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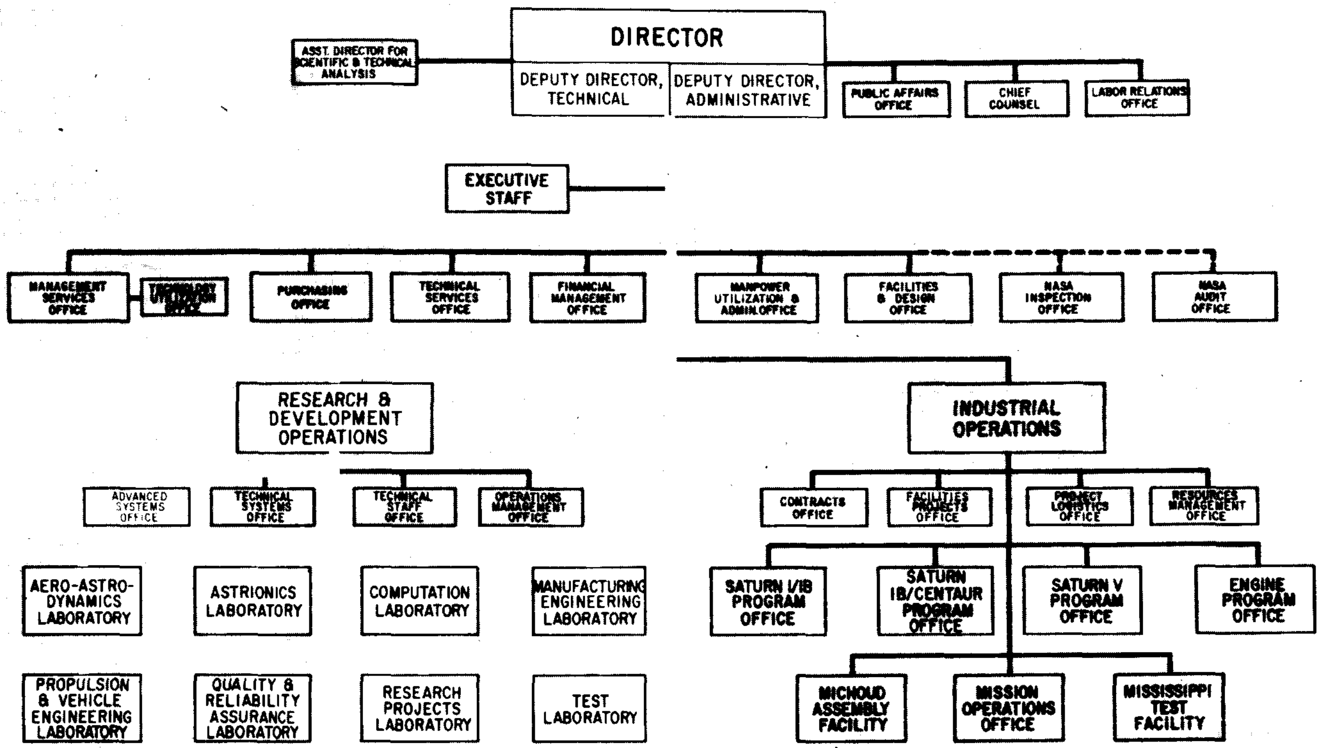
RESEARCH ACHIEVEMENTS REVIEW SERIES NO. 3

RESEARCH AND DEVELOPMENT OPERATIONS
GEORGE C. MARSHALL SPACE FLIGHT CENTER
HUNTSVILLE, ALABAMA

GEORGE C. MARSHALL SPACE FLIGHT CENTER



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NATIONAL AERONAUTICS AND SPACE ADMINISTRATION
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CRYOGENIC TECHNOLOGY RESEARCH AT MSFC

RESEARCH ACHIEVEMENTS REVIEW SERIES NO. 3

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PREFACE

In 1955, the team which has become the Marshall Space Flight Center (MSFC) began to organize a research program within its various laboratories and offices. The purpose of the program was two-fold: first, to support existing development projects by research studies and second, to prepare future development projects by advancing the state of the art of rockets and space flight. Funding for this program came from the Army, Air Force, and Advanced Research Projects Agency. The effort during the first year was modest and involved relatively few tasks. The communication of results was, therefore, comparatively easy.

Today, more than ten years later, the two-fold purpose of MSFC's research program remains unchanged, although funding now comes from NASA Program Offices. The present yearly effort represents major amounts of money and hundreds of tasks. The greater portion of the money goes to industry and universities for research contracts. However, a substantial research effort is conducted in house at the Marshall Center by all of the laboratories. The communication of the results from this impressive research program has become a serious problem by virtue of its very voluminous technical and scientific content.

The Research Projects Laboratory, which is the group responsible for management of the consolidated research program for the Center, initiated a plan to give better visibility to the achievements of research at Marshall in a form that would be more readily usable by specialists, by systems engineers, and by NASA Program Offices for management purposes.

This plan has taken the form of frequent Research Achievements Reviews, with each review covering one or two fields of research. These verbal reviews are documented in the Research Achievements Review Series.

Ernst Stuhlinger
Director, Research Projects Laboratory

This paper presented March 25, 1965

TABLE OF CONTENTS

A REVIEW OF CRYOGENIC TECHNOLOGY RESEARCH AT MARSHALL SPACE FLIGHT CENTER

by C. C. Wood

LIST OF ILLUSTRATIONS

LIST OF ILLUSTRATIONS (Concluded)

LIST OF TABLES

A REVIEW OF CRYOGENIC TECHNOLOGY RESEARCH AT MARSHALL SPACE FLIGHT CENTER

By

C. C. Wood

SUMMARY

Cryogenic research conducted in support of current space vehicle programs and in anticipation of future vehicle requirements is summarized in this review. Results are reported for investigations in fluid mechanics, propellant storage, and instrumentation. Advances in cryogenic technology are needed for missions lasting many days. Special support is needed for studies in transient fluid mechanics at reduced gravity, superinsulation for flight tankage, efficient cryogenic reliquefaction of evaporated cryopropellants, and system integration.

I. INTRODUCTION

Cryogenic technology research in support of space-vehicle programs encompasses a diverse field of investigation. A part of this research effort is discussed in this review under the divisions of fluid mechanics, cryogenic propellant storage, and instrumentation.

Hydrogen as a rocket-engine fuel has not been used as extensively as oxygen, and specific parameters are not commonly known. As an introduction to the work discussed in the next sections, comparative data on key characteristics are given in Figure 1. In part A of the illustration, a comparison is made of heat transfer coefficients as a function of temperature ratio, $T/T_{saturation}$ For temperature ratios expected in high-pressure thrust chambers, the liquid heat transfer ratio of hydrogen to oxygen is 12. This indicates that hydrogen systems would require simpler, lighter equipment, but the opposite would be required for propellant storage. Part B shows the liquid hydrogen temperature increase as twice that of oxygen for equal storage volumes and heat leaks. The pressure rise ratio of hydrogen and oxygen per unit propellant temperature increase varies between 8 and approximately 15. Obviously, hydrogen is far more difficult to store than oxygen.

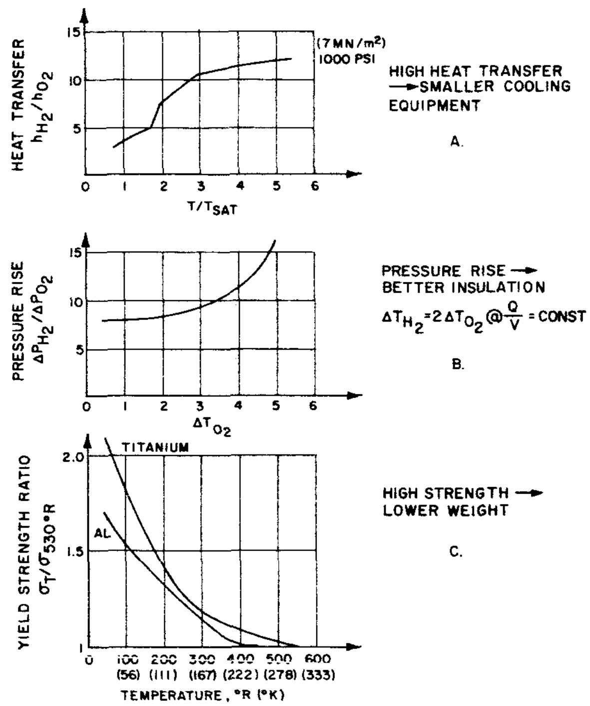


FIGURE 1. COMPARISON OF SOME H_2/O_2 PROPERTY EFFECTS

Mechanical design considerations are illustrated in part C, in which yield strength of aluminum and titanium is shown to increase with decreasing metal temperature. As the temperature decreases from 160° to 40° R (89° to 22° K), there is a significant increase in yield strength for titanium and a more modest increase for aluminum.

II. FLUID MECHANICS

A. PRESSURIZATION SYSTEMS

Several aspects of pressurization systems design have so far defied adequate analytical representation and require extensive testing for design verification. Two primary examples are the thermodynamic behavior of pressurization gases inside the propellant tank and flow instabilities in the pressurization gases of cryogenic liquid/gas systems. An understanding of the thermodynamics of gases within a tank is important for minimizing pressurant and weight and also for designing systems that do not possess transient flow instabilities. Flow instabilities originating within the heat generator of a pressurization system usually cannot be tolerated, and elimination of instabilities afterward imposes significant weight penalties and loss of system flexibility. Marshall Space Flight Center (MSFC) is currently studying both propellant tank thermodynamics and flow instabilities. Some results of these studies are explained in Figures 2 through 5.

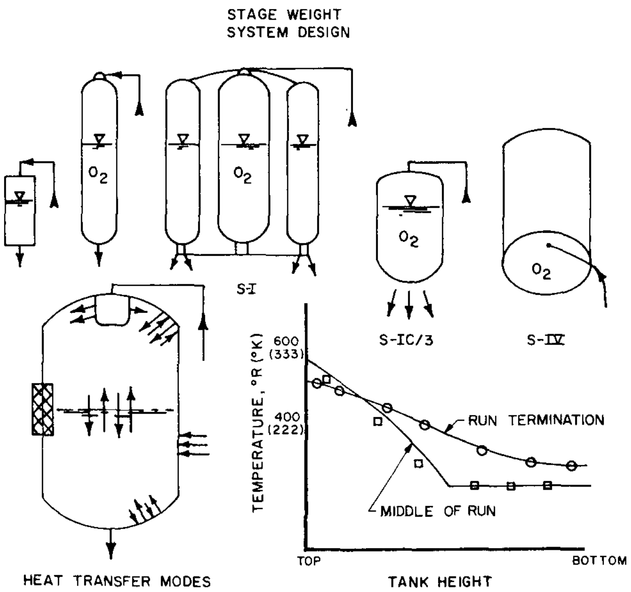


FIGURE 2. PROPELLANT PRESSURIZATION

Figure 2 illustrates tanks used in the MSFC experimental program to verify analytical procedures for predicting thermodynamic behavior. Existing modes of heat transfer and a comparison of analytical predictions with experimental data are shown. The test tanks vary significantly in size, shape, and number, and should provide ample variables for validating developed analytical procedures. The effects of propellant sloshing can be assessed on the large single oxygen container (6 ft by 40 ft; 1.8 m by 12.2 m) and on the one-third-scale S-IC container (13 ft by 26 ft; 4 m by 8 m). Liquid oxygen and nitrogen were pressurized by vaporized oxygen, nitrogen, and helium. Data in Figure 2 obtained midway and near the end of propellant drainage agree well with analytical procedures. Measured pressurant weights and pressurant weights established by analytical methods in most cases agree within 10 percent. Data accumulated from tests with these various tanks were published in NASA Technical Memorandum S-53165. Currently, liquid hydrogen data are being analyzed; the completion of these analyses will be the culmination of several years' work in this field by MSFC.

Figure 3 is an interesting plot of oxygen tank pressurant mass per unit pressure versus vehicle

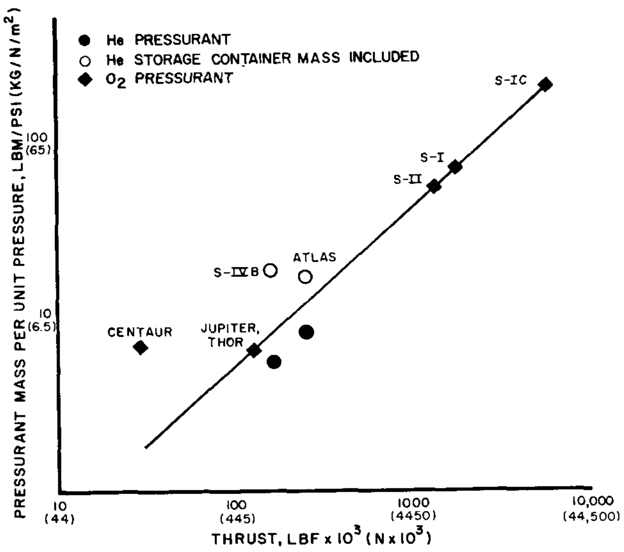


FIGURE 3. RELATIONSHIP BETWEEN OXYGEN TANK PRESSURANT MASS AND VEHICLE THRUST

thrust. Data are presented for several vehicles. The Saturn S-IVB stage and the Atlas vehicle use helium as pressurant. Helium is stored in these vehicles at liquid-hydrogen and liquid-oxygen temperatures, respectively. The pressurant masses for the S-IVB and Atlas include the mass of the storage bottles. All remaining vehicles, except Centaur, use liquid oxygen, vaporized and superheated in heat exchangers, as pressurant. The Centaur stage is pressurized by flash boiling of the propellant; the resulting pressurant approaches the propellant saturation temperature and is heavy. Contrary to general opinion, the S-IVB and Atlas helium pressurization systems are heavier than those in vehicles of comparable thrust that use vaporized and superheated oxygen. Helium systems which weigh less than oxygen systems can be designed, but such systems are complex and costly.

Fluids undergoing phase change from liquid to gas usually undergo violent oscillations. Typical pressure instabilities of the J-2 rocket engine heat exchanger, which is designed to vaporize liquid oxygen, are shown in Figure 4. Also shown are

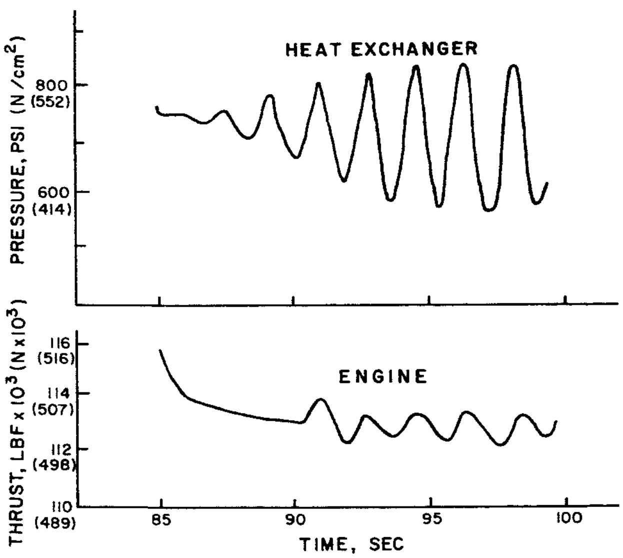


FIGURE 4. HEAT EXCHANGER INSTABILITY AND EFFECTS ON ENGINE INSTABILITY

variations in the J-2 rocket engine thrust caused by pressurant flow oscillations. Preliminary results of a study to determine factors that contribute to fluid instability are shown in Figure 5. Fluid studies for the subcritical state have been completed and regions of instability are mapped in this preliminary figure. Flow instabilities depend on fluid density ρ, fluid entrance velocity U, heat flux to the fluid Q/A, flow rate $\dot{W}$, heat exchanger geometry (tube diameter D and tube length L), and pressurant temperature rise ΔT. Flow instability for supercritical conditions is also being investigated.

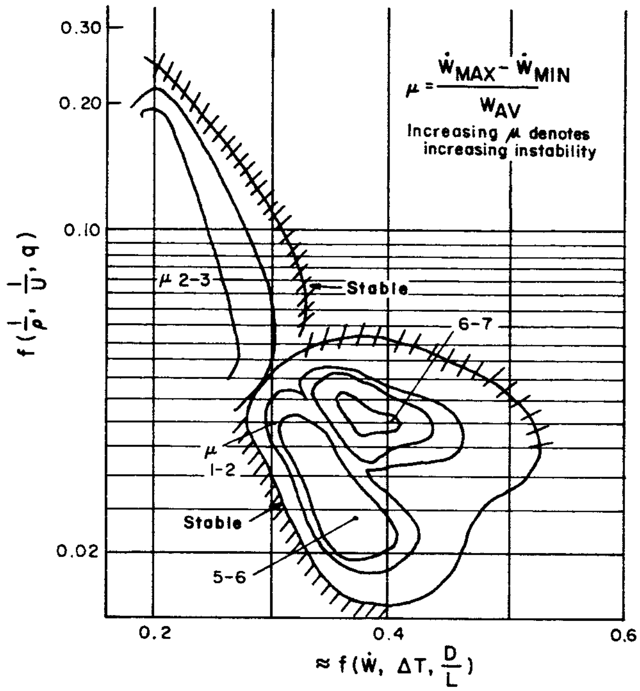


FIGURE 5. HEAT TRANSFER INSTABILITY MAP

B. ROCKET-ENGINE COOLING

Rocket engines developed for high-pressure operation appear to offer many advantages. One advantage is reduced size. Figure 6 illustrates anticipated rocket-engine thrust-chamber cooling requirements versus engine chamber pressure for two engine propellant combinations: liquid oxygen and RP-1, and liquid oxygen and hydrogen. The heat flux expected for a 5000-psi (34 MN/m^2) thrust chamber is approximately 100 Btu/in^2 sec (1.6×10^8 W/m^2), which is sufficient to evaporate a cup of water in approximately 2 seconds. This is approximately twice the heat flux on the nose cone of an intermediate range ballistic missile, during maximum reentry heating. Cooling criteria limit operating pressures of regeneratively cooled thrust chambers to approximately 2000 psi (14 MN/m^2) for hydrogen and 1500 psi (10 MN/m^2) for RP-1. Limitations are wall coking for the RP-1 system and excessive tube pressure drop for the hydrogen system. Stated limits of 2000 and 1500 psi represent significant increases relative to current engine designs.

Current studies to improve thrust-chamber cooling at higher pressure include other aspects of

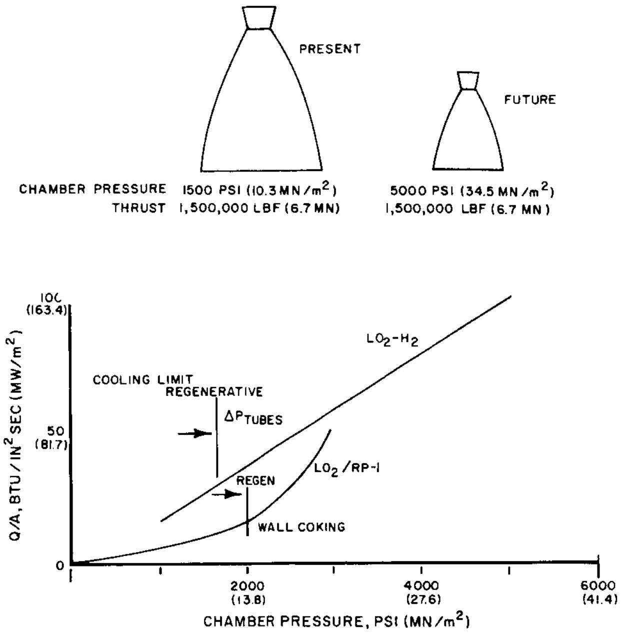


FIGURE 6. ENGINE COOLING

heat transfer and of heat dissipation principles such as film cooling, dump cooling, ablative cooling, etc. Figure 7 shows the results of one of these studies, the influence of surface roughness on heat transfer. The advantage of surface roughness, although significant, is not sufficient to extend existing design limits on chamber pressure.

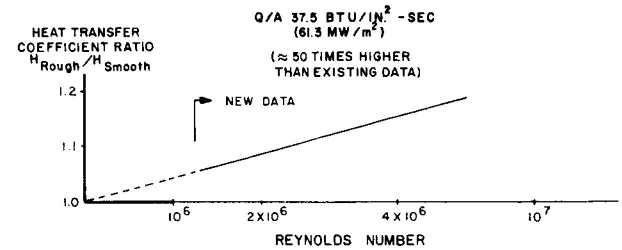


FIGURE 7. INFLUENCE OF SURFACE ROUGHNESS ON HEAT TRANSFER COEFFICIENT

C. GRAVITATIONAL EFFECTS ON HEAT TRANSFER

Gravitational influence on heat transfer for gravity levels both greater and less than 1 g_o are shown in Figure 8. Gravitational levels less than 1 g_o were obtained with a counterweighted package and a 32-foot (9.8-meter) drop tower. A centrifuge

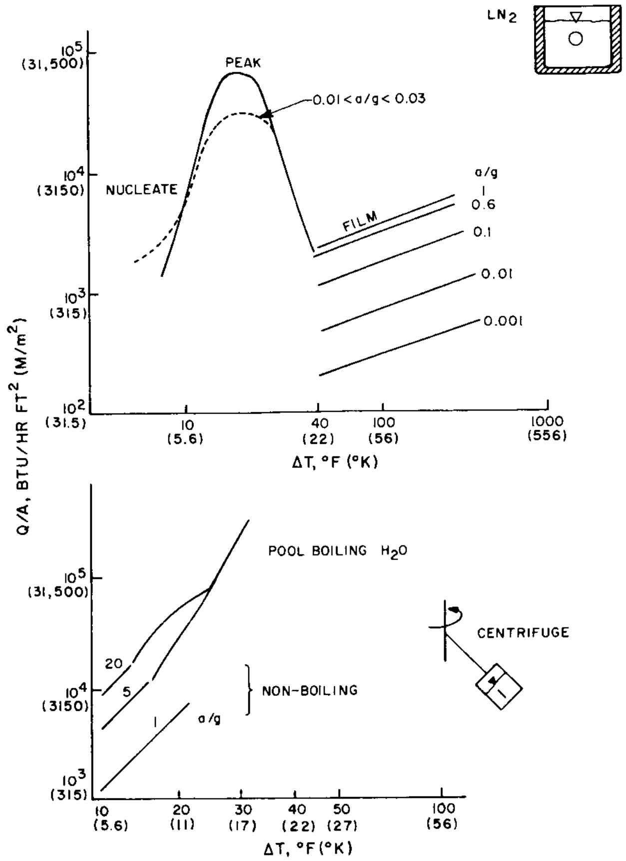


FIGURE 8. GRAVITATIONAL INFLUENCE ON HEAT TRANSFER

was used to produce gravity levels greater than 1 g_o. Saturated nitrogen and water were studied. Variation in the gravitational field from 1 to 0.001 g_o results in a heat-flux reduction of one order in magnitude in the film boiling region. Differences in peak heat flux and heat flux in the nucleate boiling region are smaller. Data are now being obtained with subcooled nitrogen and with a flat plate rather than a sphere as the test specimen. Similar data are to be obtained with liquid hydrogen. Of special interest is the critical heat flux for incipient boiling in liquid hydrogen.

The influence on heat transfer rate of gravitational fields greater than 1 g_o is important in the nonboiling region. These data, shown in Figure 8, were obtained with the heated surface perpendicular to the gravity vector. Data with the heated surface parallel to the gravity vector are also being obtained.

D. FLUID GEYSERING

Geysering is the sudden eruption of cryogenic propellants from lines and is caused by a

Taylor bubble formation within vertical lines attached to containers of large diameter or horizontal line sections. Geysering propellants create potentially dangerous forces when reimpacting with the remaining propellants within the tank. Pressures in excess of 900 psi (6 MN/m^2) were measured at the suction line/rocket engine interface immediately after a large liquid-oxygen tank had geysered. Pressures resulting from line geysering frequently exceed design pressures of feed system components. Considerable effort has been spent on controlling propellant geysering in vehicle suction and facility fill lines.

Figure 9 shows a correlation for predicting the initiation of geysering and the design used on the first stage (S-IC) of the Saturn V rocket to prevent geysering. The correlation is a function of suction-line geometry and heat flux Q/A entering the fluid, the Prandtl number N_{pr} and thermal diffusivity α

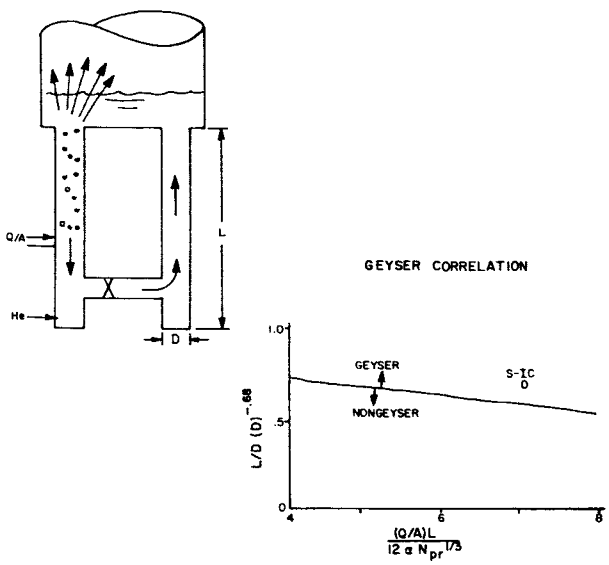


FIGURE 9. PROPELLANT GEYSERING

of the fluid. The correlation is valid for most fluids and has been applied and proven adequate for liquid hydrogen, oxygen, and nitrogen. Line diameters up to 8 inches (20 cm) were used to establish the correlation. Propellant feed lines for two MSFC stages fall within the geysering region as established by the correlation: the hydrogen lines for the S-II stage (which appear to offer no problem because of low propellant density), and the oxygen lines for S-IC. Fluid circulation between interconnecting suction lines prevents geysering in the S-IC; bubble accumulation within the lines, the suspected driving force for a geyser, is eliminated. Helium gas is injected to augment the flow as required. S-IC system performance has been successfully demonstrated, although complexities result because of the many possible operational situations. The correlation for predicting the initiation of geysering has also been validated for large vehicle lines.

E. PROPELLANT STRATIFICATION

Thermal energy entering the propellant tank through the side wall and tank bottom warms the adjacent subcooled propellant and initiates propellant flow within the tank. The propellant temperature increases, warm propellant collects at the liquid surface (stratification), or the two events combine. Figure 10 shows the expected flow patterns and temperature profiles of propellant within a tank at different times during propellant explusion. Stratification

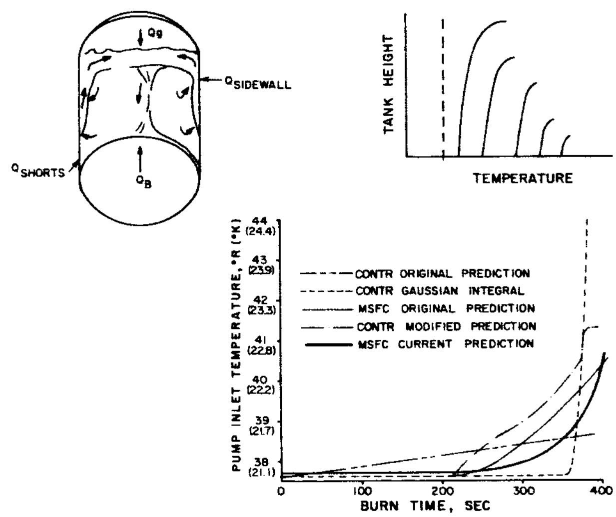


FIGURE 10. PROPELLANT STRATIFICATION

harmfully reduces available net positive suction head (NPSH) to propellant pumps. For compensation, tank pressure must be increased, the relatively warm propellants must be wasted as unusable residuals and left on board, or a combination of the two must be used. The usual solution is an increase in tank pressure. For hydrogen, however, each degree of temperature rise requires a tank pressure increase of 3 psi (21 kN/m^2) for NPSH compensation. This can result in a significant weight penalty. Also shown in Figure 10 are various propellant stratification predictions evolved during stage development. The curve marked "MSFC Current Prediction" is supported by test data from several test configurations including a test of a 22-foot (6.7 m)-diameter tank with special instruments to provide stratification data. Identical prediction techniques have accurately represented propellant stratification during static tests. Between the initial and final stratification prediction, the efficiency of the thermal tank insulation was improved; this explains part of the differences in stratification temperatures shown.

Stratification can also occur under reduced gravity during orbital flight. Since the boundary layer for such conditions is usually laminar, the Navier-Stokes equation, the first law of thermodynamics, and the equation for mass conservation were solved and used for stratification prediction. Typical results are presented in Figure 11. These

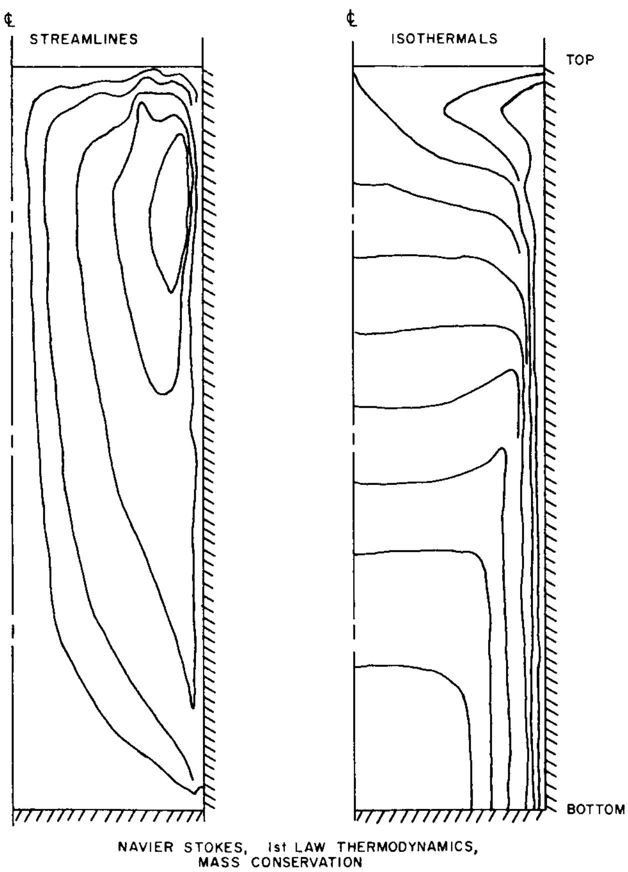


FIGURE 11. STRATIFICATION PREDICTION

results are preliminary and have no experimental verification. One interesting aspect of the data on streamlines is the apparent vortex located close to the tank wall and liquid surface.

F. FLUID DYNAMICS

Fluid dynamics problems of propellant feed systems are continually studied. Potential remedies for one such problem being explored, the so-called "Pogo" effect (the vibrational coupling of structure, propellant delivery system, and engine of a rocket vehicle), is discussed. Oscillatory forces may cause structural and propulsion problems as well as discomfort to astronauts. One potential solution to the Pogo effect is to change the acoustic velocity of the feed system by injecting a noncondensable gas; thus, frequencies of the feed system and structure are separated. Figure 12 shows how gaseous helium

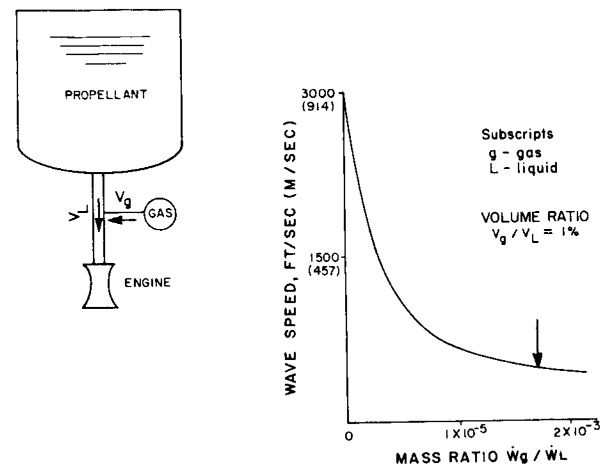


FIGURE 12. POTENTIAL SOLUTION TO FEED SYSTEM INSTABILITIES

injected into a column of liquid oxygen changes the acoustic velocity. In Figure 13, which shows the effect on frequency, the curve designated "system f" is based on actual data from a feed system including line, pump, etc. The large difference in frequency between the line and the complete feed system (Fig. 13) for low gas injection quantities results from a gas pocket or bubble near the pump inlet that accomplishes, to some extent, the same effect as helium injection into the line. This gas pocket is created by the pump and is not unusual. Gas injection significantly affects system frequency and offers a practical solution to the problem of undesirable oscillations. During developmental testing of the F-1 engine at MSFC, other oscillations were detected. Injected helium has shifted the frequency of these oscillations and reduced the amplitude. Gas injection below 2 percent by volume, the maximum tested, had no significant effect on engine thrust.

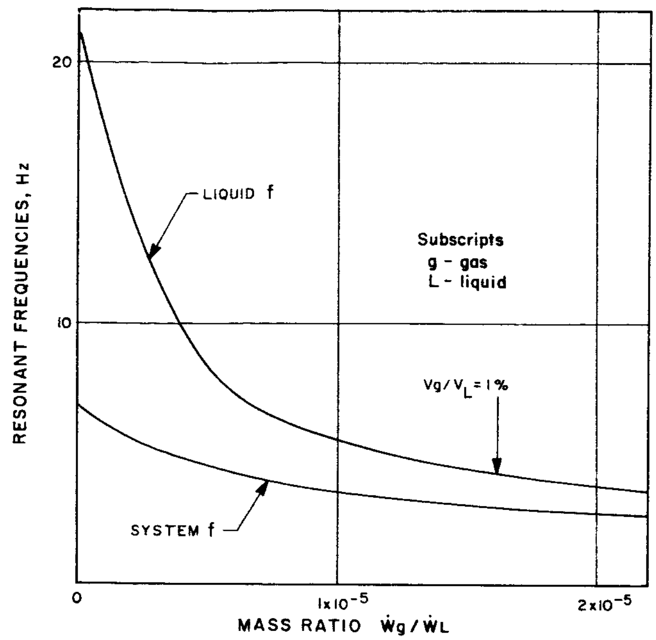


FIGURE 13. RESONANT FREQUENCIES VERSUS MASS RATIO

G. REDUCED GRAVITY FLUID MECHANICS

Figure 14 illustrates the equilibrium fluid configuration for a wetting fluid in gravitational fields of one and zero. Data are shown for two tank shapes and for varying amounts of propellants. Most research has dealt with equilibrium fluid mechanics for zero gravity and has ignored transient fluid

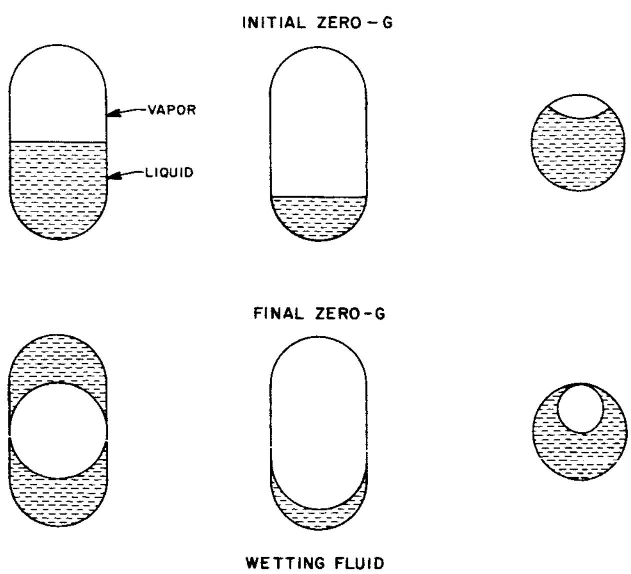


FIGURE 14. REDUCED GRAVITY FLUID MECHANICS

mechanics, the situation most likely to be encountered on a practical mission. Marshall Space Flight Center is conducting analytical and experimental studies of transient fluid mechanics and has the assistance of several universities and industrial groups.

The Saturn S-IVB stage, developed by Douglas Aircraft Company and MSFC, burns cryogenic propellants and must restart in earth orbit after exposure to the orbital environment for not less than 1.5 hours and not more than 4.5 hours. Because of the reduction in gravitational force from the boost flight to the orbital phase, pre-insertion disturbances of propellants are magnified significantly after insertion. Figure 15 lists some sources of fluid distrubance at insertion of the S-IVB into orbit, expected

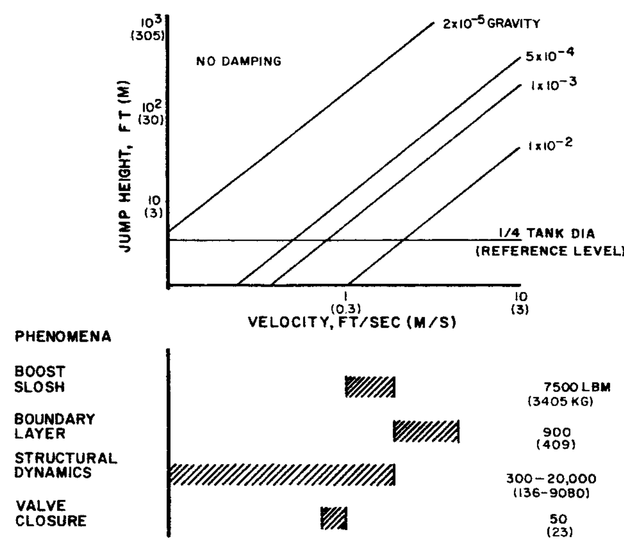


FIGURE 15. EXPECTED TRANSIENT PHENOMENA IN S-IVB

mass of hydrogen propellant involved, and velocity imparted to the propellant for each type of disturbance. The ordinate in Figure 15 shows propellant jump heights for the various disturbances; zero propellant damping and full conversion of all kinetic energy to potential energy are assumed. Gravitational forces provided on the S-IVB are $5 \times 10^{-4}\ g_o$ for approximately 100 seconds immediately after orbital insertion. Gravitational forces are then reduced to $2 \times 10^{-5}\ g_o$ and remain constant thereafter for earth orbit. These fluid motions are a potentially serious problem because hydrogen has low damping characteristics, and the hydrogen tank must be vented. As an example, a fluid motion possessing a vertical velocity of 1 ft/sec

(0.3 m/sec) during booster flight would jump 100 feet (30.5 m) at the sudden reduction of the gravitational field to 2×10^{-5} g_o. Although not listed in Figure 15, the attitude control system that limits vehicle attitude drift while in orbit is considered a major source of fluid disturbances. Firing in a random fashion, this system may create fluid disturbances during the entire orbital period.

Marshall Space Flight Center constructed a drop tower to study these problems, and experimentation began in mid-1965. Various features of the tower are shown in Figures 16 and 17. Free-fall distance for the test package is 294 feet (89.6 m), and the test duration is 4.3 seconds. Figure 18 illustrates

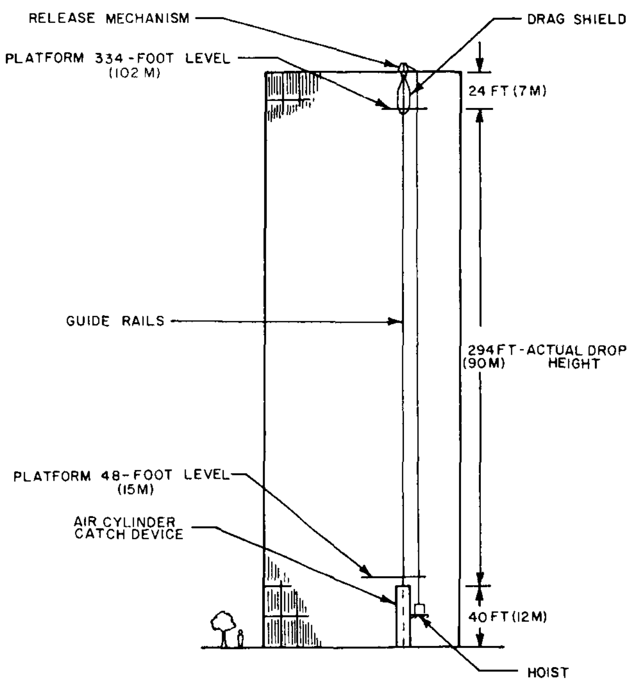


FIGURE 16. MSFC 300-FOOT DROP TOWER

typical subjects being explored through analyses and experimentation. An appropriate system design for the Apollo program requires adequate information about the following: time for propellant to travel along the propellant tank, type of propellant interface failure, time required to dampen fluid motion, optimum thrust for settling propellants, tendency for bubbles to collect on walls and in crevices, extent of propellant foaming and frothing caused by sudden tank pressure drop during venting, etc.

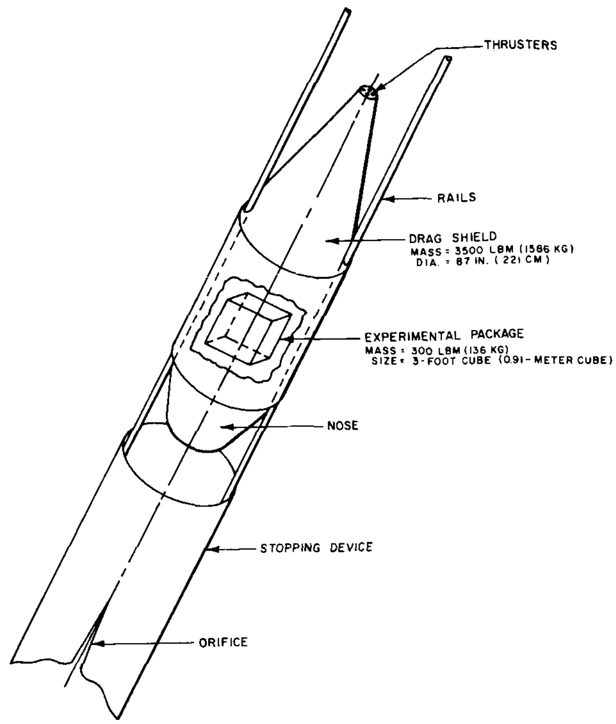


FIGURE 17. MSFC DROP TOWER DECELERATION DEVICE AND EXPERIMENTAL PACKAGE

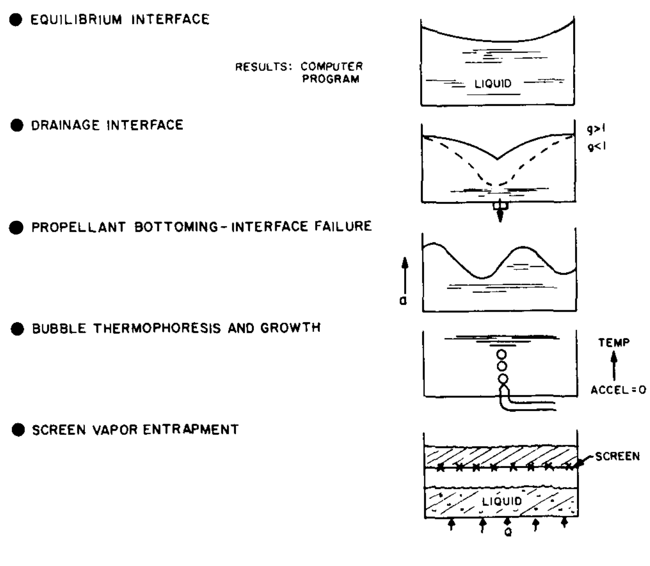


FIGURE 18. REDUCED GRAVITY FLUID MECHANICS STUDIES

H. DETONATION AND EXPLOSIVE HAZARD

Contributing to combustion hazards are propellants exhausted in the gaseous, liquid, or solid state from tank vents on engine chill systems. Vehicle systems designed to eliminate hazards on current vehicles are shown in Figure 19. The systems are heavy and complex, they degrade reliability, and they are costly. Available criteria for assessing hazards caused by oxygen and hydrogen mixtures in liquid, gaseous, and solid states are inadequate. Such criteria are being established under various degrees of vacuum and for various ignition sources. The investigation has only recently been initiated and results are not available.

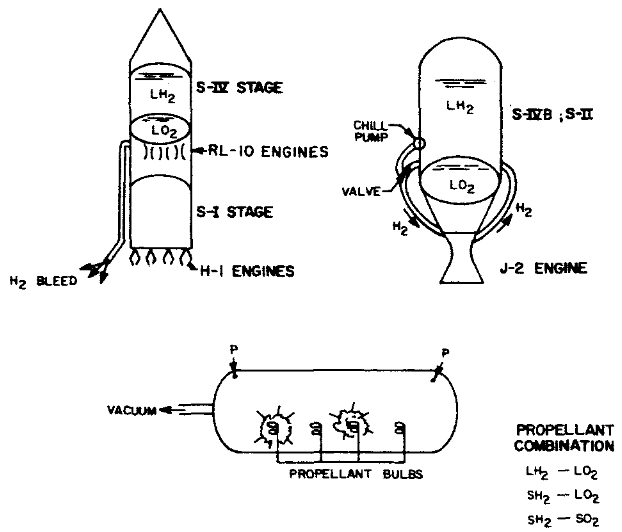


FIGURE 19. SYSTEMS TO ELIMINATE DETONATION AND EXPLOSIVE HAZARD

III. PROPELLANT STORAGE

The storage of cryogenic propellants involves many technical concepts, such as insulation for reducing heat flow into the tank, surface coating for reflecting incident energy, shadow shields for reducing energy available to the storage container, devices for reliquefying propellant boiloff, the optimum state for stored propellant (slush, jell, etc.), and a combination of these (Table I). Current research dealing with some of these concepts is discussed in this section.

TABLE I. CRYOGEN PROPELLANT STORAGE

INSULATION SYSTEMS	
Booster Vehicles,	6 hrs.
Spacecraft,	96 hrs.
Space Operations,	6 - 12 months
PASSIVE SYSTEMS	
Coatings, Shadow Shields	
ACTIVE SYSTEMS	
Refrigeration	
CRYOGEN STATE	
Slush, Subcritical, Supercritical	
SYSTEM INTEGRATION	

A. INSULATION

Insulation is arbitrarily classified according to three types: Booster insulation (6 hours maximum storage), spacecraft insulation (96 hours maximum storage), and space operation insulation (6 to 12 months storage). Booster insulation used on the hydrogen tank of several stages and the index to insulation performance for each (product of insulation density and thermal conductivity) are shown in Figure 20.

The Centaur insulation is outside the hydrogen tank and has a passage for helium purge between the insulation and the tank wall. The Centaur insulation can be jettisoned; thus, the effective insulation performance index, K, is dependent upon flight time for insulation jettisoning. Insulation that is not jettisoned must have a K-value between 0. 05 and 0. 12 to be as efficient as the Centaur insulation.

The S-IV and S-IVB stage insulations are inside the hydrogen tank. They consist of polyurethane foam reinforced with small fiberglass threads (analogous to reinforced concrete) and have a vapor or hydrogen barrier of fiberglass cloth coated with epoxy resin. Performance degradation occurs with use because of hydrogen permeation of the barrier. The insulation performance index is approximately half that of the Centaur insulation, a significant improvement.

The S-II stage insulation is external to the hydrogen tank and consists of fiberglass honeycomb filled with polyurethane foam. A composite of epoxy-impregnated nylon and Teflon constitute the outer

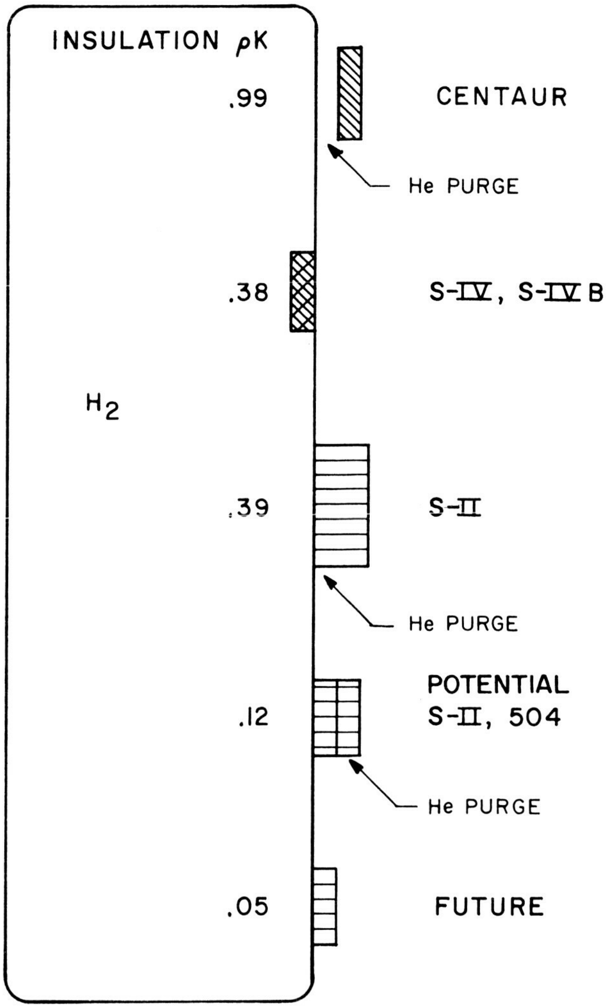


FIGURE 20. BOOSTER STAGE INSULATION

layer and vapor seal. The insulation is helium purged to eliminate hazards. Insulation performance for the S-II and S-IVB vehicles is equal. The S-II insulation considered for use on later vehicles, 504 and subsequent, will consist of a mylar honeycomb sealed inner layer and a fiberglass honeycomb outer layer. The outer layer is merely a channel for the required helium purge gas. The intermediate layer and outer layer are aluminum foil. No foam is utilized. The performance index of this insulation, 0.18, is significantly less than that for any known insulation exclusive of the superinsulations. Figure 21 shows this insulation applied to a 24-inch (61 cm)-diameter calorimeter tank.

The availability of higher temperature materials permits development of an insulation that omits the outer channel portion of the S-II insulation. A performance index of 0.05 is believed to be possible for

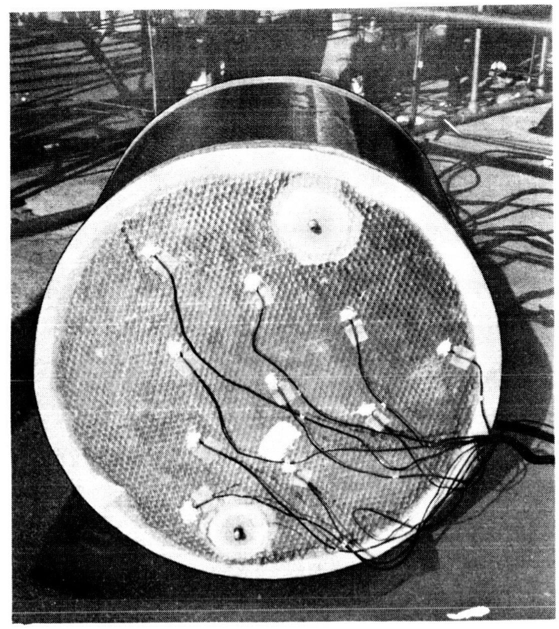


FIGURE 21. CALORIMETER WITH S-II STAGE, BACKUP 504-TYPE INSULATION

this insulation, a reduction factor of 8 from the current S-IVB insulation. Although significant progress in insulation for hydrogen containers has been made in the past few years, further improvements are imperative.

Spacecraft insulation of cryogenic vehicles for a 96-hour mission requires heat-leak reduction ratios of 150 to 500 relative to the S-IVB insulation. Marshall Space Flight Center is engaged, as are other organizations, in an insulation development program for realizing this goal (Fig. 22). This program includes significant contractor participation. Goodyear Aircraft Corporation, for example, is conducting basic research aimed toward new insulation systems that will be simple to apply and that will provide for meteoroid protection.

Lockheed Aircraft Corporation is attempting to improve existing computer programs and to devise new programs for predicting thermal system performance. The tank shown in Figure 23 is used as a model for the Lockheed work on helium-purged superinsulation. A typical propellant boiloff curve for hydrogen, for tank volumes comparable to that of the cryogenic service-module-type tank, is shown in Figure 24. Also given are ambient and internal pressure of the insulation versus time for a typical mission. Tests to verify insulation evacuation rates and thermal performance for various mission profiles, insulation concepts, and designs will be conducted in the test chamber illustrated in Figure 25. This chamber uses radiation barriers and controlled

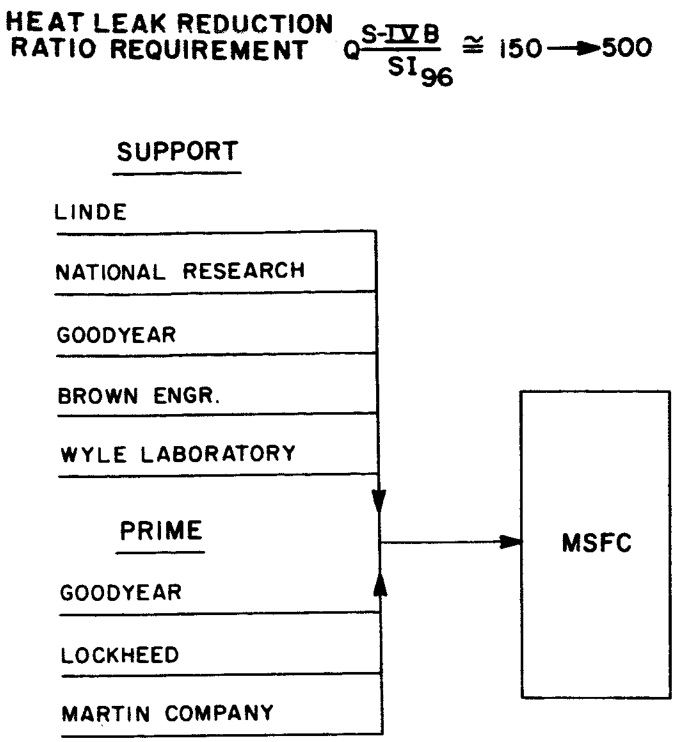


FIGURE 22. CONTRACTOR PARTICIPATION IN SPACECRAFT INSULATION PROGRAM FOR 96-HOUR MISSION

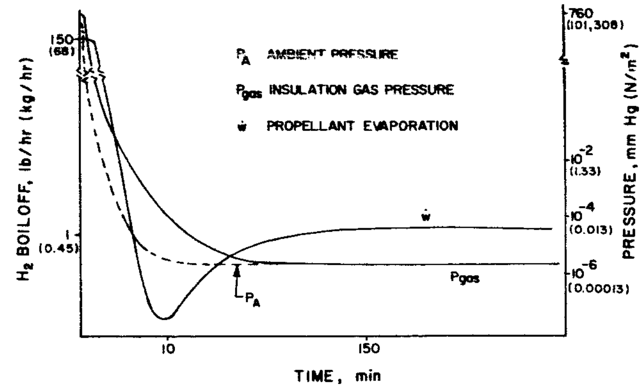


FIGURE 24. PREDICTED PROPELLANT EVAPORATION RATES AND INSULATION PRESSURES FOR SUPERINSULATED TANK

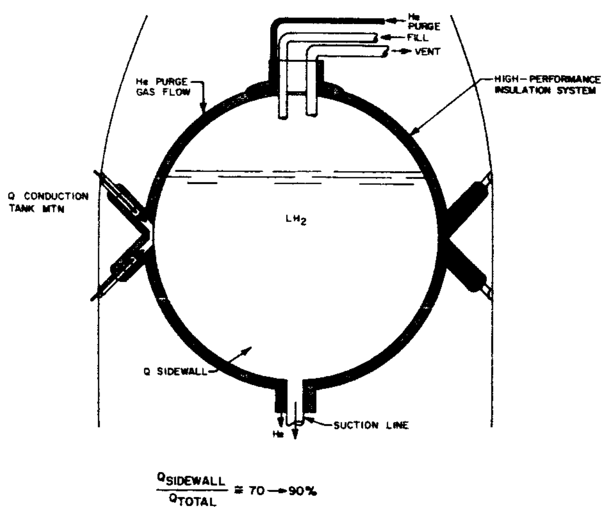


FIGURE 23. TANK MODEL FOR SUPERINSULATION RESEARCH

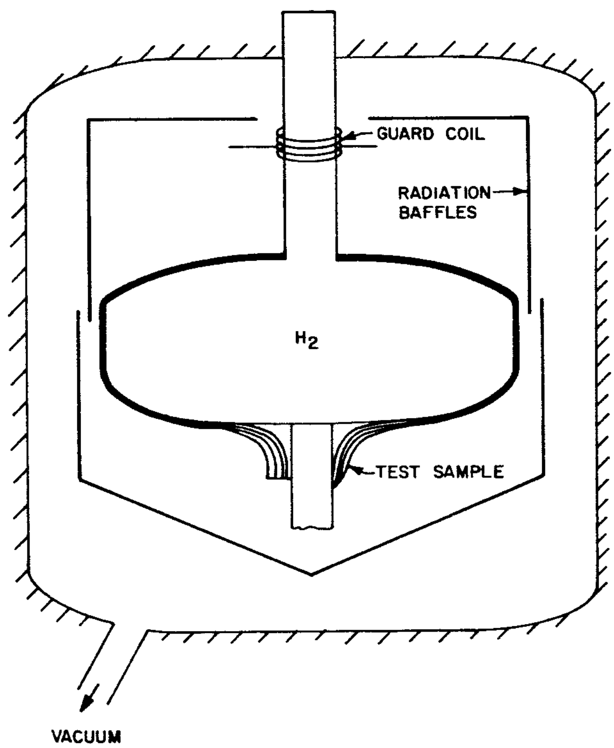


FIGURE 25. THERMAL PERFORMANCE TEST CHAMBER

conduction heat leaks for accurate thermal assessments of individual components.

Martin-Marietta Corporation is studying structural and thermal integration of insulation systems and will demonstrate performance for the selected concept at Marshall Space Flight Center, using a 105-inch (267 cm) diameter tank.

The MSFC in-house program uses two commonly known superinsulation concepts (Fig. 26), the National Research Corporation's (NRC) aluminized mylar radiation shield enclosed within a flexible jacket and

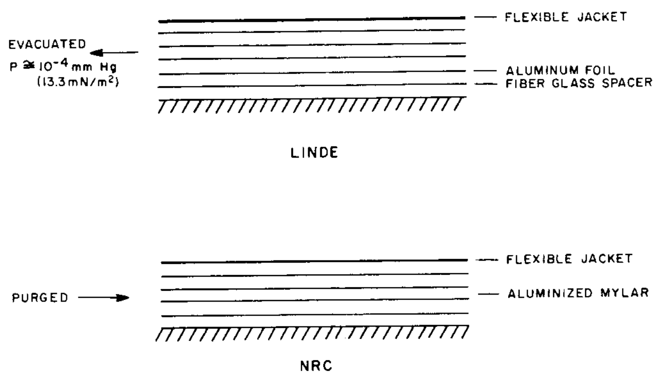


FIGURE 26. SUPERINSULATION CONCEPTS

purged, and the Linde Corporation's alternating layers of fiberglass and aluminum foil enclosed within a flexible jacket and evacuated. The milestone of the program is the demonstration of adequate thermal and structural performance for an insulated 105-inch-diameter flight configuration tank as required for a 96-hour mission (Fig. 27).

There are many supporting programs in progress, and many have been completed. Typical component programs are flexible jacket material development, jacket fabrication, insulation evacuation rates, and penetration wrapping techniques. The program status for the 105-inch-diameter tank is shown in Figure 28. The performance of the tank insulated with Linde SI-62 and tested in 1964 was unsatisfactory. Component tests have been initiated and are virtually complete, thus permitting this concept to be reapplied to the 105-inch-diameter flight-configured tank. The NRC insulation purged with helium gas (concept 2 of Figure 28) has relatively high propellant boiloff during ground hold (Fig. 24). This system has been successfully tested. Insulation concept 3 (Fig. 28) uses a sublayer of insulation for reducing ground boiloff and is expected to perform the same as concept 2 at the low pressure in orbit. Application of insulation concept 3 to the 105-inch tank depends upon successful development of the mylar honeycomb sublayer for the tank bulkhead regions. Concept 4 will be established for the work of Martin-Marietta Corporation, and cannot be defined at this time.

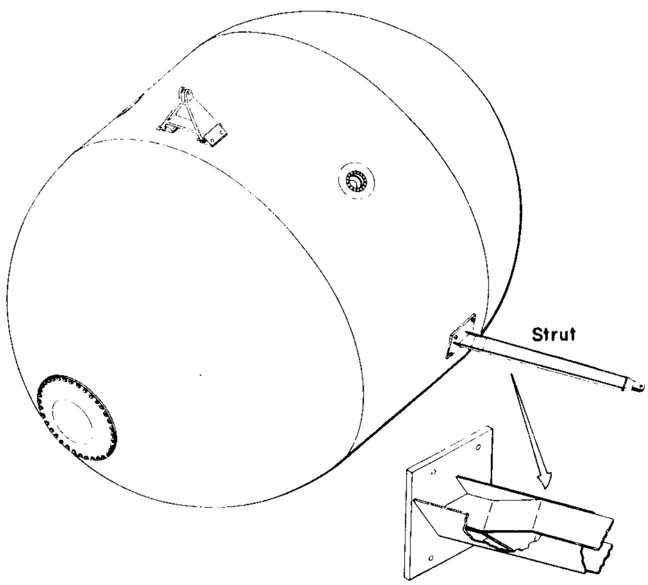
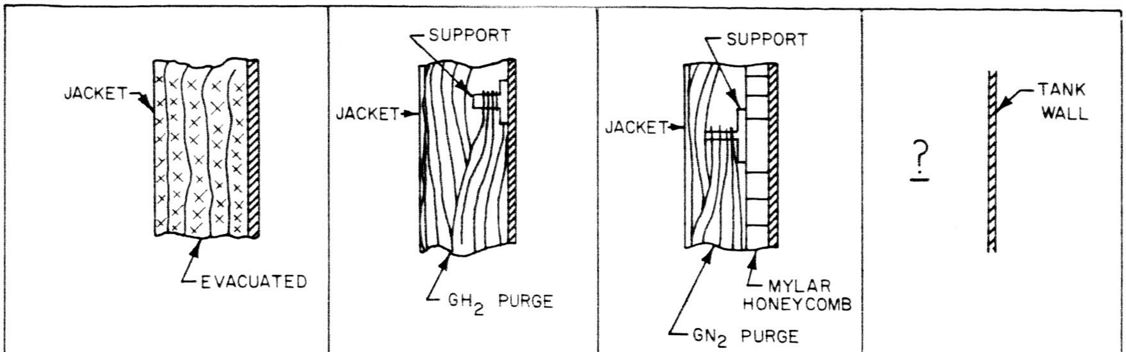


FIGURE 27. MSFC 105-INCH TANK ASSEMBLY

Tests are being conducted for both ground standby and orbital conditions. Figure 29 shows the vacuum chamber test facility used to simulate the orbital environment. Acceptable insulations will be tested under expected flight accelerations (Fig. 30), and retested for thermal performance at the expected orbital environment. Figures 31 through 33 show various insulated components, and Figure 34 shows the insulated tank.

Cryogenic propellant storage durations of 6 to 12 months or longer require heat transfer reductions between 2500 and 8500 relative to the S-IVB-type insulation. The present state of the art for long-duration storage of small quantities of propellant (liquid hydrogen and liquid oxygen), thermal requirements for the Gemini and Apollo spacecraft, and a possible Molab design (established during feasibility studies) are shown in Figure 35. Significant advances in technology are required to obtain the established Molab heat-leak requirement of approximately 2 Btu/hr (0.586 W) for each of the liquid-hydrogen and liquid-oxygen containers. Both MSFC and Manned Spacecraft Center (MSC) have initiated development programs for long-term storage of cryogenic propellants. In studies at MSFC, superinsulation is used for reducing the heat leak, while at MSC discrete radiation shields are used. Results of



	MSFC (1) LINDE	MSFC (2) NRC, SHINGLES	MSFC (3) NRC, SHINGLES	MARTIN (4) DECISION PENDING
SUMMARY				
INSULATED	1964	CURRENT	1966	1966
TESTED	1964	JAN 1966	1966	1966
CURRENT EFFORT	TANK RETEST-1966	TANK AND COMPONENT	TANK AND COMPONENT	COMPONENT
MAJOR PROBLEMS	● JACKET LEAKS ● WRAPPING PENETRATIONS ● INS. RECOVERY	● GROUND HEAT LEAK ● SLOW EVACUATION	● BONDING MHC TO CURVED SURFACES	?

FIGURE 28. IN-HOUSE PROGRAM STATUS

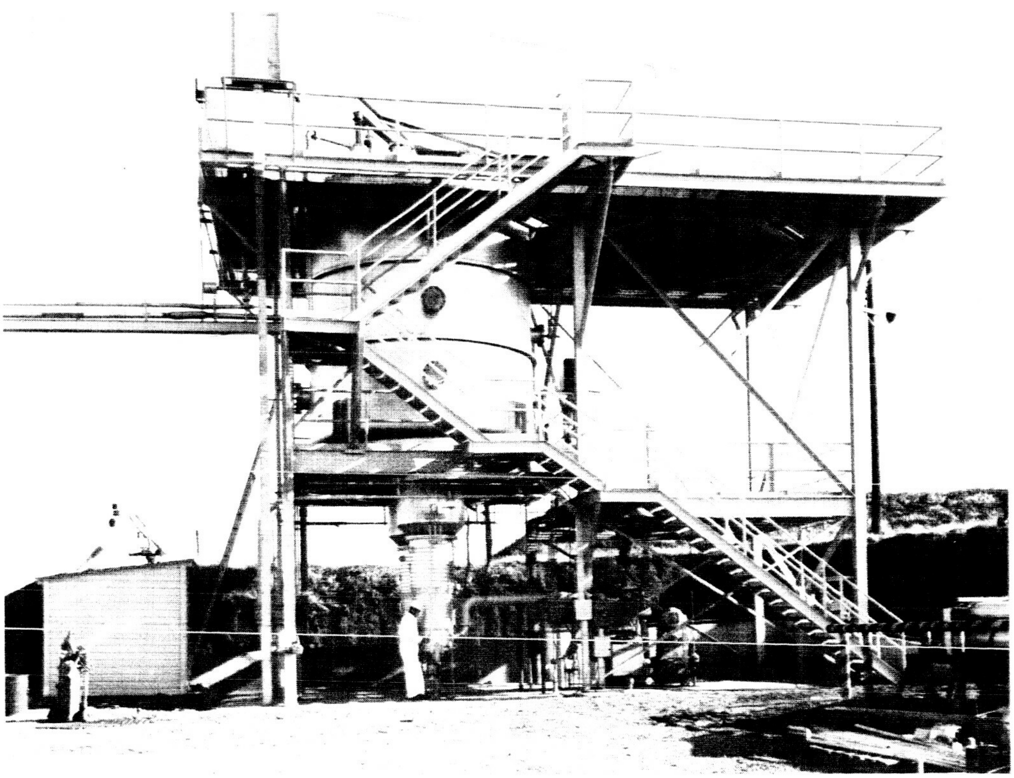


FIGURE 29. VACUUM FACILITY AT MSFC

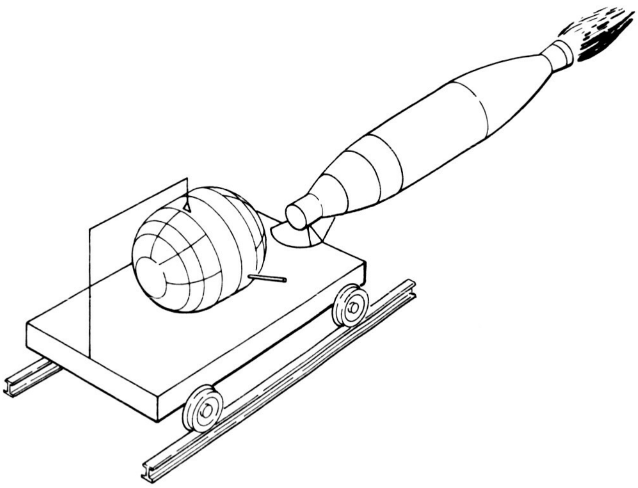


FIGURE 30. ACCELERATION TEST

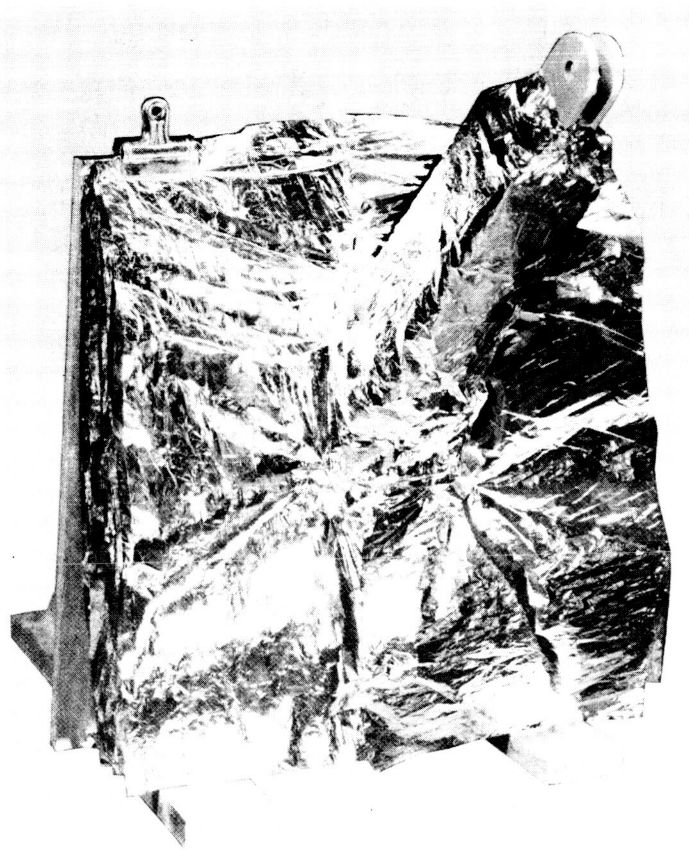


FIGURE 32. INSULATED SUPPORT-STRUT MOCK UP

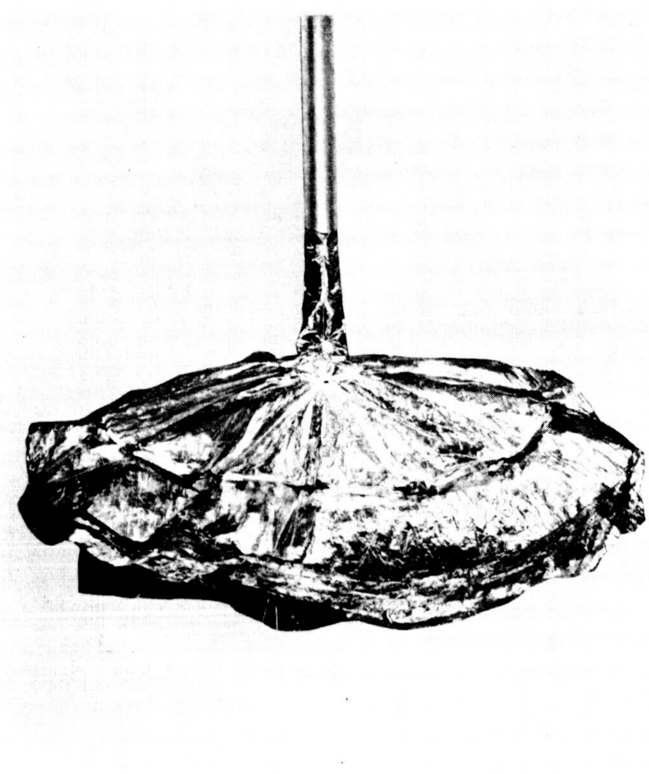


FIGURE 31. INSULATED MANHOLE COVER

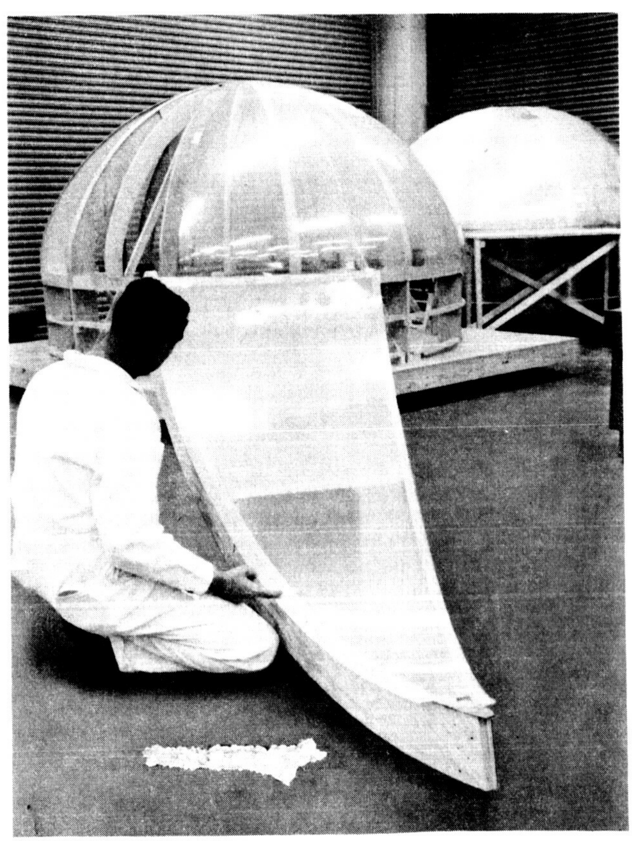


FIGURE 33. FLEXIBLE BAG MANUFACTURING

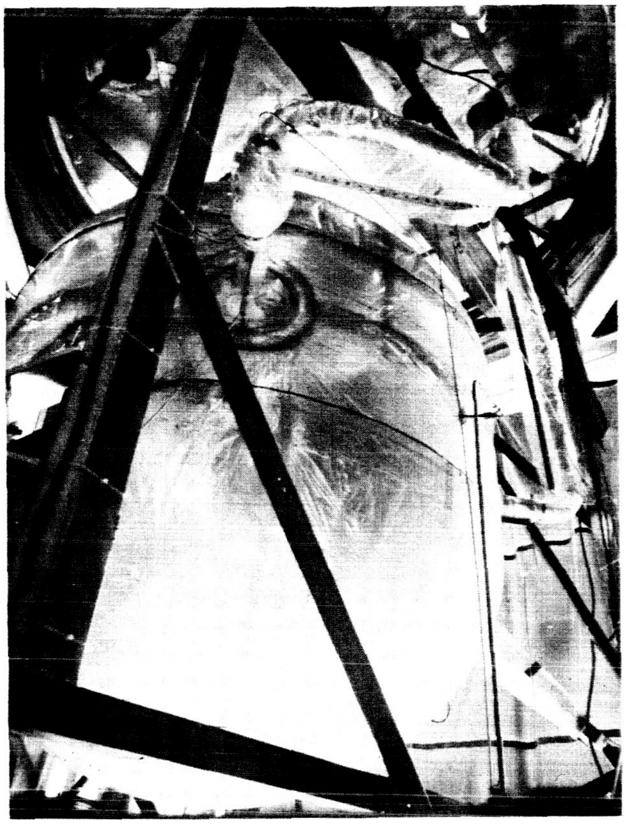


FIGURE 34. INSULATED TANK IN TEST CHAMBER

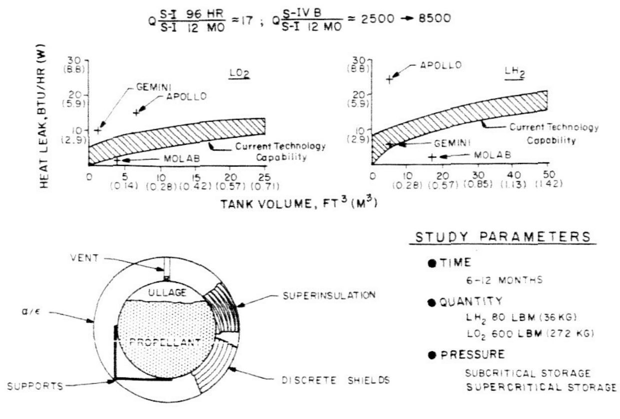


FIGURE 35. CRYOGENIC STORAGE (6-12 MONTHS)

preliminary analyses that consider nonvented liquid-hydrogen storage (Fig. 36) indicate that approximately 600 lbm (272 kg) of system mass is required per 100 lbm (45 kg) of usable liquid hydrogen for

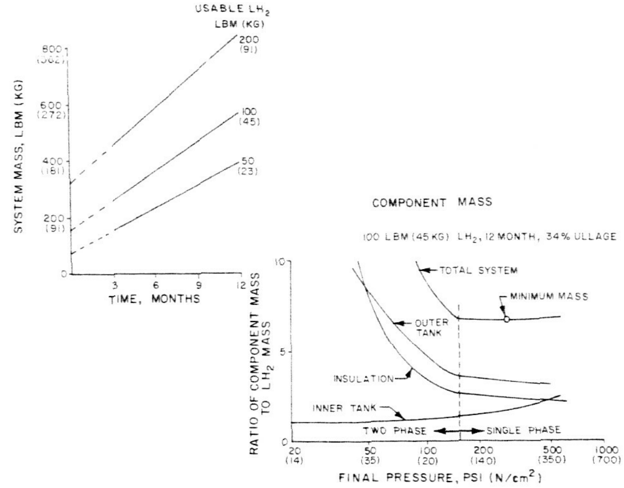


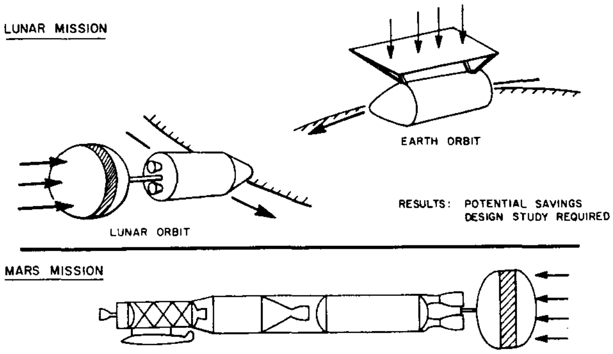
FIGURE 36. LH_2 NONVENTED LUNAR STORAGE SYSTEM

storage durations of 12 months. The required insulation thickness is approximately 12 inches (30 cm), and the primary mass-contributing component is the outer shell. Numerous other analytical studies are in progress, including storage without the nonvent requirement, system optimization, liquid-oxygen storage, and others.

B. SHADOW SHIELDS

Another approach to cryogen storage in space uses shadow shields for reducing the energy incident on the storage vessel. Results of a recently completed feasibility study of an inflatable shadow shield are shown in Figure 37. Inflatable shadow shields were optimized and analyzed for a typical Lunar Orbit Rendezvous (LOR) mission with a cryogenic service module and for a manned Mars landing mission. The shadow shield configurations depend on mission phase; and such factors as basic shape, optical coating, development and storage mechanism were optimized.

Distinct weight advantages are not obvious for the lunar mission; however, shadow shields showed payload savings in excess of 5000 lbm (2268 kg) for the Mars mission. Effective shield design protects against direct solar radiation but does not completely exclude planetary albedo and planetary thermal emission. Therefore, the thermal effectiveness of a shadow shield is reduced during the LOR mission and during the Mars orbit phase of the manned Mars landing mission. However, system mass penalties with and without a shield are comparable during planetary orbit phases, and a detailed design analysis,



SUPERINSULATION MASS PENALTIES, LBM (KG)

STAGES	SHIELDED	UNSHIELDED
• MARS BRAKING	2281 (1033)	5295 (2399)
• MARS ESCAPE	4507 (2042)	4897 (2218)
• EARTH BRAKING	1140 (516)	2563 (1161)

FIGURE 37. SHADOW SHIELDS (INFLATABLE)

including thermal integration of all vehicle systems, is required before final vehicle design.

Conventional and high-performance insulation systems were considered in conjunction with shadow shields. Shield configurations were optimized and preliminary design studies were used to establish system mass comparisons. Structural design criteria and properties of particular thermal coatings were determined in a test program. The importance of inflatable shadow shields was established for future cryogenic vehicles.

C. CRYOGENIC RELIQUEFACTION/REFRIGERATION

Cryogenic refrigeration studies have been in progress for two years. Initial studies were limited to reliquefaction of stored hydrogen boiloff on the lunar surface. Recent studies have considered both hydrogen and oxygen reliquefaction on the lunar surface and in earth orbit. Figure 38 shows, in simplified form, the basic elements of the systems considered. The prime energy source for lunar systems operation is electricity from the nuclear auxiliary power systems (SNAP). Design investigation was based on hydrogen boiloff of 1 lb/hr (0.5 kg/hr) from a 20-foot (6 m) spherical superinsulated storage tank containing 19,600 pounds (8800 kg) of liquid hydrogen. The reliquefier would operate only during the lunar night to take advantage of the lower effective sink temperature (thus, there would be higher cycle efficiency). During the lunar day, the pressure in the storage tank would be allowed to rise approximately 5 psi (34 kN/m^2). Propellant

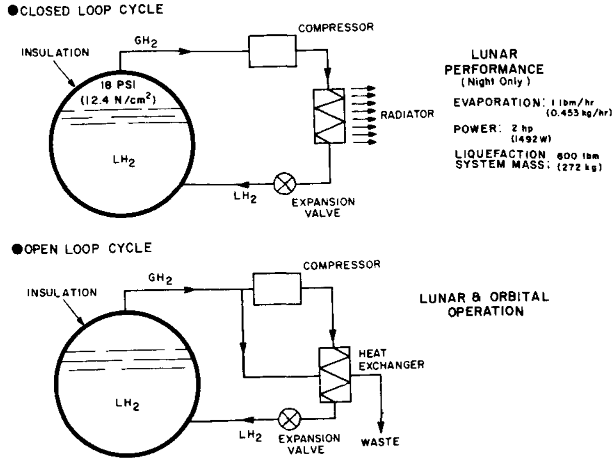


FIGURE 38. H_2 RELIQUEFACTION

boiloff during 12-month storage without reliquefaction would be approximately 2200 pounds (1000 kg). The estimated mass of the reliquefaction system is 600 pounds (272 kg); thus, the break-even point is approximately 4 months storage time. For smaller tanks, the boiloff rate would be less, although the ratio of boiloff to stored propellant would be more.

Preliminary studies of the closed-loop reliquefaction system for earth orbital operation showed excessive weight penalties due to greater radiator area and power requirements. As a consequence, open-loop systems are currently under investigation for these missions. Studies considering the energy of the stored hydrogen in open-loop systems show that theoretical limits of about 60 percent reliquefaction can be reached. Realistic reliquefaction percentages and systems weights are being determined.

D. THERMAL INTEGRATION

A study to determine vehicle thermal integration criteria for interplanetary travel is under way. This study (Fig. 39) considers effects of surface coatings, shadow shields, insulation, refrigeration, and required acceleration for propellant control. Thermal integration criteria for each of the following flight missions are being established: (1) unmanned Mars orbital reconnaissance, (2) unmanned Venus orbital reconnaissance, (3) manned Mars flyby, and (4) manned Mars landing. The importance of this study cannot be overemphasized, and it should establish a guide for future research efforts and vehicle configurations.

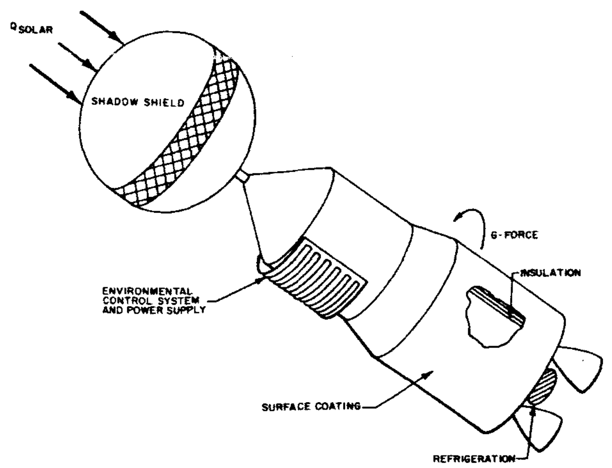


FIGURE 39. VEHICLE THERMAL INTEGRATION CRITERIA

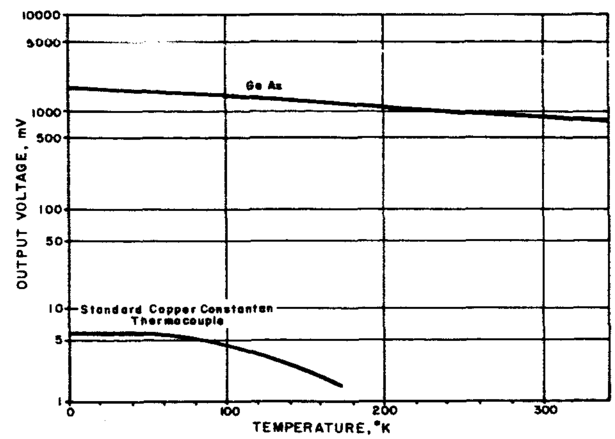


FIGURE 40. GALLIUM ARSENIDE DIODE SENSOR

IV. INSTRUMENTATION

Many commercially available temperature, pressure, and point-density sensors, and liquid-level systems, are being tested at MSFC to obtain background information that will facilitate proper equipment selection for each application. The programs are far too numerous for detailed reporting in this paper. Instrumentation research and development programs to provide improved instrumentation for special requirements are in progress. Some typical programs are discussed in this section.

A. TEMPERATURE SENSOR

The performance of a gallium arsenide diode sensor is compared with that of a standard thermocouple in Figure 40. Response time of the gallium arsenide diode sensor in the liquid hydrogen temperature region, where standard thermocouples have poor response, is 0.5 second to produce 63 percent of the total temperature change when the probe is extracted from the liquid and exposed to circulating gas.

B. FIRE DETECTION AND WARNING

A fire detection and warning system under development is illustrated in Figure 41. The system compares rocket plume radiation, solar radiation, and radiation from within the compartment that contains the system. The detection system discriminates between OH^- radicals, ultraviolet bands, and the flicker frequencies to differentiate a fire, the rocket

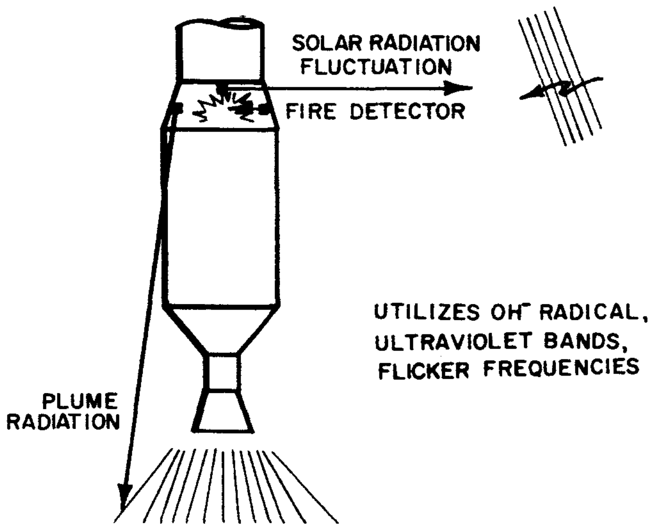


FIGURE 41. FIRE DETECTOR AND WARNING SYSTEM

plume, and solar radiation. The system either warns of a hazard or initiates the operation of protection devices to combat the hazard.

Two other hydrogen detectors under development are shown in Figure 42. The polarographic detector, restricted to the prelaunch pressure environment, is useful for hydrogen concentration ranges of 0.01 to 98 percent and has a response of 60 milliseconds. The ultrasonic detector is applicable for concentrations ranging between 0.01 and 0.1 percent hydrogen for any pressure environment. Preliminary results suggest that this detector may be thermally unstable. The polarographic detector is a disk approximately

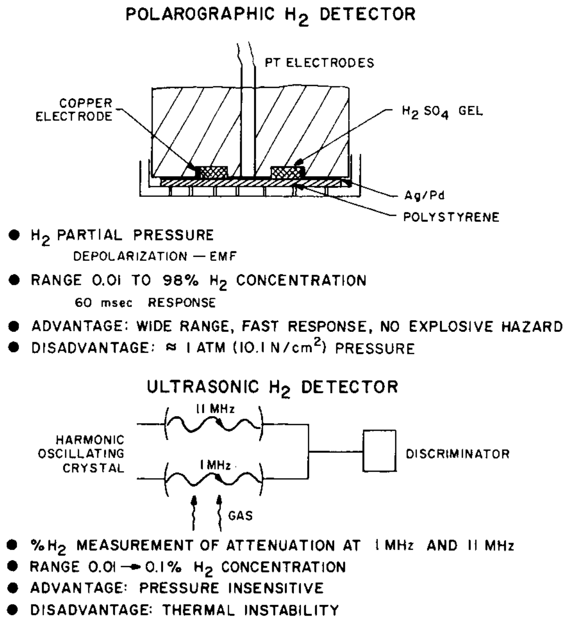


FIGURE 42. POLAROGRAPHIC AND ULTRASONIC H_2 DETECTORS

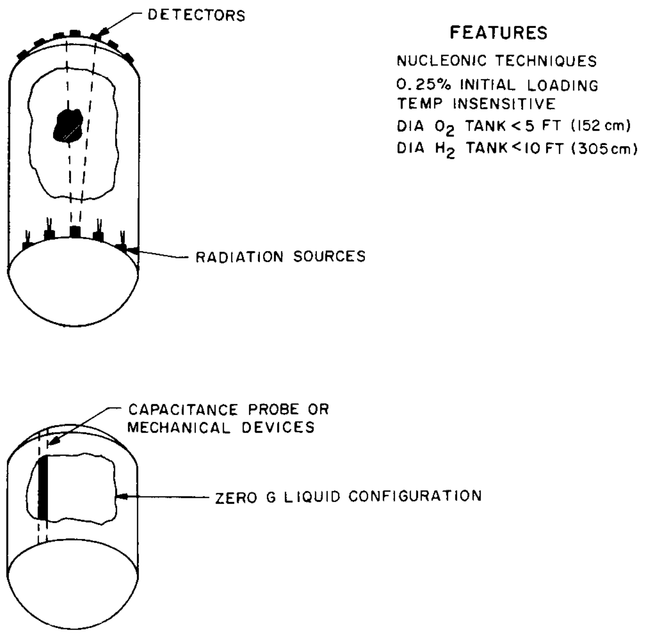


FIGURE 43. PROPELLANT MASS MEASUREMENT

$1\frac{1}{2}$ inches (3.8 cm) in diameter and $\frac{1}{4}$-inch (0.64 cm) thick. In this detector, the hydrogen gas permeates a polystyrene or similar membrane and depolarizes the two electrodes to establish an electric potential. The hydrogen concentration can be established from the measured electromotive force and appropriate device calibrations.

The ultrasonic detector uses two harmonic oscillating crystals and discriminators. The influence of hydrogen on attenuation at 1 MHz and 11 MHz can be used to establish the hydrogen concentration. This principle of operation can also be used for detection of other gases.

C. PROPELLANT MASS DEVICE

Propellant mass determination has always caused problems in launch vehicle design and is expected to cause even more difficulties for vehicles operating in a reduced-gravity environment. The existence of small surface-tension forces of the fluid (in reality large forces relative to gravitational forces) will result in liquid collection in the often-used capacitance probe or similar device. Studies indicate that such difficulties can be overcome through the use of nuclear techniques; therefore, research and development efforts with nuclear systems are in progress. In Figure 43 one or more gamma radiation sources, about one-half curie (1.85×10^{10} disintegrations/sec) each, are on the common bulkhead of the propellant tank, and the detectors are on the opposite tank bulkhead. For defined propellant location, propellant mass is estimated to be predictable within 0.25 percent accuracy. For random propellant orientation, propellant mass determination within 1 percent is considered possible. The nuclear-principle technique appears to offer the additional advantage of reduced weight. An earth orbital experiment is being designed for exploration of the system at near zero gravity. The feasibility for early verification of a complete system on the S-IVB vehicle is also being established.

D. FLUID QUALITY METER

The same nuclear principle (as applied to the propellant mass device) was studied for the quantitative assessment of the proportion of gas within a liquid (Fig. 44). A device for installation within a vent pipe of a liquid-hydrogen container has been designed and tested. The device is accurate to ± 5 percent for vent mixtures of gas-to-liquid ranging between 0 and 100 percent. The device constitutes a vital portion of the instrumentation for a large hydrogen tank to be launched into earth orbit for studying fluid behavior under reduced gravitational forces. Similar devices are currently being calibrated to measure propellant quality in vehicle suction lines.

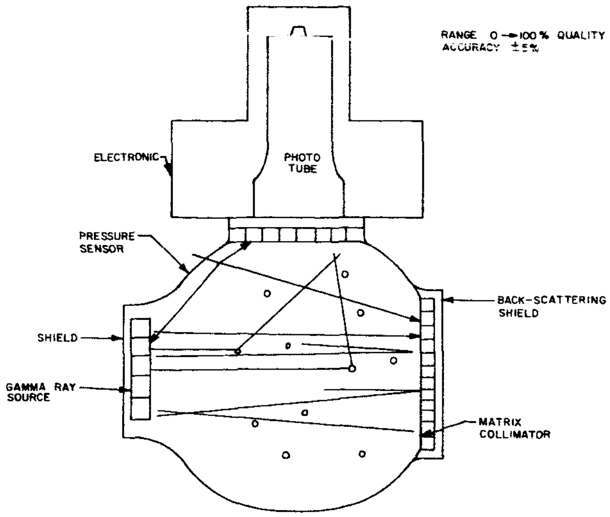


FIGURE 44. FLUID QUALITY METER

V. CONCLUSIONS

Technology was inadequate during principal design periods for current vehicle programs. Deficiencies existed in knowledge of booster insulation schemes, fluid behavior, and instrumentation and detection techniques. Should a new vehicle program be undertaken today, deficiencies in technology would be as significant as those that existed at the start of current vehicle programs. Much work remains in these major areas: propellant storage; cryogenic fluid behavior; integration of thermodynamics, propulsion, and structures; and advanced instrumentation and detection methods.

APPROVAL

RESEARCH ACHIEVEMENTS REVIEW SERIES NO. 3

A Review of Cryogenic Technology Research at Marshall Space Flight Center

By C. C. Wood

The information in this report has been reviewed for security classification. Review of any information concerning Department of Defense or Atomic Energy Commission programs has been made by the MSFC Security Classification Officer. This report, in its entirety, has been determined to be unclassified.

This document has also been reviewed and approved for technical accuracy.

W. R. Lucas

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Director, Propulsion and Vehicle Engineering Laboratory