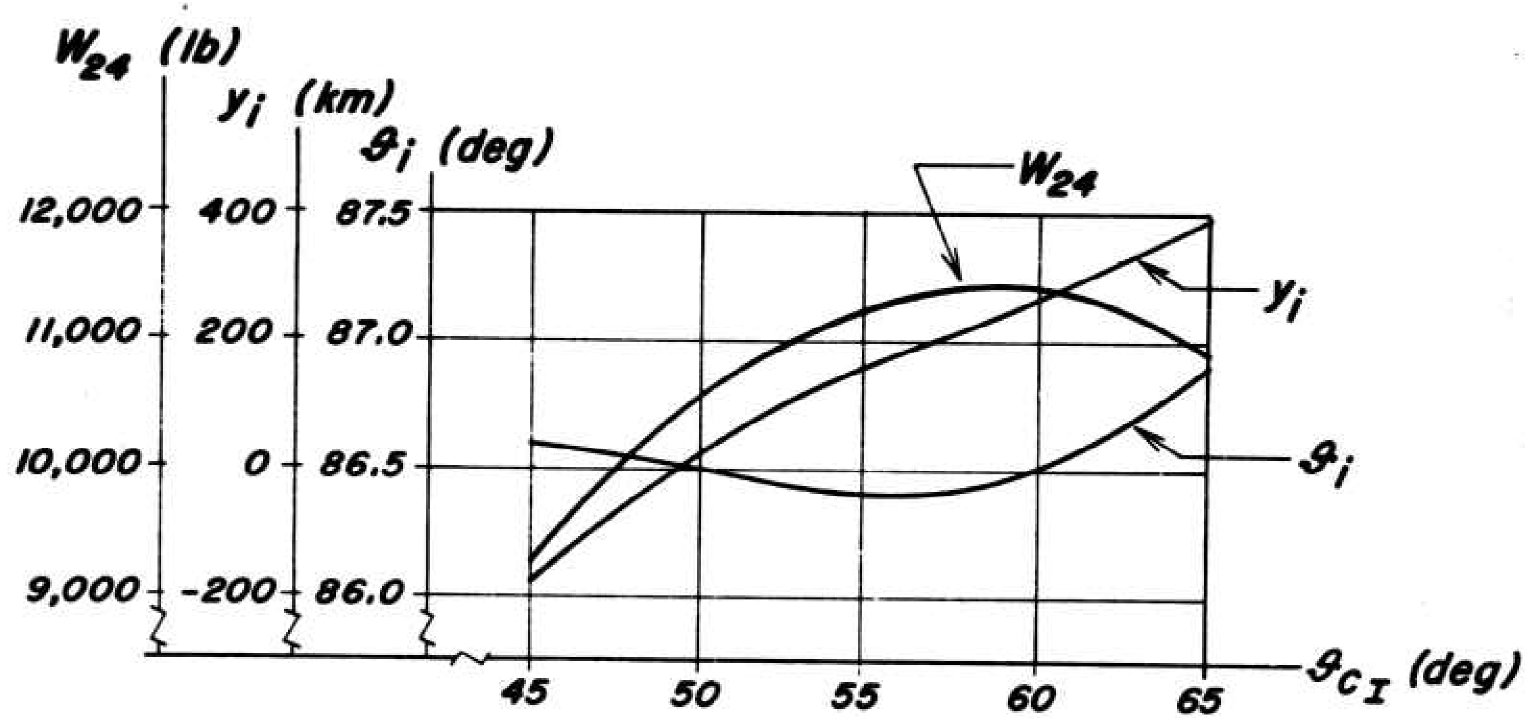

Fig. 81 WEIGHT IN FINAL ORBIT (W_{24}) TRANSFER ELLIPSE INJECTION ALTITUDE (y_i) AND PATH ANGLE (ϑ_i) FOR OPTIMALLY SHAPED TRAJECTORIES VERSUS EARTH FIXED FIRST STAGE CUTOFF PATH ANGLE (ϑ_{c_I}) (LOW ALTITUDE CIRCULAR COASTING)

and thereby the angle which is eventually encountered for inclination change to effect an equatorial orbit (dog-legging), and to make full benefit of the contribution of the earth's rotational velocity. Optimization of these phenomena is very significant since launching azimuths deviating from due east by 10°, 30°, and 50°, e.g., give rise to respective payload deficits of the order of 100 lb, 650 lb, and over 1600 lb.

Departure from the parking orbit always occurs at either the first or the second nodal equatorial crossing. Departure is effected by imparting to the vehicle a velocity increment which places the vehicle into a Hohmann ellipse terminating at its apogee at the altitude of the geo-stationary orbit. Departure is restricted to the first two nodes in order to minimize the time spent in the parking orbit, minimizing the power requirements for the guidance, any existing drift of the platform, and build-up of existing coordinate errors during transit of the Hohmann ellipse.

c. High Altitude Loop

In order to resolve the problems of launching azimuth and time requirements, it has been stipulated that the injection scheme employ a due east launching azimuth and use only the first or second node of the parking orbit as the departure node. This leads to only two possible injection longitudes for the geo-stationary orbit; one results from departure at the first node and one from departure from the second node. These two points are at the approximate longitudes of 105° east for the first node departure and 85° west for the second node departure. Should these longitudes correspond to those desired, an appropriate velocity increment imparted at the apogee of the Hohmann ellipse would place the vehicle into the desired geo-stationary orbit. (It is at this point that dog-legging occurs, with the Hohmann ellipse having been in the same 28.3° inclined space-fixed plane as the parking orbit.)

If the desired longitude is not one of these two, the following scheme may be used. Instead of imparting the entire velocity increment at the apogee of the Hohmann ellipse, some fraction of it may be given instead. The resultant orbit will be an ellipse with the same apogee as that of the Hohmann ellipse but a higher perigee. As the imparted velocity increment varies from zero to the full requirement for obtaining circular velocity for the geo-stationary orbit, the perigee altitude will vary from the altitude of the parking orbit to the altitude of the geo-stationary orbit. Simultaneously, the period of the orbit will vary between these two extremes. If this period is adjusted correctly, then after one or more complete revolutions in the ellipse, longitudinal adjustment (phasing) will be effected. After this, the remainder of the velocity increment may be imparted to the vehicle, placing it into the geo-stationary orbit.

The intermediate ellipse just discussed may be referred to as a "loop" and the loops all belong to a one-parameter family of ellipses with a common axis and apogee point, but varying perigee. The injection scheme considered as a whole will be referred to as "high altitude looping" and is illustrated in Figure 82.

d. Analysis of the High Altitude Loop

Numerical analysis of the looping is presented in Figure 83. The time for loops is shown as a function of the desired longitude for different values of ℓ. The period of the individual loop is then one part in ℓ of the total time. Since the period depends on the desired longitude, the apogee velocity of the loop is shown in Figure 84 as a function of longitude for different ℓ.

Two properties of the high altitude loop were determined. First, the period of the loop differs less and less from one sidereal day as ℓ increases. In the limit, for infinitely large ℓ, the period is exactly one sidereal day. Thus, as ℓ increases, the loop becomes more nearly the geo-stationary circle itself and the drift of the vehicle becomes more gradual towards the desired longitude.

Secondly, for increasing ℓ the optimum inclination tends to zero (equatorial orbits).

If one assumes a limitation of 100 m/s on the second velocity increment in the loop, which is to be imparted by the satellite nozzles, the situation in Figure 85 results. The figure shows all longitudes may be obtained with no more than seven revolutions in the high altitude loop (for the most unfavorable longitude).

By restricting v_2 (the second velocity increment) to less than or equal to 100 m/s, the optimum inclination is kept to less than about two degrees. For this reason, the high altitude looping can be equatorial, which would enable all of the dog-legging to be accomplished by the third stage upon injection into the high altitude loop.

Figure 86 shows Mercator projection (onto the surface of a rotating earth) of the maneuvers from launch to the final injection into the geo-stationary orbit at a longitude of 45° west.

3. Vernier Phase

A satellite in a geo-stationary orbit is in a circular equatorial orbit, the period of which is one sidereal day, so that to an observer on the earth the satellite would appear stationary. If the position of the satellite is assumed to be preselected (with respect to longitude), for a satellite to remain in a geo-stationary orbit, it must possess specific values for both its positional and velocity coordinates. If errors occur in any of these coordinates the satellite will be seen to oscillate and in some cases to drift away from the desired point.

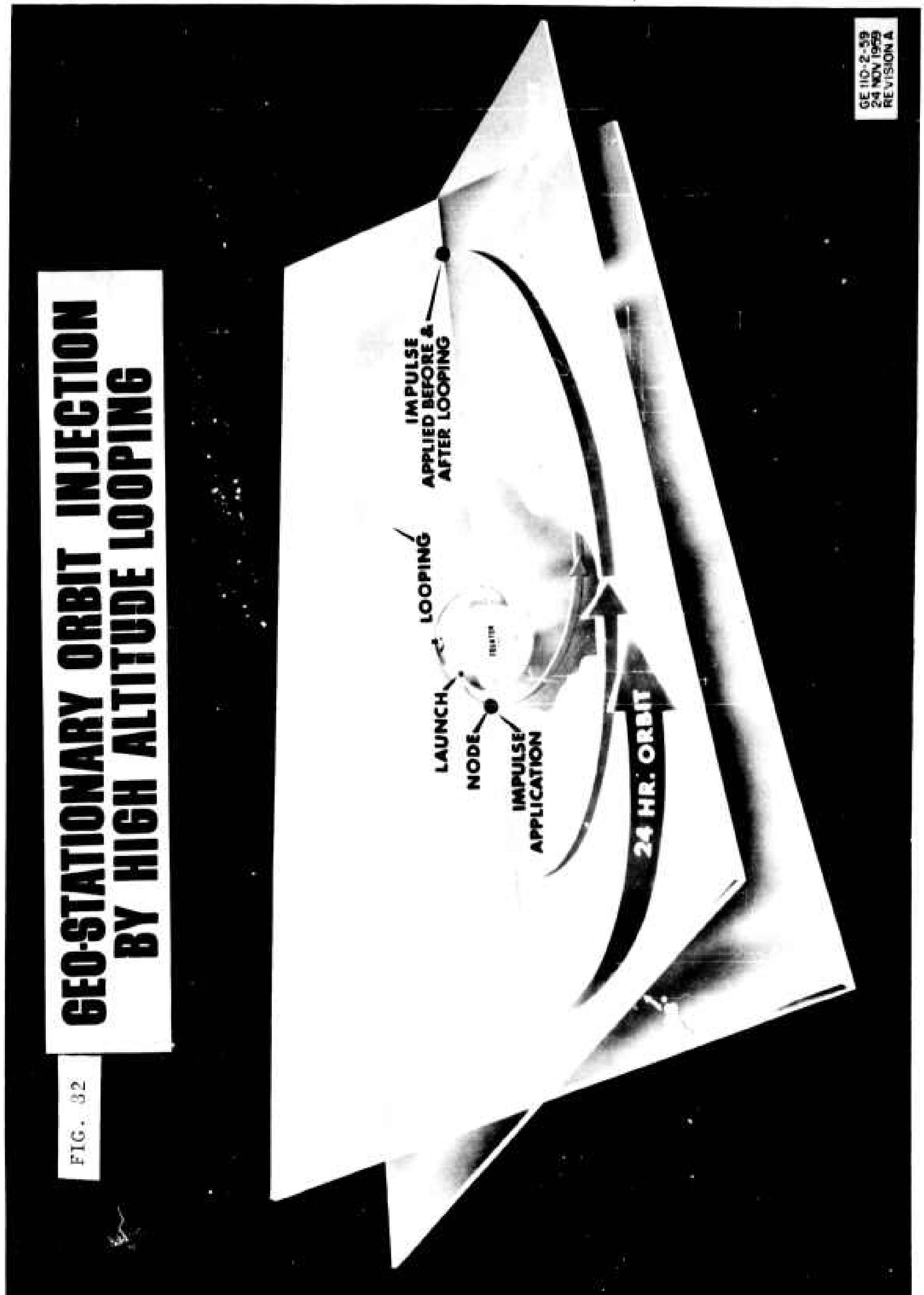
GEO-STATIONARY ORBIT INJECTION
BY HIGH ALTITUDE LOOPING
IMPULSE
APPLIED BEFORE &
AFTER LOOPING
LOOPING
LAUNCH
NODE
IMPULSE
APPLICATION
24 HR. ORBIT
GE 110-2-59
24 NOV 1959
REVISION A

FIG. 32

CONFIDENTIAL

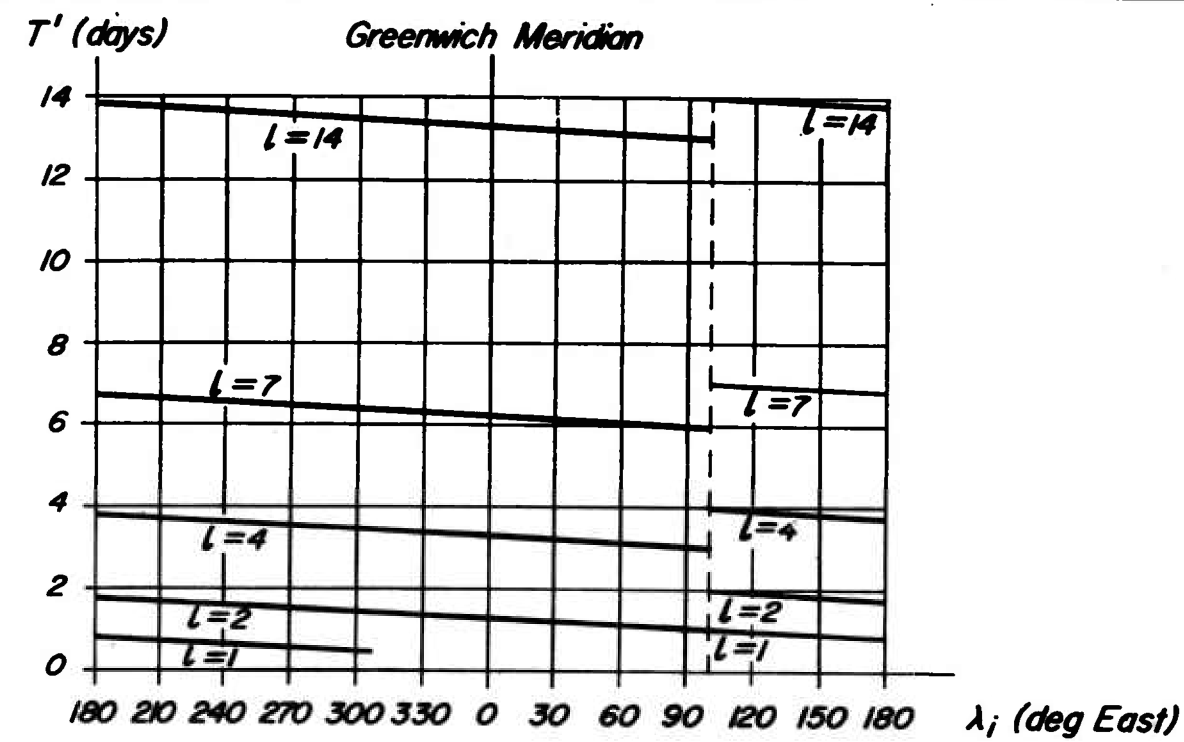

Fig. 83 TIME (T') SPENT IN HIGH-ALTITUDE LOOPING AS A FUNCTION OF THE DESIRED INJECTION LONGITUDE (λ_i) FOR DIFFERENT NUMBERS OF REVOLUTIONS IN LOOP (l) FOR DEPARTURE FROM PARKING ORBIT AT FIRST NODE

CONFIDENTIAL

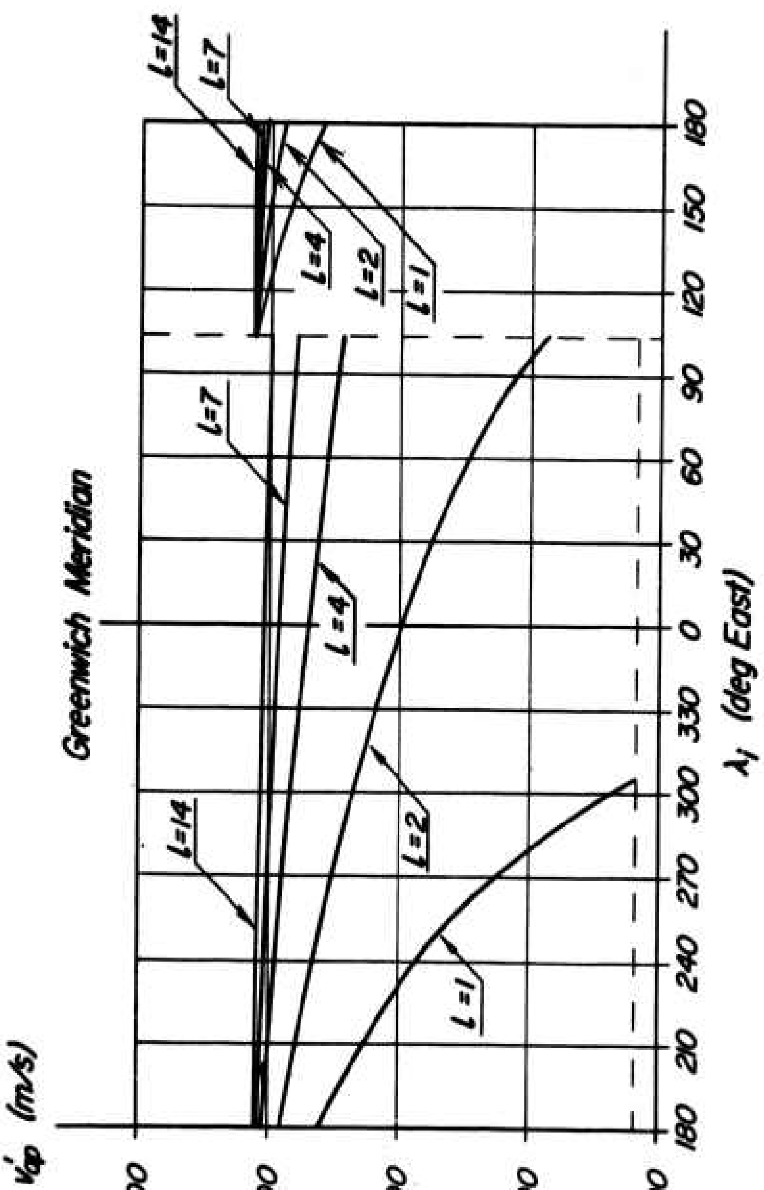

Fig. 84 APOGEE VELOCITY OF LOOPING ELLIPSE (v'_{ap}) AS A FUNCTION OF DESIRED INJECTION LONGITUDE (λ_i) FOR DIFFERENT NUMBERS OF REVOLUTIONS IN LOOP (l) FOR DEPARTURE FROM PARKING ORBIT AT FIRST NODE

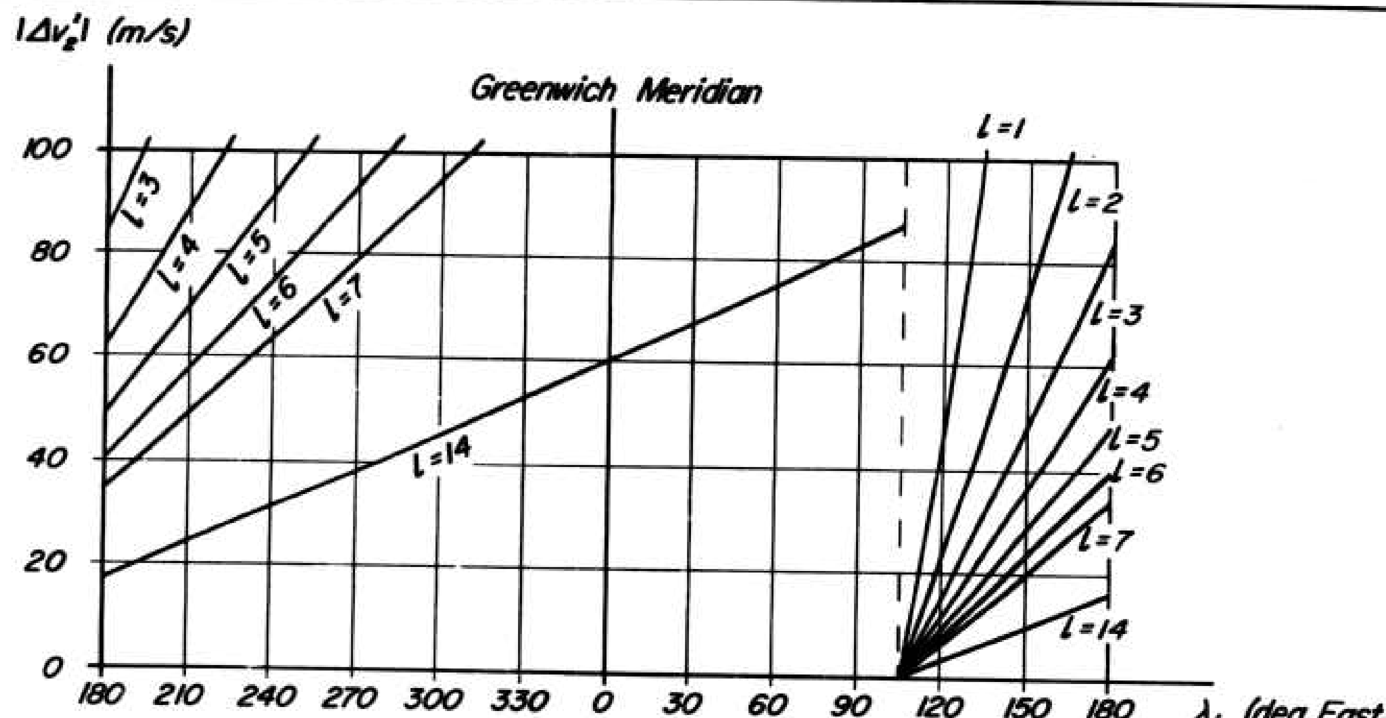

Fig. 85 MAGNITUDE OF SECOND APOGEE VELOCITY INCREMENT ($|\Delta v_2'|$) FOR HIGH-ALTITUDE LOOPING AS A FUNCTION OF DESIRED INJECTION LONGITUDE (λ_i) FOR DIFFERENT NUMBERS OF REVOLUTIONS IN LOOP (l) FOR DEPARTURE FROM PARKING ORBIT AT FIRST NODE FOR $|\Delta v_2'| \leq 100$ m/s

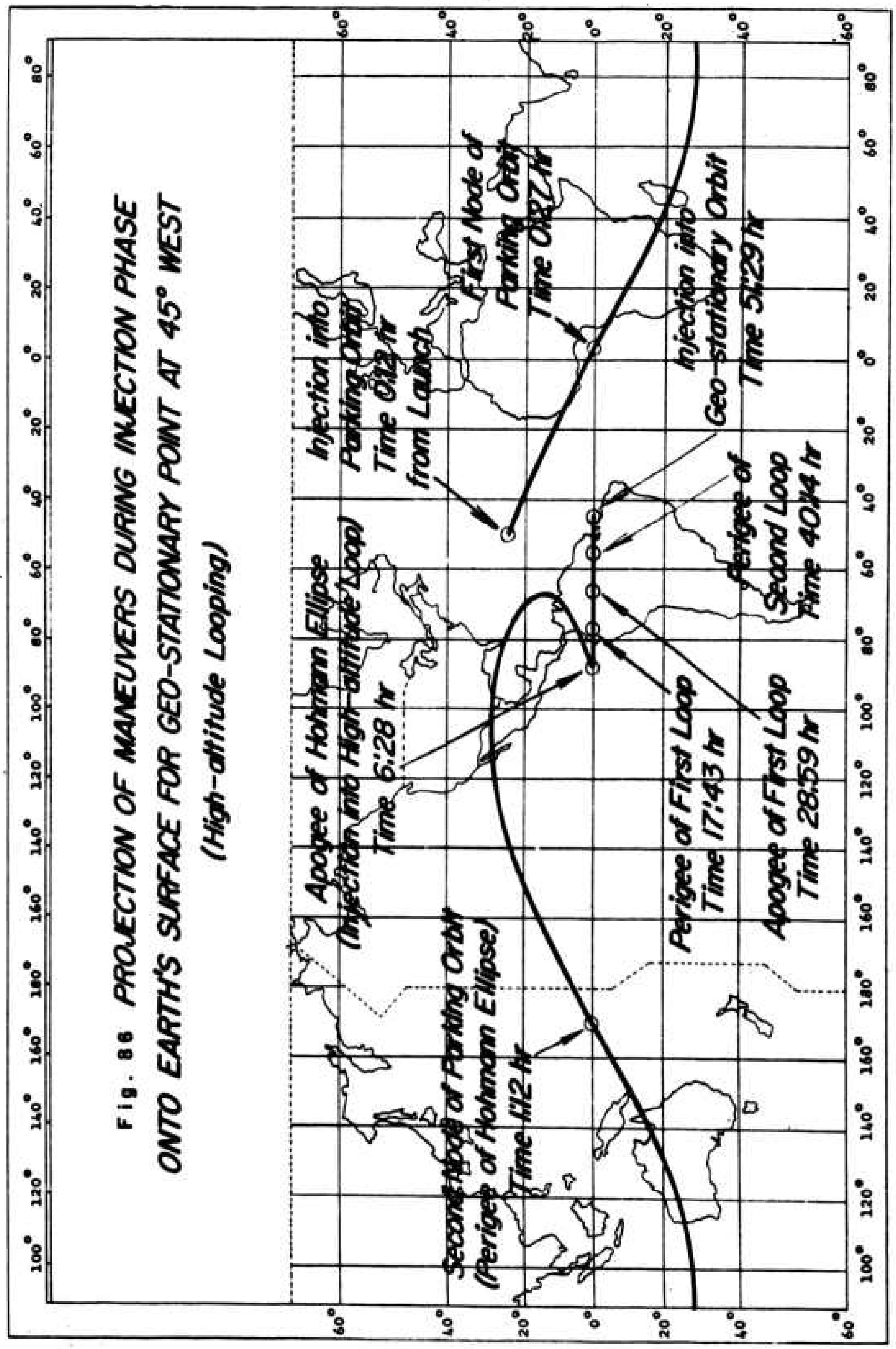

Fig. 86 PROJECTION OF MANEUVERS DURING INJECTION PHASE ONTO EARTH'S SURFACE FOR GEO-STATIONARY POINT AT 45° WEST

(High-altitude Looping)

Restrictions will be placed upon the observable deviations of the satellite from its intended position, and vernier corrections will be required following the injection.

The scheme considered consists of three "kicks" or impulsive thrust applications.

The function of the first kick is two-fold (Fig. 87). First, it is applied in such a manner that the velocity vector following application is in the plane of the equator. This equatorialization requires a rotation of the original nodal velocity vector through an angle equal to the inclination of the erroneous orbit to the equatorial plane.

Second, the first kick places the satellite at an apsis of a Hohmann ellipse whose other apsis lies at the desired geo-stationary radius.

After the first impulse application, the satellite coasts through 180 degrees (space-fixed) to the subsequent apsis of the Hohmann ellipse at which it possesses the correct radius. The second kick is given colinear with the velocity vector upon arrival at this apsis so that the orbit remains equatorial (Fig. 88).

During the transit of the Hohmann ellipse, the longitude error existing at the application of the first kick will either increase or decrease by only some relatively small magnitude.

The purpose of the second kick is to correct the longitude error existing upon arrival at the point of application of the second kick. The period of the orbit resulting from the second kick is such that following one complete revolution in the orbit, the rotation of the earth will have brought the desired subpoint underneath the satellite. The third kick brings the satellite into the desired orbit by applying a final velocity correction.

C. (U) Noise Radiation from Multiple Engines

The use of multiple engines in large booster systems has created some concern about the sound and vibration effects such clustering would bear upon the immediate environment at either the static test or launch site. In order to define the severity of such effects, ABMA first conducted theoretical studies which indicated that sound pressure level could be as high as 131 db at a 3200-foot distance from the eight-engine SATURN cluster. However, the attenuation imposed by the test stand configuration, weather conditions and surrounding terrain, for example, is theoretically indeterminate and puts the results of studies on uncertain grounds.

To aid in achieving a closer definition for the conditions encountered, a test program was conducted by surveying noise radiation from an

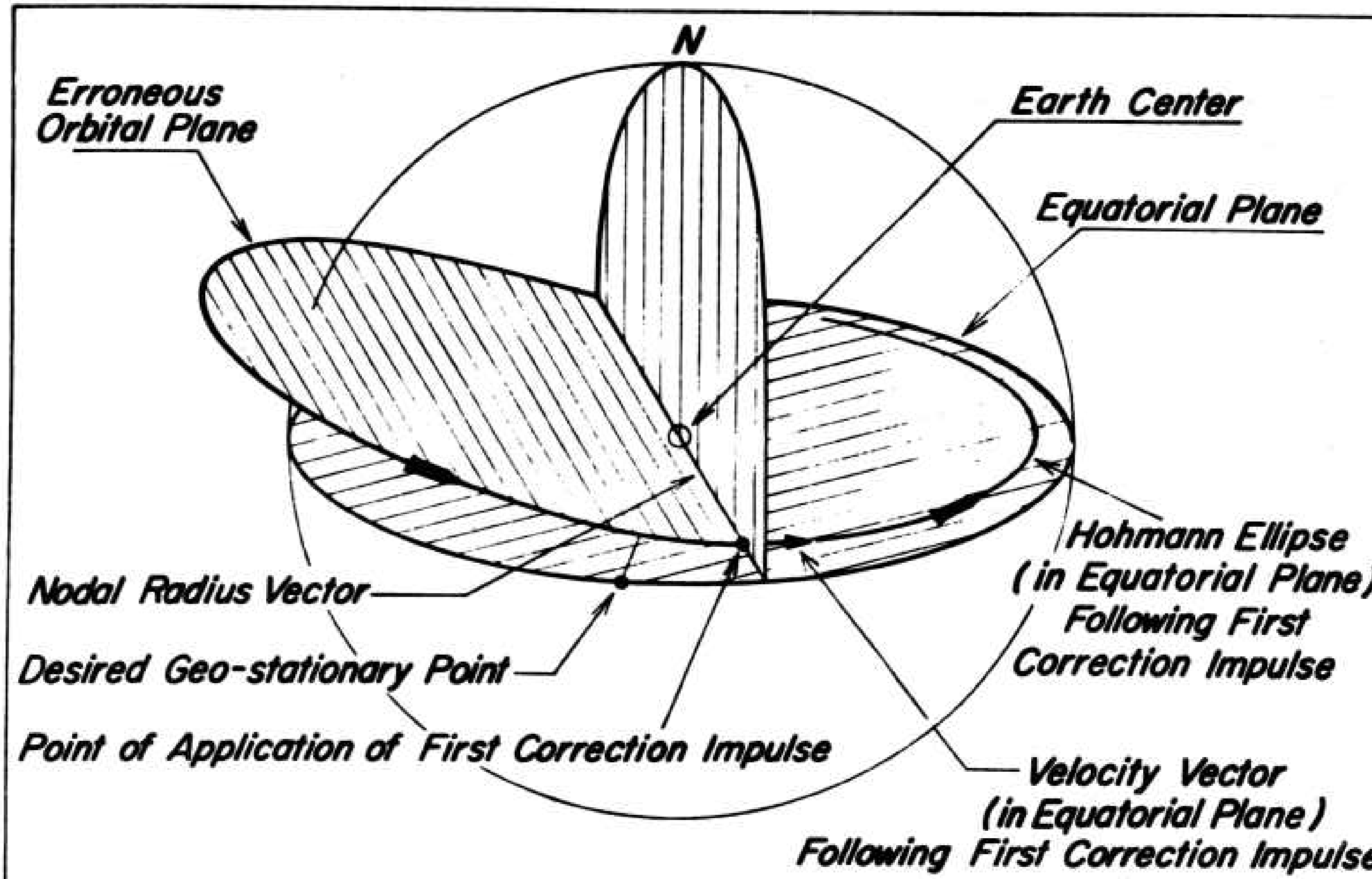

Fig. 87 Schematic Showing Geometry of First Impulse of Vernier Phase

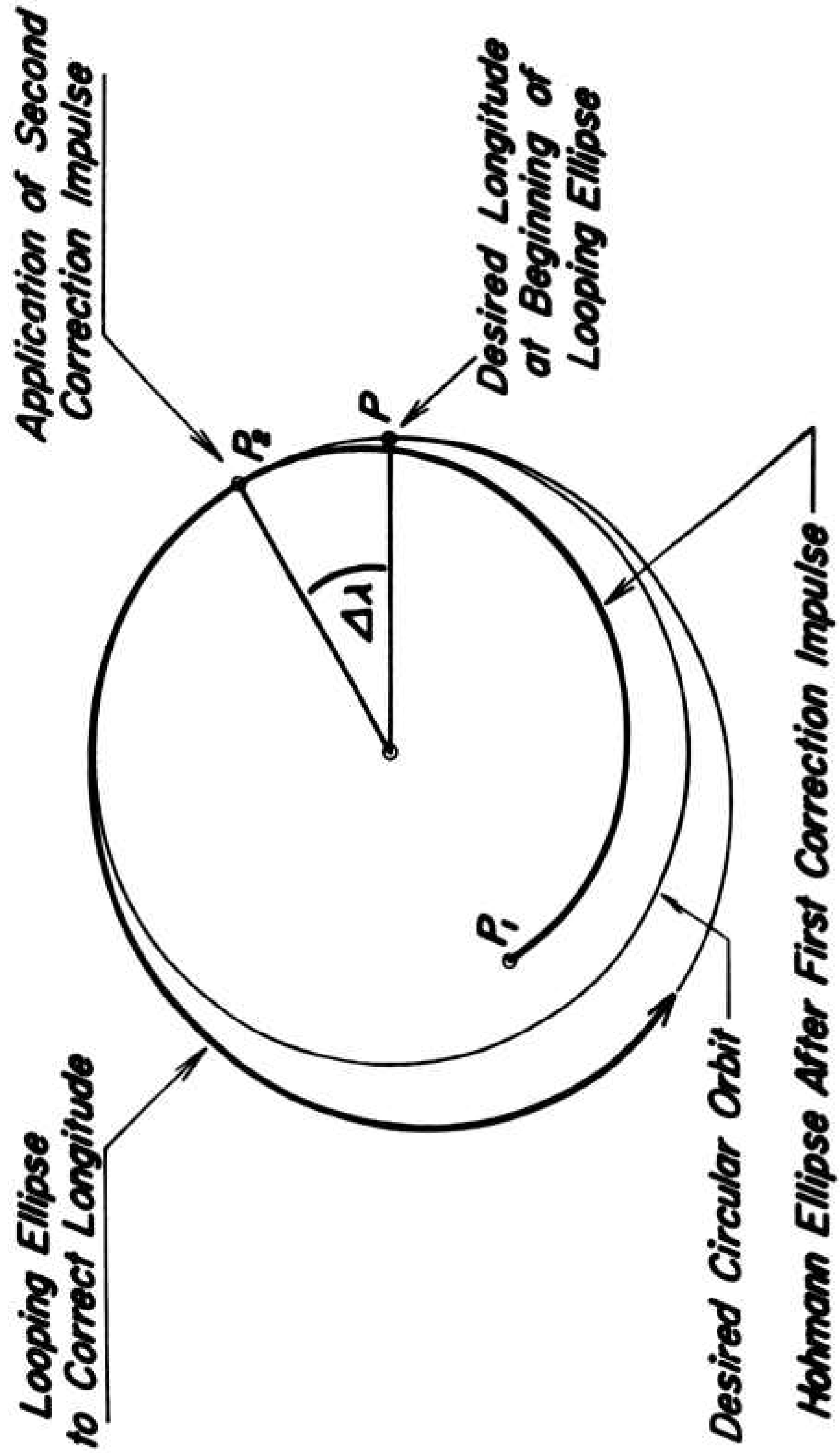

Fig. 88 Schematic Showing Geometry of Second Impulse of Vernier Phase

available 500-lb thrust liquid propellant motor in configurations of one, two, four and eight rockets (Fig. 89). Measurements of sound pressure were made along an arc of a semi-circle with a radius of 200 feet, from directly in front of the rocket exhaust to a position almost perpendicular to the exhaust stream. Pickups were also placed along a common bearing at radii of 50, 100, 200, and 300 feet from the source.

Measurements along the common bearing gave an indication of the effects of the physical structure of the test facility, effects of the atmosphere, and effects of the acoustic near-field on the over-all results of the survey. These measurements were reduced in relatively narrow frequency bands to give more detail about the frequency spectra of the sound radiated by the rockets.

Summary of the results of the survey includes the following:

a. The acoustic power level radiated by the clusters went up either 2 or 3 db each time the number of engines in the cluster was doubled.

b. The over-all space average sound pressure level (SPL_{av}), which does not necessarily correspond to any individual measured SPL, increased from 112 db for the single engine firing to 115 db for a cluster of two engines, to 117 db for a four-engine cluster, to 119 db for a test of the full cluster of eight engines (Fig. 90). The SPL_{av} was calculated for a radius of 200 feet and a measuring arc of 80 degrees.

c. The directivity of the radiation was fairly narrow for a single engine, with most of the energy being radiated in a sector 30° to 50°. This sector tends to broaden as the number of engines in the cluster is increased until, for eight engines, the lobe of most intense radiation is 40° wide, covering the arc from 50° to 90°.

d. One parameter of the acoustic noise specifications of the various test configurations can be fairly well correlated with one parameter of the rocket engine performance. Specifically, the over-all acoustic power developed by a rocket engine noise source can be correlated with the mechanical power developed by the rocket exhaust. In this survey, the acoustic power was almost directly proportional to the mechanical power developed by the cluster for any test configuration. This relation will probably hold true for sound radiation from the full-sized SATURN booster.

The power spectra of the noise produced by the different configurations of the cluster revealed that as the number of engines in the cluster increased, the acoustic power tended to shift to the lower frequencies. The higher frequency octave bands increased by only 3 db, while power in the lower frequency bands increased 20 db as the number of engines in the cluster increased from one to eight.

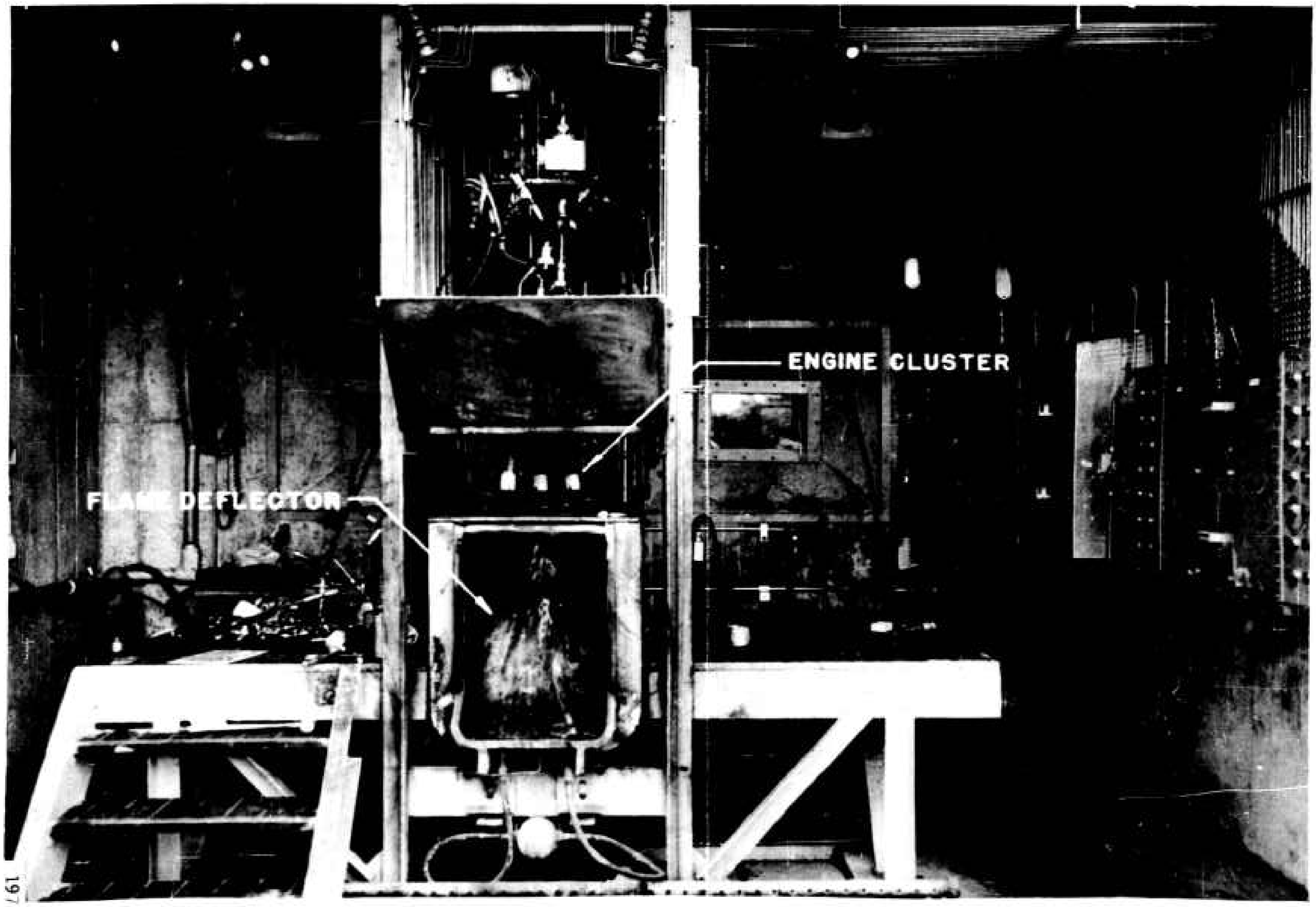

FIG. 39 SCALE MODEL ENGINE CLUSTER SETUP FOR SOUND INTENSITY MEASUREMENTS

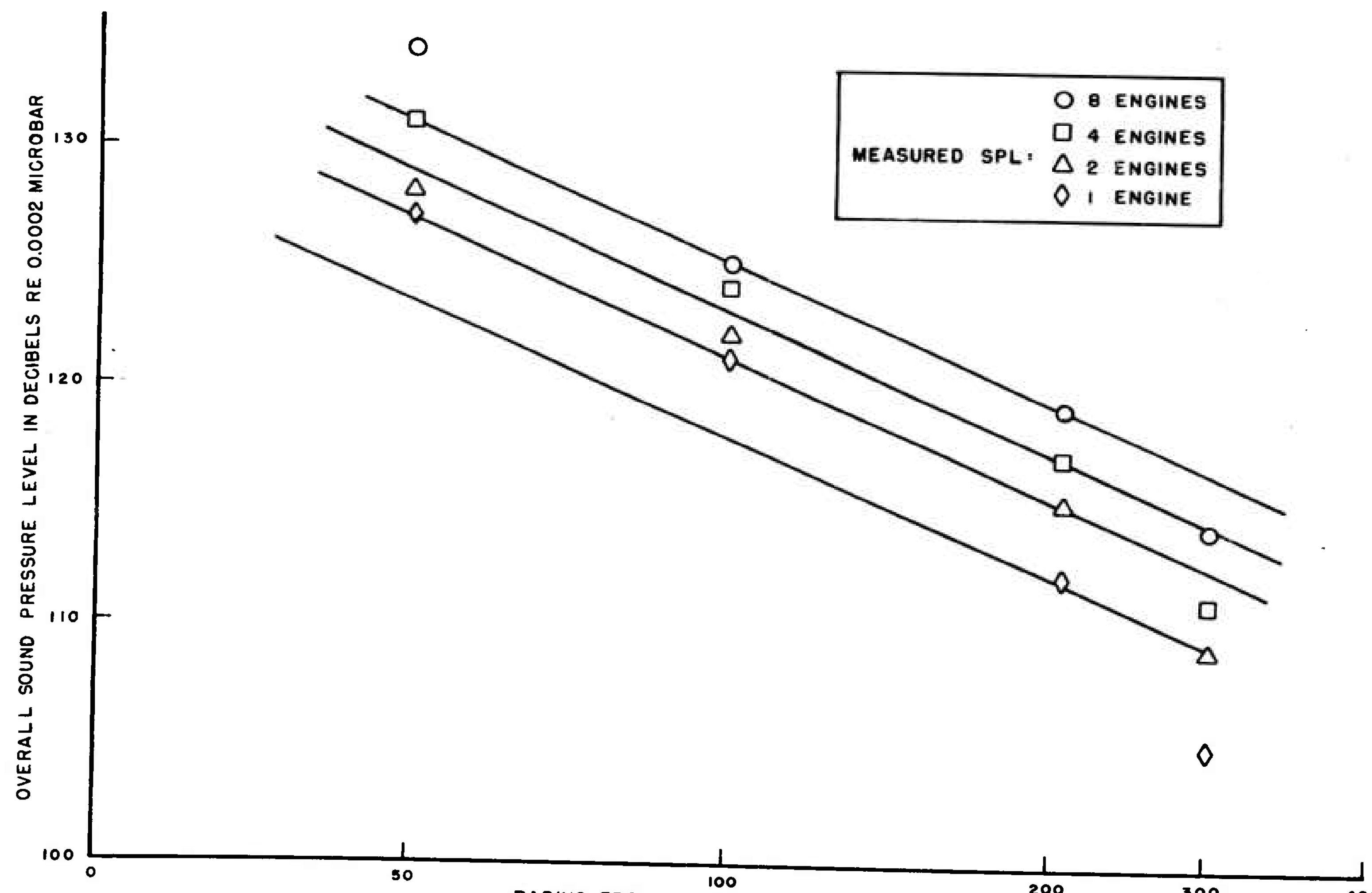

FIG. 90. OVERALL SOUND PRESSURE VERSUS DISTANCE FOR EACH ENGINE TEST CONFIGURATION

The most important inconsistency between the model and the full-sized SATURN booster was in the frequency scale. Probably the shape of the sound spectra for the model and the full-sized configuration will be similar; but the model data are close to 20 times higher in frequency than is likely to occur in sound radiation from the SATURN booster.

In general, the data received from the above tests agreed relatively well with the data determined by the theoretical studies. However, it should be pointed out that a foolproof basis for converting the results from scaled-down tests to a full-scale firing has not yet been found.

SECTION VII. (U) PROBLEM AREAS

There are no major technical problem areas presently apparent in the SATURN program that might delay scheduled static or flight tests of the booster. Probably the least defined area in the booster, and the most fertile for unforeseen problems, is that of structural dynamic behavior, both during static test and flight. A precise theoretical definition of dynamics, especially for a structure as complex as that of the SATURN booster, is extremely difficult to achieve. For this reason ABMA has proposed the construction of a dynamic test tower which will allow experimental verification of the booster's and the complete vehicle's dynamic characteristics prior to the first flight. Other areas being closely monitored, so that any problems may be caught at an early stage, are accessibility, environmental parameters during standby and flight, propellant flow, and system control requirements.

ARPA approval of the recommendations made by NASA concerning the development of the SATURN C is urgently needed. In the interim, ABMA is proceeding with development according to the recommended concept, under the assumption that ARPA approval will be received. Although the delay of this approval to date has not caused technical problems, it is causing delays in procurement actions which could result in schedule slippages.

APPENDIX

REPORT TO THE ADMINISTRATOR, NASA

ON

SATURN DEVELOPMENT PLAN

BY

SATURN VEHICLE TEAM

15 December 1959

INTRODUCTION

The President of the United States, on 2 November 1959, announced his intention to transfer the Development Operations Division of the Army Ballistic Missile Agency (ABMA) and the Saturn project to NASA. In anticipation of this transfer, the NASA and Department of Defense have established an interim working agreement that provides for immediate assumption by NASA of responsibilities for technical management of the Saturn vehicle development. On 17 November 1959, the Associate Administrator of NASA requested the Director of Space Flight Development to

> "form a study group with membership from NASA, the Directorate of Defense Research and Engineering, ARPA, ABMA, and the Air Force from the Department of Defense to prepare recommendations for guidance of the development, and specifically, for selection of upper stage configurations.
>
> Attention in the study should be directed toward
>
> 1. Missions and payloads,
> 2. Technical development problems,
> 3. Cost and time for development, and
> 4. Future growth in vehicle performance."

A Saturn vehicle team was established with the following membership:

Dr. Abe Silverstein, Chairman	NASA
Col. N. Appold	USAF
Mr. A. Hyatt	NASA
Mr T. C. Muse	ODDR&E
Mr. G. P. Sutton	ARPA
Dr. W. von Braun	ABMA
Mr. E. Hall, Secretary	NASA

The results and recommendations of the Saturn vehicle team are summarized in this report and the more detailed findings are presented in Appendices A, B, and C, which are attached.

The Saturn project was initiated on 15 August 1958 by an order from the Advanced Research Projects Agency to the Army Ordnance Missile Command to develop a large booster vehicle of approximately 1.5 million pounds of thrust using available engines. Authorization was given for construction of test facilities, development and early captive firing of the first stage, launchings of three first stages with dummy upper stages, and one with a live upper stage. A brief chronology of important actions relative to the Saturn project are contained in Appendix A.

For the past several months technical studies have been conducted by ABMA, ARPA, and NASA to establish the performance characteristics of the Saturn vehicle with various upper stages. The results of these independent studies were in close agreement and form a basis for this evaluation.

Presentations were made to the Saturn vehicle team on missions for the Saturn vehicle by both NASA and the Department of Defense. The following missions, listed in their order of importance, were established for the Saturn vehicle (Appendix B).

a. Lunar and deep space missions with an escape payload of about 10,000 pounds.

b. Payloads of about 5,000 pounds in a 24-hour equatorial orbit.

c. Manned spacecraft missions such as Dyna Soar, with a weight of about 10,000 pounds in a low orbit (two-stage launch vehicle)

These missions were established for the initial Saturn vehicle configuration. It is recognized that the initial Saturn configuration must provide for growth to permit increased payload capability in the lunar, deep space, and satellite missions. Early capability with an advanced vehicle and possibilities for future growth were accepted as elements of greatest importance in the Saturn vehicle development.

The current Saturn first stage with eight engines giving a total thrust of nearly 1,500,000 pounds was reviewed. The many problems associated with its development and operation were discussed. Attention was given to alternate configurations for the first stage including the use of solid propellant rockets, a cluster of four 400,000-pound thrust engines, and a single engine of 1,500,000 pounds of thrust. The problems of clustered tanks as compared with those of a single large tank were also considered.

A wide variety of upper stages utilizing conventional and high-energy propellants and of various weights were compared on the basis of performance, technical feasibility, growth potential and probable time and cost to develop. Various tank configurations, including clusters of existing IRBM's, which were independently analyzed by ABMA and NASA, were also studied by the group. A discussion of the technical items covered is contained in Appendix C.

SUMMARY OF RESULTS

After a review of the many possible configurations of Saturn vehicles, the team reduced its detailed considerations to those shown in Table I.

The payload capabilities of the configurations shown in Table I for the most important missions listed in the Introduction are given in Table II.

Vehicle A-1, with upper stages consisting of a modified Titan stage 1 and Centaur upper stage, makes maximum utilization of existing hardware and would most likely have earliest flight availability and lowest cost. It fails, however, to meet the mission requirements for the lunar and 24-hour missions and, because of its slenderness (120-inch diameter upper stages), vehicle A-1 is a structurally marginal configuration. Development of a 160-inch diameter second stage similar in construction to the Titan first stage was reviewed and eliminated from detailed consideration because it limited the growth potential of the Saturn.

The A-2 vehicle, with a cluster of IRBM's as the second stage, is similar to the A-1 configuration in its use of existing hardware. Vehicle A-2 fails to meet the requirements for lunar and deep-space missions and for the 24-hour equatorial orbit.

Vehicle B-1 meets the requirements of the missions, but requires the development of a new conventionally fueled second stage that is approximately twice the size of our current ICBM's. The cost and time to develop this large second stage which seemed to be interim in character for advanced missions raised doubts as to the desirability of developing this vehicle.

In examining vehicles A-1, A-2, B-1, and others, it became apparent that the highest priority missions for the Saturn vehicle could not be accomplished in a reasonable design without the use of high-energy propellants in the top stages. If these propellants are to be accepted for the difficult top-stage applications, there seems to be no valid engineering reasons for not accepting the use of high-energy propellants for the less difficult

TABLE I

DESCRIPTION OF POSSIBLE SATURN VEHICLE CONFIGURATIONS

STAGE	1	2	3	4
A-1	LOX/RP EIGHT H-1 ENGINE CLUSTER	LOX/RP TITAN 120" DIA.	CENTAUR 120" DIA. TWO 15K ENGINES	
A-2		CLUSTER OF IRBM'S	CENTAUR 120" DIA. TWO 15K ENGINES	
B-1		LOX/RP 220" DIA.* FOUR H-1 TYPE ENGINES	LOX/LH 220" DIA.* FOUR 15-20K ENGINES	CENTAUR 120" DIA. TWO 15K ENGINES
C-1		LOX/LH 220" DIA.* FOUR 15-20K ENGINES	CENTAUR 120" DIA. TWO - 15K ENGINES	
C-2		LOX/LH 220" DIA.* TWO 150-200K ENGINES	LOX/LH 220" DIA.* FOUR 15-20K ENGINES	CENTAUR 120" DIA. TWO 15K ENGINES
C-3	LOX/RP 2.0+ MILLION POUND THRUST ENGINE CLUSTER	LOX/LH 220" DIA.* FOUR 150-200K ENGINES	LOX/LH 220" DIA.* TWO 150-200K ENGINES	LOX/LH 220" DIA.* FOUR 15-20K ENGINES

*Nominal tank diameter
H-1 Engine - 165,000 to 188,000 lb. thrust

TABLE II

PERFORMANCE OF POSSIBLE SATURN VEHICLE CONFIGURATIONS

	*DRY GROSS PAYLOAD, POUNDS (AMR LAUNCH)		
MISSION	LUNAR CIRCUMNAVIGATION	DYNA SOAR-TYPE (2 STAGE)	24 HR. SATELLITE
A-1	6,800 (3)	12,500 (2)	3,800 (3)
A-2	7,000 (3)	8,500 (2)	3,500 (3)
B-1	11,000 (3) 15,000 (4)	12,600 (2)	5,000 (3) 9,000 (4)
C-1	9,000 (3)	24,000 (2)	5,500 (3)
C-2	15,500 (3)	40,000 (2)	9,000 (3)
C-3	25,000 (3) 34,000 (4)	54,000 (2)	12,000 (3) 21,800 (4)

*DRY GROSS PAYLOAD includes net payload, shrouds, instrument compartment and instrumentation, and guidance and control. (Does not include flight reserve propellant) application to intermediate stages. Of course, the maximum payload capability with the Saturn first stage booster will be achieved if high-energy propellants are used in all the upper stages. Current success in the Centaur engine program substantiates the choice of hydrogen and oxygen for the high-energy propellants.

The C-1 configuration (Tables I and II) is the first phase in the development of a vehicle using all hydrogen and oxygen upper stages (see figures 1 and 2). Succeeding phases are C-2 and C-3 with progressively increasing payload capability. As the development proceeds from phase to phase, a new stage is added to the vehicle. Stages developed for early phases continue to be used in all latter phases (see figure 2). Thus all developments lead to increased flight capability and reliability.

Configuration C-1 permits early flights and essentially meets the established mission requirements. The upper stages consist of a four engine hydrogen-oxygen second stage (S IV) and a Centaur upper stage (S V) as a third stage. The engines for the second and third stage are the same. Uprating of the 15K Centaur engine to 20K is necessary for the second stage.

Configuration C-2 is adapted from C-1 by the addition of a new hydrogen-oxygen second stage (S III). The development of a 150K - 200K pounds of thrust hydrogen-oxygen rocket engine is required to power the new stage.

Configuration C-3 increases the payload capability by adding a second stage (S II) with four 150K - 200K pound thrust engines. The thrust of the first stage is also increased to over two million pounds. This thrust may be obtained by replacing the four center engines with one F-1 engine or by uprating all eight H-1 engines.

RECOMMENDATIONS

It is recommended that:

1. A long-range development plan for the Saturn vehicle be established that will provide, through a consecutive development of building-block upper stages, a substantial early payload capability and a final configuration that exploits the maximum capability of the Saturn first stage. Vehicle reliability will be emphasized in the building-block program through a continued use of each development stage in later vehicle configurations.

2. All upper stages be fueled with hydrogen-oxygen propellants.

3. The initial vehicle configuration, C-1, consists of the following:

a. The eight engine first stage currently under development at ABMA,

b. A newly developed second stage using four of the current Centaur engines uprated to 20,000 pounds of thrust.

c. The third stage using the current Centaur stage modified only as required for vehicle and payload attachments.

4. The following developments be initiated immediately:

a. A 150-200K hydrogen-oxygen fueled rocket engine for stages S II and S III.

b. A design study of hydrogen-oxygen upper stages S II and S III using the 150-200K engines.

5. The development schedule shown in Table III be adopted.

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FIGURE I
SATURN VEHICLE BUILDING BLOCKS

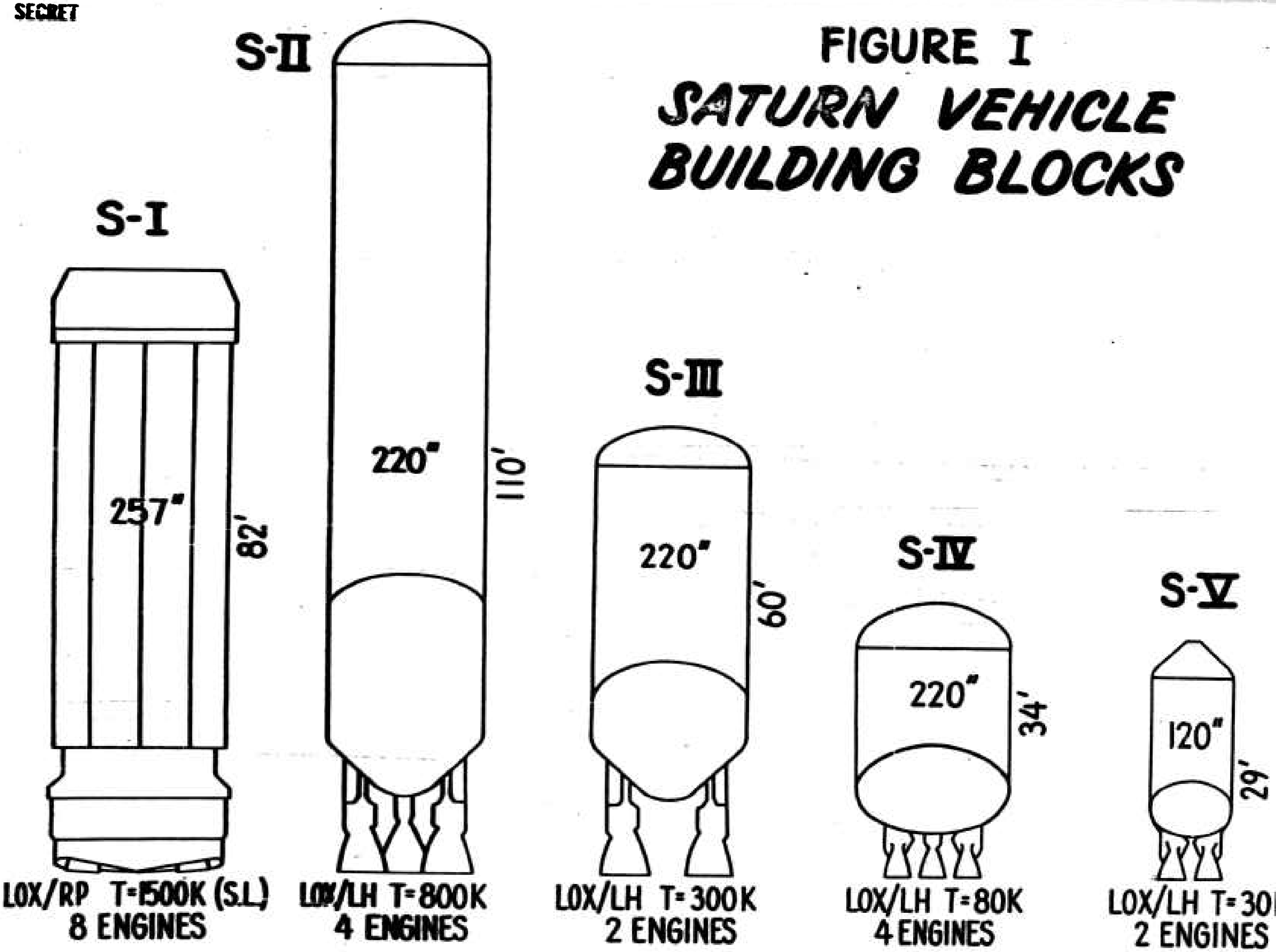

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Figure I

Submitted by:

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Abe Silverstein, NASA (Chairman)

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Abraham Hyatt, NASA

GP Sutton

George P. Sutton, ARPA

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T. C. Muse, ODDR&E

Norman C. Appold

Norman C. Appold, Col., USAF

Wernher von Braun

Wernher von Braun, ABMA

Eldon W. Hall

Eldon Hall, NASA (Secretary)

SECRET

SATURN CONFIGURATIONS

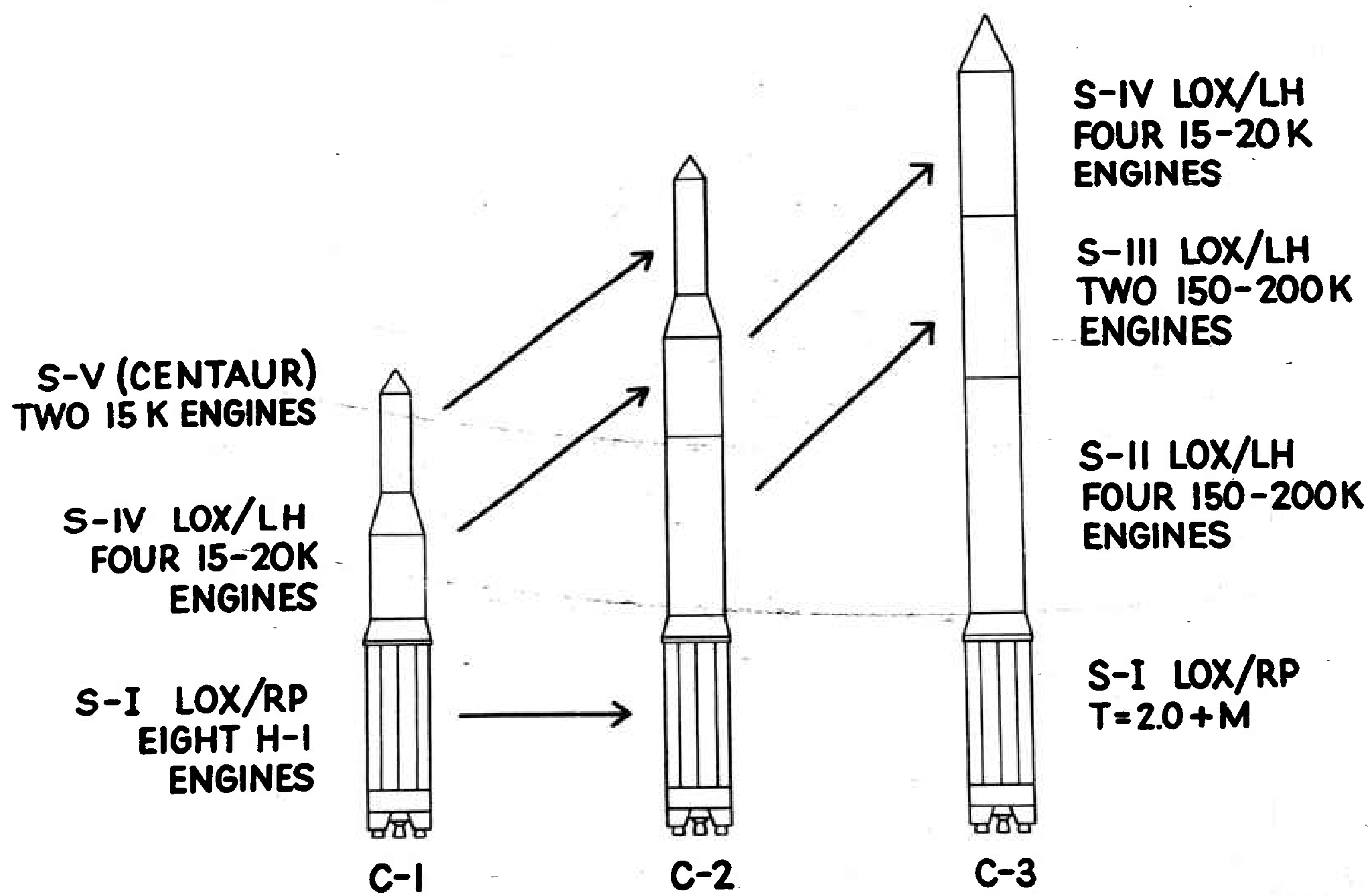

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Figure 2

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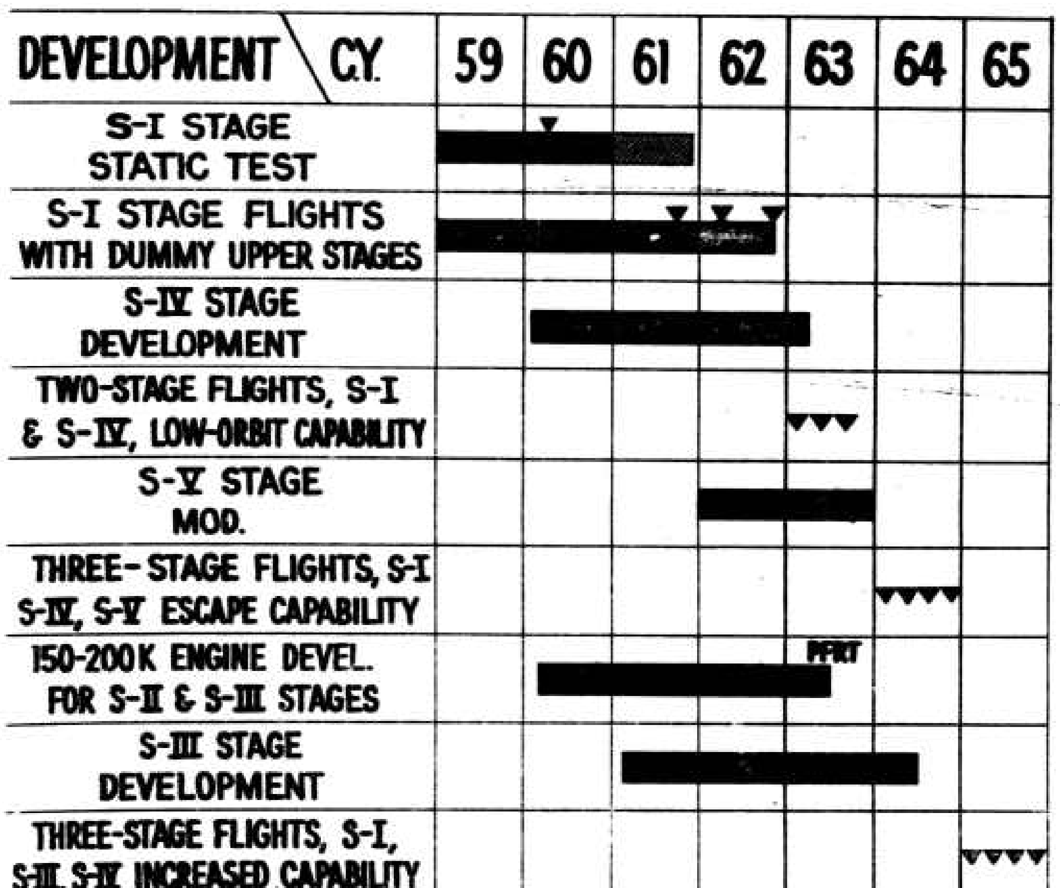
TABLE III

SATURN DEVELOPMENT SCHEDULE*



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1. SCHEDULE BASED ON FUNDING OF $70M IN FY 60, $140 M IN FY 61, AND ANTICIPATED FUNDING OF $150 M TO $200 M IN SUBSEQUENT YEARS AT A NOMINAL FIRING RATE OF 4 PER YEAR.

2. FUNDING INCREASES OF $15 M IN FY60 AND $30M IN FY61 WOULD PROVIDE INCREASED RELIABILITY AND EXPEDITE SCHEDULE 3 MONTHS FOR 2-STAGE FLIGHTS AND 6 MONTHS FOR 3-STAGE FLIGHTS.

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