Orbiter Vehicle Hydraulic Power System. At least two APU-hydraulic

systems must be operational to assure safe return of the crew and

vehicle. Operational flight control requirements for the Orbiter

for the approach and landing phase can be met with any one of the three

APU systems failed. With two systems failed, the remaining system

with overspeed cannot meet all operational requirements and may not,

therefore, be capable of returning the crew and vehicle safely under

all mission design conditions.

The forward RCS provides precise attitude control and three-axis

translation during separation from the External Tank, orbit insertion,

and orbital phases of the flight. The aft RCS does all of these same

functions in conjunction with the forward RCS and also provides thrust

for the reentry phase of the mission. The forward RCS has eleven pri-

mary and two vernier thrusters mounted under doors and six thrusters

mounted exposed. The doors remain closed and latched during boost and

reentry phases and are deployed and locked in place for ET separation,

orbit insertion and orbital phases. The aft RCS is composed of twelve

primary thrusters and two vernier thrusters located on either side of

the aft Orbiter fuselage for a total of 24 primary and 4 vernier units.

The primary RCS engine specification requires the engine to in-

corporate a burn-through detector to sense an incipient thrust chamber

burn-through and to provide an appropriate signal to be used by engine

shutdown. This is a difficult item to develop and qualify and may also

cause operational problems due to false shutdown. It is now considered that burn-through is not one of the primary failure modes. The contractor was asked to process a Master Change Record (request), MCR, to delete the burn-through detector per the 102 PDR (February 1975).

The Orbital Maneuvering System (OMS) provides the propulsive thrust necessary to perform the following maneuvers: (1) final velocity increment for orbit insertion, (2) orbit circularization, (3) orbit transfer, (4) rendezvous, and (5) de-orbit. Although one OMS engine could be used for these operations, reliability considerations dictate that the loss of an OMS engine is cause for abort.

The OMS has single failure points in the pressurization and propellant feed areas and the failure mode would be rupture and excessive leakage. Any excessive pod differential pressure could result in structure and TPS damage preventing safe reentry. The OMS is fail safe otherwise, except for such catastrophic events as engine or propellant explosion.

Current Status

There are numerous mechanical connections used on the forward and aft RCS in lieu of welded connections. This approach permits removal and installation of equipment in minimum time while minimizing contamination hazard to the remaining portion of the system. Where possible the fittings and seals being used were already qualified in the same application in Apollo and Skylab programs. After reconnect

all mechanical connections will be pressurized to system pressure with helium and externally leak-tested to system requirements.

NASA and contractor have agreed to maintain tight surveillance of mechanical connections (fittings) to assure both the number and possibility of leakage are minimized.

Verification of component propellant compatibility of OMS/RCS hardware is under review. Based on the demonstrated Apollo CSM experience, the current requirement is that components be constructed of materials with demonstrated propellant compatibility. However, sub-system design features and operational methods, as well as program funding limitations precludes compatibility testing at the component level of the OMS high pressure helium isolation valve, helium pressure regulator, low pressure vapor isolation valve, and the tank pressure relief valve.

In the RCS the plan is to authorize only those materials in the helium system where there is proven compatibility with the propellants. The data and analysis will be accomplished during the development and qualification programs. Because of the propellant system components total exposure to liquids, a qualification compatibility test will be conducted at the subcontractor level.

Deletion of the vibro-acoustic test of the forward fuselage has meant cancellation of the vibration test of the forward RCS module. However, the need for system certification of the RCS prior to first

verticle flight has not been eliminated, so a reassessment of means and techniques is underway to provide the required certification data base. Plans are to review aft pod vibro-acoustic tests, system similarity and analytic techniques to see if aft pod data can be extrapolated for application to the forward RCS module. In addition, alternate forward module test plans and schedules are being studied to determine a cost effective vibration test for the forward module only. Resolution of these alternatives and a recommendation is due around 1 July 1975.

3.1.1.4 Avionics

Systems Design

The avionics subsystems provides commands, guidance and navigation and control, communications, computations, displays and controls, instrumentation, and electrical power distribution and control for the Orbiter, external tank and the solid rocket booster. The avionics are configured to facilitate checkout, access, and replacement with minimal disturbance to other subsystems. Equipment locations are shown in Figure 12 .

Computations or data processing is accomplished through the use of five digital computers. Three are dedicated to the guidance and navigation function. One can be used for either guidance and navigation or payload and performance monitoring, and one is dedicated to payload and performance monitoring. Software or computer programs are integral to this data processing and control system since these five general purpose computers are the same mode. It is the resident software that

determines the computer function.

Verification of the avionics/software systems as an independent and integral part of the Orbiter/Shuttle system is accomplished through the following test programs:

(a) Software Development Laboratory program to verify the flight data on flight computers.

(b) Avionics Development Laboratory program to verify "single string" and redundant hardware system operation and the hardware/software compatibility.

(c) Shuttle Avionics Integration Laboratory (SAIL) program to verify redundant hardware system operation for Orbital Flight Test as well as the hardware/software compatibility for OFT.

(d) Simulations to verify flight crew operations of vehicle and the guidance and navigation performance accuracy in a manner similar to simulations for prior manned spaceflight operations.

(e) Approach and Landing Test (ALT) program using Orbiter 101 will be used to verify the aerodynamic capability of the Orbiter, the aerodynamic guidance and navigation performance, aerodynamic system integrated operation and the aerodynamic dependent software.

(f) Orbital Test Flight program to verify the total mission vehicle capability with avionics and associated software.

Orbiter 102 will have the following avionics elements not on Orbiter 101·

(a) Startracker/Light Shield

(b) Those portions of the flight control system that involve the Reaction Control System, Orbital Maneuvering System, Thrust Vector Control for the SSME's.

(c) SSME interface unit portion of the system for processing engine data.

(d) Many items of the communications and tracking system, e.g., KU band radar, payload interrogator, signal processes, portions of the S-band, etc.

Current Status

The relationship of avionics to the flight and ground crew safety is multifaceted, since every action and reaction during the mission is controlled to some extent by the avionics system. The Panel has, therefore, had to be selective. We have chosen to review three areas most significant to crew safety: (1) Orbiter/SSME-Controller interface, (2) ALT/OFT flight control modes, and (3) abort operations.

A review by the Panel was to determine if there are potentially critical failures across the Orbiter/SSME interface, and, if so, to understand those steps being taken to minimize or eliminate such efects. Where hazards are not eliminated we wanted to assure that the assessment of the risk and the rationale for accepting it had been given appropriate management attention.

Operational and checkout commands and engine flight data are

supplied via the electrical interface connectors, at the engine-supplied electrical interface connect panel. Commands consist of engine start, shutdown, thrust level changes, checkout, and sequence checks. Engine flight data transmitted to the vehicle consist of information necessary for malfunction display, fault isolation, maintenance recording, trend analysis, performance monitoring and checkout. Three parallel redundant connectors provide a reliable path for the Orbiter to engine commands. Further a minimum of two of the three commands must be received before the engine response will be initiated. Two of these connectors are also employed to transmit the engine flight data back to the Orbiter. Failure to provide correct command during ascent or to transmit engine performance back to the Orbiter do not appear to be a direct threat to the crew safety since the engine will continue to operate on the last correct command received.

Flight control utilizes automatic commands determined by the guidance and navigation subsystem manual commands provided by the crew, vehicle motion sensed by the sensors, logic decisions processed by the control laws, and those forces produced by actuation of the aerodynamic surfaces TVC's, RCS, etc. to perform stabilization and control. The control laws are software. The flight control requirements for each mission phase (ascent, on-orbit, reentry, and atmospheric) are specified in terms of control mode elements. These mode elements or control modes are the building blocks which can be used in combi-

nations to provide the actual operational control modes. During

ascent through the SRB staging the nominal baseline has been de-

fined as automatic mode. While there is manual redundancy it will

not be used unless there is a significant benefit. After that

portion of the ascent period, the flight control modes can be (1)

manual direct, (2) manual command augmentation, (3) hold, (4) select,

and (5) automatic. These are defined in Table V . One of the areas

being worked by the program that will be examined by the Panel is the

identification of OFT launch failures which require manual guidance

and control. Another area is the aerodynamic tolerance effects on

response and stability of the flight control/structures design cap-

ability. Structural constraints have been reflected back in a manner

which indicates a need to restrict the angle of attack and side-slip

variations to a minimum consistent with ability to provide for high

aerodynamic load relief. Systems studies have indicated that these

constraints are only marginally reached with nominal system para-

meters. Flight control margins are tight and vehicle dynamics are

pushing the margins (plus/minus tolerances or limitations on system

input/output lag, accelerations, roll rates, etc.). The first stage

ascent is the period of greatest concern from the standpoint of com-

puter cycle time. There is a possibility that sample frequency re-

quirements may increase. If so, this would further aggravate the

computer timing problem.

The role of the avionics system in abort operations is particularly significant because of the need for large quantities of information concerning the vehicle and its performance as well as the need for fast reaction to on-going events. Confidence in the design capability of the Orbiter vehicle and its avionics subsystem to perform the once-around-orbit, return-to-landing-site or any other abort mode is being examined on a continuous basis as the design matures and the system capabilities are further designed. The Panel will examine this area in more detail as the concepts and design mature.

A back-up flight control system is being installed in Orbiter 101 only to provide protection against generic software problems or problems with the complex hardware, crew interfaces, and mechanization. No new hardware is anticipated. This approach should provide an additional measure of safety during the early flights of the ALT program.

This concern with overloading the computer capability in the Orbiter is real. It has been stated that at this time the word requirements are in the range of: ALT 2700-2800 words, OFT on-Orbit 2000-5000 words and entry 5000-6000 words (on orbit and entry are additive). The main drivers on the computer and the flight control requirements are speed and memory.

A number of flight control support tasks are being carried out by NASA Centers. Marshall is working on:

(a) Ascent flight dynamics and control.

(b) FCS requirements and constraints.

(c) Flight dynamics/stability performance.

(d) Body-mounted sensor complement and locations.

(e) Digital sampling/filtering and quantization.

Langley is working on:

(a) Entry guidance and control.

(b) Independent evaluation of flight crew role in controlling Shuttle.

(c) Orbiter G&C entry design verification.

The Flight Research Center is working on:

(a) Entry aerodynamic flight control, developing an F-8 digital fly-by-wire program for DPS and flight control redundancy management and flight control system design.

A number of avionics elements have not been placed on contract as yet or design has not evolved sufficiently to review it. The Integrated Electronics Assembly is not yet on contract. Many of the operational communications and tracking hardware will not be contracted for until 1976-77 period. This also holds true for display and control equipment for 102. Those areas, with safety implications, will be reviewed by the Panel at the appropriate time.

3.1.1.5   Electrical Power Subsystem

Systems Design

The electrical power subsystem generates the electrical power and is active throughout the vertical flight test program and operational flight and during ground operations when ground support equipment is not connected.

This electrical power subsystem is comprised of the power reactant supply and distribution and three fuel cell power-plants. The electrical power subsystem is shown schematically in Figure 13 . During peak and average power loads, all three fuel cells and buses are used; during minimum power loads, only two fuel cells are used but they are interconnected to the three buses. The third fuel cell is shut down but can be reconnected within 15 minutes to support higher loads. Excess heat from the fuel cells is transferred to the Freon cooling loop through heat exchangers.

Most of the active elements of the electrical power system have been designed to sustain two failures and remain operationally safe, in other words fail-operationally then fail-safe. The power reactant supply and distribution tanks, electrical power subsystem plumbing, and passive elements have been designed to provide fail-safe operation after a single failure by means of redundant subsystem flow paths which are physically separated. A single product water-line is provided to the environmental control and life support subsystem since fail-safe water requirements are provided with the environmental control and life support subsystem.

The operational use of fuel cells for manned space flight evolved during the Gemini, Apollo, and Skylab programs. The Space Shuttle fuel cells will be serviced between flights and reflown until each one has accumulated some 5000 hours of online service.

Interfaces of the electrical power subsystem with other subsystems, such as the avionics for control, and environmental control and life support subsystem, have not as yet been examined to any degree by the Panel. The Panel's major concerns here will deal with (1) crew hazards resulting from subsystem failures, e.g., loss of power to critical functions, (2) fire hazards resulting from short circuits or other failure modes, and (3) system design to prevent or inhibit deleterious events from propagating.

Current Status

Based on latest available data, it was noted that the current power requirements exceed the electrical power subsystem capability. The present electrical power requirement of 2006 KWH exceeds the 1609 KWH capability for the Orbiter 102. Mission energy requirements for seven days exceed the baseline cryogenic storage capability, i.e., tank sized for 1530 KWH. Activities underway are normal for this type of concern at this stage of vehicle development. The program is scrubbing electrical loads and equipment duty cycles to eliminate unnecessary power loadings. Monthly electrical power status reports are now being issued to assure high level contractor and NASA

visibility and continued control.

Also, based on the prior experience of the Panel, particular interest is focused on the electrical power subsystem fluid tubing connections and the fluid line insulation. These two areas are shown schematically with brief descriptive material in Figures 14 and 15 . A test program is being developed to provide insulation, packaging, venting and installation design data for all insulated fluid lines, particularly polyurethane foam insulations and TG-15000.

3.1.1.6   Crew Compartment Pressurization and Toxic Gas Control

The pressurized crew compartment has a volume of approximately 70 m^3 or 2300 ft.^{3}, and contains three levels. The upper section, or flight deck, the mid-section containing an airlock, avionics and living area, and the lower section containing the environmental control equipment.

An atmospheric revitalization pressure control system provides the crew compartment and habitable payload modules with a two-gas atmosphere of nitrogen and oxygen. It also provides the oxygen to the emergency breathing subsystem and airlock support subsystem, and provides nitrogen for pressurization of the potable and waste water tanks. Table VI is a recap of the functions and performance requirements of this subsystem. Also, the atmospheric revitalization loop circulates and filters cabin air, controls the atmosphere CO_2 level, provides temerature control, and removes latent and sensible heat

through the humidity control heat exchanger.

Cabin pressure is normally maintained at $14.7{-}^{+}0.2$ psia, but in the event of excessive cabin leakage an $8{-}^{+}0.2$ psia regulator is used. Sufficient make-up gas is available for 165 minutes pressure maintenance at this 8.0 psia value, assuming leakage equivalent to a 0.45 inch diameter hole. The atmosphere venting control provides for the relieving of excessive crew compartment pressure differentials whether negative or positive. This is a part of the pressure control system. The pressurization system is not designed to handle a second failure after 8 psia cabin condition exists. The crew will be on oxygen masks during emergency cabin pressure maintenance of 8 psia. Smoke detector units located in the avionics' bays require refurbishment every 2400 hours of operation.

Orbiter 101's pressurized compartment has passed its qualification tests.

3.1.1.7 Hydraulic Subsystem

Hydraulic subsystem provides power to actuate the aerodynamic flight control surfaces, main engine gimbals, main and nose landing gear, main landing gear brakes, the main engine valve controls and nose wheel steering.

Hydraulic power is provided by three independent, fifty percent power systems that provide the required degree of redundancy. The Panel was told that this approach minimizes weight, power extraction,

and system complexity and emphasizes balanced design between systems. A number of components have been standardized through commonality procedures thus reducing the cost, development time, and logistic support.

This subsystem is active during liftoff, ascent and orbital insertion. It provides for concurrent operation of rudder, main engine thrust vector control and main engine valves. The subsystem is passive in orbit except for a low pressure, electrically driven pump in each subsystem. The pump provides circulation to assure thermal conditioning. Activation of the subsystem is prior to deorbit burn and operates through reentry and landing. The main pumps are driven by hydrazine fuel auxiliary power units.

Each hydraulic system utilizes a 63 gpm variable displacement pump, powered by an individual auxiliary power unit, all of which contributes to the redundancy of hydraulic power sources. Assignment of functions to each system is based upon optimum power extraction and distribution, maximum flight safety, and minimum weight without segregation of flight control and utility functions.

The hydraulic subsystem equipment is compatible with fluid specification MIL-H-83282. Its bulk fluid temperature is maintained below 275° F. by a hydraulic fluid/water boiler heat exchanger.

The hydraulic distribution system consists of tubing and fittings fabricated from titanium. Approximately eighty percent of the tubing connections are of the permanent welded type. Minimum use of separable

fittings improves the system integrity. Flared tube fittings are not used. Metal lines, designed to flex, are used in lieu of hoses, where possible, to reduce maintenance and improve safety.

Metallic, non-elastomeric and elastomeric seals are used as best suited for individual applications. Because of the upper temperature limit of 275° F., elastomeric seals can be used where they offer advantages over other sealing techniques. Experience with aircraft hydraulic systems has also demonstrated that satisfactory system operation can be achieved with non-elastomeric and metallic seals.

A hydraulic subsystem working pressure of 3,000 psi was selected on the basis of minimum cost, minimum risk and better stiffness quality. The system is capable of operating when subjected to normal g, zero g, and hard vacuum encountered in orbit.

The three fifty percent system configuration (fail-safe) was selected in preference to an original design of four fifty percent (fail-operational/fail-safe) configuration as a result of an extensive study of historical failure data of hydraulic components, the limited operational exposure time during ascent (abort decision time) and, of course, weight and cost savings.

From the point of view of reliability, the system requirements state that the hydraulic subsystem shall provide safe flight and landing in the event of any single failure which causes loss of one hydraulic string (fail-safe). The avionics/hydraulic interface is

required to have a design that is two failure tolerant (fail-operational/fail-safe). The subsystem also has a maintenance requirement that it be consistent with the turnaround operation and be capable of being maintained in the horizontal as well as vertical position. Aerosurface controls operated by the hydraulic system are shown in Figure 16 .

The hydraulic subsystem interfaces with the following space orbiter subsystems:

       (a)   Flight control surfaces - elevons, rudder, speed brake, and body flap.

       (b)   Main engine thrust vector control.

       (c)   Utility loads.

       (d)   Steering, and landing gear brakes.

       (e)   Avionics - displays and controls, and flight control electronics.

Actuators used in the flight control subsystems (elevons, main propulsion system thrust vector controls and landing gear) have been approved by Rockwell International, Space Division, as acceptable risks based upon the very low probability of rupture or mechanical binding modes of failure.

While the Panel has not had the opportuntity to review this area in depth, the following questions would appear appropriate based on experience with other systems:

(a) To what extent are failure isolation techniques, such as hydraulic fuses, hydraulic circuit breakers, and return line check valves used to isolate a failed component.

(b) It has been a general rule that whenever hydraulic power is necessary for critical safety items, two independent subsystems are used. Why is this not the case for the Orbiter?

(c) Is there assurance that sufficient fluid cooling is available to maintain compatible fluid and seal temperatures?

(d) What parameters relating to actuator failure modes and life expectancy are being measured on the approach and landing test vehicle and on the Orbiter used for the first vertical flights? Does a mathematical model exist so that these measurements can be related to the design and component test data to further enhance hardware verification?

(e) What failure modes of the hydraulic subsystem result in the loss of the Orbiter - either directly or through the failure of a second system impacted by the failure of the first system?

(f) What is the method of validating these systems to achieve the necessary confidence in the design selected by NASA/Rockwell International. In other words, if the testing is not beyond the true expected conditions, how valid is the risk acceptance logic?

(g) What specific hardware/management controls are placed
on the designers and manufacturers other than the prime Orbiter sub-
contractor?

3.1.1.8 Orbiter Separation Systems

The separation of the Orbiter from the External Tank involves
three separation systems: (1) forward structural attach, (2) aft
structural attach, and (3) Orbiter/ET umbilical plate separation,
including the electrical umbilical separation. See Figure 17 .

Separation from the carrier aircraft (Boeing 747) involves for-
ward and aft structural separation areas that are different from the
Orbiter/External Tank arrangement, but the method of separation is
essentially the same. See Figure 18.

The forward structural attach/separation configuration consists
of a dual piston pressure actuated frangible attach bolt coupled with
a standard nut. Each piston can fracture the bolt at the Orbiter
Thermal Protection Subsystem moldline utilizing pressure generated by
one of two Apollo-type pressure cartridges. Subsequent to separation,
three centering plungers/springs align the bolt separation plane with
the Orbiter TPS moldline by rotating the retained portion of the bolt
within the Orbiter. No close-out door is required since the stub bolt
and spherical bearing are essentailly flush with the TPS moldline.

The aft structural attach/separation configuration consists of
two (right and left side) dual detonator frangible nuts coupled with

two corresponding attach bolts. Each bolt has a retraction spring which, after nut fragmentation, retracts the bolt into the ET hemisphere so there will be no interference in the separation sequence. On the Orbiter side, the dual Apollo-type detonators are enclosed in a cover assembly whose function is to contain nut fragments and hot gas generated by the operation of the detonators, either of which will fracture the nut.

The Orbiter/ET umbilical plate separation configuration consists of two assemblies (right and left side). Each assembly contains three dual detonator frangible nut/bolt combinations which hold the Orbiter and ET umbilical plates together during mated flight. Each bolt has a retraction spring which, after release of the nut, retracts the bolt to the ET side of the interface. On the Orbiter side, each frangible nut with its Apollo-type detonators is enclosed in a debris container. Each Orbiter umbilical plate has three retractors which, after release of the three frangible nut/bolt combinations, retract the plate approximately two and one-half inches. Retraction motion does a number of things: (1) disconnects the Orbiter/ET electrical umbilical in the first half inch of travel, (2) releases the trapped fluids between the Orbiter and the ET oxygen and hydrogen shutoff valves, and (3) serves as a backup for closing the oxygen and hydrogen shutoff valves. Each Orbiter umbilical plate has three stabilizing bungees to hold it in position after separation.

The questions that would seem most appropriate at this time are:

(a) During separation of the Orbiter and External Tank, propellants are released from the feedlines. With hot surfaces, hot wires and so on, what is the potential hazard of the oxygen and hydrogen being ignited?

(b) What is the adequacy of the separation system and the operational procedures to assure a safe physical separation of the Orbiter and External Tank under nominal and non-nominal flight conditions? For instance, all separation modes normally require the use of the forward Orbiter RCS operation, assurance that the separation of each of the three points to be separated are done within the required time period. At what point during thrusting by propulsion units of the total Shuttle system can separation occur?

(c) What is the hazard of the Orbiter and External Tank recontacting after separation?

(d) What is the ability to maintain the oxygen valves and hydrogen valve in the open position up to separation and the ability to assure closure after separation?

(e) What is the basis for confidence that there is no potential hang-up problem at the aft structural separation interface after the attachment bolt is retracted?

(f) Since umbilical door release is accomplished through

the use of a spring-loaded latch on the External Tank, what is the hazard from door, door hinge, or latch failure?

3.1.1.9 Structures

The Panel has not examined the basic Orbiter structure in any detail but has opted to look at those items from the standpoint of the test program used to validate the structure. The TPS and doors are covered under separate sections of this report. Another view of the Orbiter structure is obtained from an evaluation of the interface between the Orbiter and the External Tank and the Orbiter interface with the Main Engine. Added to this is the examination of the abort operations' area which includes an understanding of the ability to meet intact abort modes requirements.

3.2 Space Shuttle Main Engine

The Orbiter Main Propulsion Subsystem consists of the Space Shuttle Main Engines (SSME), the External Tank (ET) which stores and supplies liquid oxygen and liquid hydrogen for the SSME's, and a system of valves, plumbing, pumps, etc. located in the Orbiter which deliver the propellants to the engines.

The three main engines are started during the countdown. When they attain a ninety percent thrust level, the Solid Rocket Motors are ignited and liftoff is achieved. During the burn of the engines, they are throttled as required to limit vehicle acceleration to 3g. Gimbaling of the main engines provides steering during ascent in conjunction with Solid Rocket Booster thrust vector control. The SSME's burn for about eight minutes. Final boost into orbit is provided by the Orbital Maneuvering Subsystem (OMS). Each of the three main engines is approximately fourteen feet long with a nozzle about eight feet in diameter. The engines produce a nominal sea-level thrust of 375,000 pounds each and a vacuum thrust of 475,000 pounds. They are throttleable over a thrust range of fifty percent to one-hundred and nine percent of the nominal thrust level.

Orbiter interfaces are basically of three types - fluid, electrical, and structural. The fluid connections consist of the main propellant lines which transmit liquid hydrogen and oxygen and the fluid connections located at the interface connect panel mounted on the vehicle. These

provide fluids to and from the individual engines as follows:

>   (a) Hydraulic supply to and from the engine.
>
>   (b) Nitrogen purge (ground) to the engine.
>
>   (c) Helium supply to the engine.
>
>   (d) Fuel and oxidizer bleed from the engine.
>
>   (e) Gaseous fuel and oxidizer (pressurant) from the engine.

The propellant fluid connections at the interconnect panel con-
sist of bolted swivel flanges. All remaining fluid connections are
attached with bolted flanges except for the hydraulic system which
uses self-sealing quick disconnects. Flexibility for these joints
are provided with flex hoses on the engine side of the interface.

Electrical interface between the engines and the Orbiter are
made at the electrical connect interface panel located on each engine.
These interfaces consist of the following:

>   (a) Single 28 vdc power connector.
>
>   (b) Two 115/208 vac power connectors.
>
>   (c) Three communication and data transmission connectors.

AC power of 115/208 volt, 400Hz, 3-phase, is supplied to the engine
controller and the controller conditions the power to the require-
ments of the various engine actuation and instrumentation subsystems.
The 28 vdc is provided to operate both the SSME controller heaters
and a redundant coil on each engine's emergency pneumatic shutdown con-
trol solenoid valve which is normally open. Engine shutdown cannot

occur when the crew activates the engine limit control to inhibit engine shutdown. Operational and checkout commands and engine flight data are supplied via the electrical interface connectors at the engine-supplied electrical interface connect panel. Commands consist of engine start, shutdown, thrust level changes, checkout, and sequence checks. Engine flight data to the vehicle consist of information necessary for malfunction display, fault isolation, maintenance recording, trend analysis, performance monitoring and checkout. Three parallel redundant connectors provide a path for the Orbiter-to-engine commands. A minimum of two of the three commands must be received before the engine response can be initiated. Two of these connectors are also employed to transmit the engine flight data back to the Orbiter. The aft Orbiter thrust structure, the third interface, is built up with a titanium/boron epoxy material. Another interface is the honeycomb-base aluminum heat shield with insulation to protect the SSME from thermal inputs.

Integrated testing of subsystems is a critical milestone in the SSME program. It will be conducted at the National Space Technology Laboratories (NSTL) in Mississippi. The first engine firing at rated power level will take place at NSTL on a modified Apollo firing test stand in the winter of 1975. This will be followed by the first throttling test over the rated power level range. The Integrated System Test Bed (ISTB) will demonstrate the design's ability to handle

the high pressures and repeatable operations required of it. The
ISTB engine configuration varies somewhat from the flight-type
engine in the following areas: there is no LOX tank pressurization
heat exchanger, changes in material (high pressure fuel line, small
fluid lines, powerhead ducts , and modified insulation), and the
electronic controller assembly is not a flight type unit but is a
bench test unit built in racks. The ISTB has progressed as follows:

Assembly completed	3/13/75
Checkout completed	3/21/75
ISTB shipped	3/25/75
ISTB at NSTL	3/28/75
ISTB installed at NSTL	4/7/75
Test Readiness Review	5/7/75
ISTB first firing	June 1975

There is no gimbaling planned during the ISTB program.

3.2.2 Subsystems Critical to Crew Safety

For the purposes of this report, the Space Shuttle Main Engine
as a system is divided into the following subsystems:

(a) Combustion devices

(b) Turbo-machinery

(c) Pneumatics

(d) Propellant valves

(e) Hydraulics

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(f) Controller

(g) Igniters

(h) Electrical harnesses

(i) Instrumentation

(j) Interconnects and SSME/Orbiter interfaces

(k) Gimbal

As with the Orbiter element of the Space Shuttle program, the
Panel recognized that any one or a combination of these subsystems
and their components may be considered as affecting crew safety, but
from the point of view of the Panel it was necessary to determine
which of these should be focused on during the review period. The
basis of this focus was (1) on subsystems and/or components extend-
ing the technical (material, fabrication, etc.) state-of-the-art in
the literal sense or in the application, (2) those subsystems and/or
components which prior program "lessons" have indicated as areas of
concern, (3) areas which the Panel members considered most vulnerable
to "human error," and (4) areas which can affect crew safety but which
cannot or will not have been adequately tested or validated prior to
first flight. With these criteria in mind the Panel examined the
following subsystems in some detail:

3.2.2.1   Engine Electronic Controller Assembly

3.2.2.2   Main Combustion Chamber

3.2.2.3   High Pressure Turbo-Pumps

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3.2.2.4   Heat Exchanger

3.2.2.5   Hot Gas Manifold

The Controller is significant for crew safety because of its responsibility for detecting, monitoring, and controlling engine failure, thrust and propellant mixture ration, and engine starts and shutdowns and engine gimbaling.

The manifold, exchanger and chamber are of particular significance because they have complex welds and are subject to hydrogen embrittlement during operation. Material safety factors may be reduced through flow erosion or fabrication problems. Finally, it is difficult to inspect the finished item.

Also, the Panel reviewed the following areas to assure that risk assessment was receiving appropriate attention:

3.2.2.6   POGO

3.2.2.7   Ground Operations and GSE

3.2.2.8   Hydraulic Fluid

3.2.2.9   Lightning Effects

POGO results from dynamic coupling of the structure, propulsion, and flight control subsystems during all phases of powered flight under all possible payload variations. Thus POGO suppression hardware has had to be designed to eliminate coupling and the resultant structural instabilities.

Ground operations and ground support equipment are being developed

to discover failures and predict malfunctions before they occur.

The Panel had asked the Program to review its use of "red oil" hydraulic fluid and consider alternative hydraulic fluids that are more fire resistant. The Program has made a change and the Panel reviewed the new choice.

Lightning was a concern because of its impact on such subsystems as the Controller.

3.2.2.1   SSME Controller (Electronic Controller Assembly)

Systems Design

The SSME utilizes a full-authority digital electronic control with hydraulic servo-actuated valves. The Controller operates in conjunction with engine sensors, valves, actuators, spark ignitors, harnesses, and an operational computer program (software) to provide a self-contained system for:

    (a)   Closed loop engine control.

    (b)   On-board engine checkout.

    (c)   Engine limit monitoring.

    (d)   Engine start readiness verification.

    (e)   Engine start and shutdown sequencing.

    (f)   Engine maintenance data acquisition.

The engine/controller functional relationships are shown in Figure 19 . The controller electronics arrangement is shown in Figure 20 . In that same figure is shown the responsibility of the

two Honeywell organizations.

Characteristics of the Controller of interest are:

(a)   Overall dimensions .... 23.5" x 14.5" x 17"

(b)   Weight .... 197 pounds

(c)   Input power .... 472 watts to 636 watts

(d)   Convective cooling .... (primary mode)

(e)   Temperature environment ...... operational -50° to $+ 95^{\circ}$ F.

                                                         Non-operational $- 200^{\circ}$ to $+ 200^{\circ}$ F.

(f)   Vibration environment .... sine 24 g's peak

                                                         random 22.5 g's root mean square

(g)   Unit is mounted on engine using a three-point hard-mount.

The electrical harness assemblies between the engine interface and the Controller are of two types - conventional and flexible armored. Conventional harness is used where redundant electrical functions are carried through separate connectors and will be physically routed independent of each other.   Flexible armored harness is used where redundant electrical functions cannot be physically routed separately.

Panel's Initial Review

Prior to reviewing the Controller program, the Panel requested specific information as background data on this critical hardware. The documents requested were (1) reliability analysis and test data that documented the Controller configuration and its projected ability to support mission objectives, (2) prediction analyses for the ex-

pected mean-time-between failure rates and the basis upon which such predictions were made, (3) trade-off studies between the Controller using plated-wire type memories and a design using the latest of the more traditional type cores. This material was received and reviewed by the Panel and staff. Typical data included in the response is shown in Tables VII to IX .

The Panel then undertook a series of inspections.

Status of the Controller program in the early summer of 1974 looked like this:

(a) Design verification tests were completed on the input electronics, output electronics and the computer interface electronics. The digital computer processor logic was proved through the use of a Honeywell HDC-601 computer unit and on the engineering and bench test SSME controller assemblies. The digital computer memory design, including the use of plated-wire, was proven through testing of a "half-stack" unit. The half-stack test was a test using a rack-mounted integrated memory assembly. The Controller power supply was undergoing expedited documentation (specifications, etc.), procurement, and fabrication. At the same time power supply breadboard tests showed that there were numerous problems with the design. Some of the problems associated with the subsystem/circuit/component items were power supply voltage below minimum allowable, output ripple, and failure of inverter transistors, master interconnect board pins and

sockets pulling out, deflecting or not matching. SM-1 (structural model) vibration testing had revealed foam retention and seal problems. There were parts' problems with integrated circuits and connectors. Integrating the digital computer unit components was a problem as was the integration of the total Controller. Noise in the memory and parity errors in the computer unit also were concerns at that time.

Thermal design of the package was verified by analysis and tests on the structural model (SM-1), which was not, of course, exactly like the flight design. However, given the excellent correlation between analysis and test results and the piece part temperatures and conduction rates to the case, there was sufficient margin remaining in the design to allow for production process variables and for some modifications.

Vibration tests were conducted with the SM-1 unit which verified that the general packaging concept would meet the requirements. Problems surfaced with regard to the case aluminum seal which leaked, excessive resonances in some of the parts, and the retention of the half-stack card and foam assemblies. Solutions for these mechanical problems were identified but further testing was necessary to prove that the solutions would actually work. Environmental test for salt, humidity, etc. were to be conducted. Design verification testing for thermal conditions was to be conducted on the memory boards, printed wire boards, and master interconnect boards.

A necessary adjunct to the development of the Controller hardware and software are the many test items and facilities which prove design and fabrication concepts and validate the prototype and flight hardware. The software verification facility was operational, the design for the command and data simulator design complete, and hardware test equipment of many types were built and in use. Such test equipment as "automatic wiring board test stations", "power Supply Conditioner Test Unit," and "Memory System Exerciser" were proceeding satisfactorily.

Software design was demonstrated on the Controller engineering model and the bench test units. The electrical interface between the engine and the Orbiter was verified as was the ability of the software to conduct engine start, mainstage control, and engine shutdown. At that time the computer acceptance test program design was complete and 95% debugged, the Controller acceptance test program baseline design was complete but not debugged, and the operational program design was complete with 50% of it coded and debugged.

There was adequate experience with the development of the plated-wire memory to warrant confidence in the technology. However, there did not appear to be an understanding of the fundamental physics to assure that surprises could be anticipated and a timely course of resolution decided upon and implemented. If additional surprises did occur, they probably could be solved by trial and error, given suffi-

cient time, but such surprises would probably impact the then very tight schedule requirements. At that time the half-stack test of the rack mounted integrated memory system and the structural thermal verification program were completed. Fabrication improvement was indicated by the acceptance trend of plated-wire assemblies.

While there was no single reporting format available which systematically stated the significant lessons learned from the Viking program and their disposition with regard to the Shuttle program, the new program manager had his staff review the minutes and audits from numerous Viking reviews and identify specific actions. As a result of this review, design changes were incorporated into the Digital Computer Unit. Daily production schedule reviews were instituted with closed loop corrective action and follow-up for all problems defined. The process specifications and the training program for the production and inspection workers were strengthened. Management and supervisory levels made it their business to have more contact with the total Viking and Shuttle personnel. Viking audit disciplines were incorporated into the Honeywell basic management and technical system.

Current Status

Since its initial review in the summer of 1974, the Panel has examined the SSME and its Controller in September 1974, January 1975, and April 1975. The current Controller status as seen from these reviews looks like this:

(a) SM-1 (structural model) thermal and vibration tests have been completed and the structural and thermal math models have been verified.

(b) The breadboard controllers BT-1 and EM-1 have been in use and the Controller functions such as start-up and shut-down have been demonstrated.

(c) The command and data simulators have been used extensively as have the Controller checkout consoles and laboratory model computer used in the integration of the Controller subassemblies.

(d) The digital computer unit number SN-1 has been completed and integrated in the first prototype controller, PP-1. This unit, however, has experienced intermittent parity errors which are under study at this time. All of the Controller functions of the PP-1 have been exercised and some out-of-specification conditions have been surfaced which also are being examined for proper resolution.

(e) The quality of the workmanship and inspection system has been improved, with the result that the rejection rates for such things as plated-wire memory boards has been reduced to a very acceptable level.

(f) The BT-1 unit, to be used with the SSME Integrated System Test Bed test program, was successfully checked in March 1975 and has been delivered to NSTL for installation into the ISTB facility. SSME to Orbiter interface documentation (ICD 13M15000) has

been issued and is under standard control of the configuration control system and the interface working group.

(g) Operational philosophy for "out of limit" signals has been defined and agreed to as shown by the current design. This design provides for engine sensor inputs to be out-of-limits three consecutive check periods before the input is "declared" failed, which is called a "three strike" concept. A part of this system provides for rechecking critical parameters immediately during the same major status loop check. A major status loop check takes about twenty mil seconds. Less time critical parameters are rechecked during the next two major sense-reporting cycles. At the same time the out-of-limits data are not used by the engine control system at that time. For internal Controller parameters the "two-strike" concept is used in which two consecutive out-of-limit conditions must exist before that item is declared "failed." Short term anomalies will not cause pre-mature loss of redundancy, e.g., shifting to the second computer section of the Controller or engine shutdown.

(h) The power supply units for use in the PP-1 and PP-2 Controllers have been completed and tested satisfactorily. Design verification tests have been conducted, resulting in a low degree of electromagnetic interference beyond specification limits. This does not appear to be a major problem.

(i) Master Interconnect Boards, because of their complexity,

have posed numerous production problems. Four have been built for use in the PP-1 and PP-2 units in addition to the development units. To date the development tests have been completed. Manufacturing processes along with alignment fixtures and insertion tools have been established. The design verification test hardware is being built. A problem still to be resolved is the noise being coupled into the memory sense lines due to wire routing and inadequate shielding. Modifications are being incorporated to add sense-line-shielding on the Master Interconnect Board and to reroute control sense lines. Additional improvements are being evaluated in case they are needed in the wiring approach to the memory area of the board.

(j) Four memory systems have been built for the PP-1 and PP-2 Controller units and twelve half-stacks have been built and tested. Several hours of memory operation have been accomplished at the digital computer unit level. There have been intermittent parity errors, and a noise problem has been identified in integrated testing of the Controller. In addition to the fixes to the Master Interconnect Board, changes to increase the memory plane shielding and plated-wire output are being studied in order to increase the signal to noise ratios. To put the parity error problem in perspective, the extent of the testing on the two memory channels should be considered. Channel "A" operated over the temperature range at the digital computer level for eight hours with only a single occurrence of parity error. Channel

"B" had error-free operation over the temperature range for some 54 hours

at the memory system level of installation, and approximately 100 hours

of operation at room temperature with comparatively few intermittent

parity errors at the Digital Computer Unit level.

(k) The basic software elements and/or routines are as

follows:

Executive

Ground checkout

Self-test

Start preparation

Power range control

Vehicle commands

Limit monitoring

Sensor processing

Output monitoring

Failure response

Post shutdown

Constraints on the software programs are the memory size of 16,384

words and the Controller major cycle time of 20 milliseconds. In

December 1974 the memory capacity was exceeded. As a result there

is an effort at this time to reduce the word requirement by proper

software programming and or some reduction in requirements. At this

time the emphasis is on meeting the SSME Integrated Subsystem Test

Bed program. MSFC noted that considerable effort has been placed on providing the proper software. For example, the contractor established a shift operation. Schedules are also established and progress is reviewed on a daily basis. "Memory scrub groups" have been established at Honeywell, Rocketdyne and NASA.

3.2.2.2 Combustion Devices

Systems Design

The function of the Main Combustion Chamber is to contain and direct the forces of combustion generated by the burning of the propellants. The hot gases are accelerated to sonic velocity at the throat and supersonically expanded to an area ratio of 5:1 at the interface with the engine main nozzle. The Main Combustion Chamber consists of a structural outer jacket, regeneratively cooled liner, and inlet and outlet manifolds. Two thrust vector control struts are attached to it as are mounts for the engine electronic controller assembly. The Main Combustion Chamber fabrication problems or concerns are similar to those described for the hot-gas manifold unit. In addition the cooling of this combustion chamber requires a rate of heat removal three times higher than any previous liquid fueled engine, 100 btu/ft^2/sec. The number of welds used in producing the chamber are about 112 of which 16 are electron beam welds.

Current Status

Main engine combusion devices have had fabrication problems dur-

ing their development period. Main Combustion Chamber and nozzle fabrication has been completed in support of the Integrated System Test Bed program hardware, including successful proof-testing to 1.2 times the rated-power level operating conditions, or about 6800 psi. The augmented spark igniter has been demonstrated successfully, including a 600 second run at full power level conditions. Subscale model of the main injector (40,000 pound thrust unit) has been demonstrated. Hot fire tests have been conducted on the oxidizer preburner and the fuel preburner, which all appear to meet performance requirements. Flow induced vibration was noted in some of these tests, but this apparently has been remedied. The LOX tank pressurization heat exchanger, located in the LOX side of the hot gas manifold assembly, is a critical item in the engine combustion system. The present heat exchanger design requires rigid manufacturing and inspection control and verification testing to assure an acceptable unit. Rocketdyne feels that this can be accomplished.

3.2.2.3 Turbomachines

Systems Design

The high-pressure fuel turbopump receives fuel from the low-pressure fuel pump and boosts the pressure to the level required for the pre-burners. The fuel is then discharged through the high-pressure fuel pump discharge duct to the main fuel valve. This turbopump consists of a three-stage centrifugal pump drive by a two-stage reaction

turbine. During propellant conditioning, the liftoff seal is closed around the pump shaft, preventing LH_2 from flowing into the turbine area and out through the hot-gas manifold to the main injector. At engine start, the liftoff seal is actuated by the pump pressure at a pump speed of approximately 7000 rpm. During mainstage firing, the pump reacts to throttling commands by changing discharge pressure and flowrate. The lift-off seal reseats when the pump pressure decreases to a speed of 7000 rpm.

The high-pressure oxidizer turbopump receives oxidizer from the low pressure pump and boosts the pressure to a sufficient level to provide adequate flow-rate and pressure to the thrust chamber and the preburners. Engine start activates the pump intermediate seal purge that provides an inert barrier between the pump and turbine during operation.

Current Status

Material presented to the Panel indicates that the turbine nozzle castings and turbine strut forgings around the turbine have been the major problem areas. The initial vendor was unable to cast the nozzles due to shrinkage, failure to fill molds, and erratic material problems. To resolve the problem quickly, a change in vendors was made in July 1974 and nozzles were successfully cast using a new material (INCO 713LC instead of MAR-M-246). It turned out that the life for the 713LC type nozzle casting was inadequate. Work was re-

sumed on the use of the original material and it was found that the
new supplier was in fact able to produce successful nozzles with
MAR-M-246 material that now appear to meet the turbine nozzle require-
ments. These nozzle castings are still receiving MSFC's attention to
assure adequate hardware is available for the early engines in the SSME
program. The turbine inlet struts had some material problems regard-
ing acceptable axial strength of the forgings. This problem has been
resolved and the forgings are adequate to meet program needs.

3.2.2.4 Heat Exchanger

System Design

The heat exchanger provides oxidizer gas pressurant for vehicle
LOX tank pressurization. This heat exchanger is a multipath, single-
pass, cross-flow device installed in the LOX side of the hot-gas mani-
fold at the high-pressure oxidizer turbopump turbine exhaust. The
supports for the heat exchanger tubes are mounted to the liner wall so
as to allow small movements during expansion and contraction of the
tubes. The tubes enter and leave the hot-gas manifold through flared
projections of the manifold liner. The flared projections provide stag-
nant gas pockets for reduction of thermal stresses at the tube-to-oxygen
manifold attach welds. The heat exchanger is depicted schematically in
Figure 21 . The major concern here is with the heat exchanger coil
material and its ability to be assembled and then to remain virtually
leakproof during its operational life. For instance, a leak could

permit ignitable mixtures of oxygen and fuel-rich hot gas to enter the oxygen supply line or allow oxidizer into the hot gas manifold with ignition that could also damage an adjacent coil or the liner and manifold wall.

Current Status

The design and manufacturing approach being used to reduce the possibility of this hazard include a number of actions.

An ultimate factor of safety of 1.75 is used rather than the usual 1.4. Where fatigue life of 240 cycles normally is required, this has been increased to 1450 cycles for bifurcation joints, to 4500 cycles for weld joints, and to 26,000 cycles for parent metal.

Design verification structural tests will include leakage checks, vibration, proof pressure cycles, ultimate pressure, and low cycle fatigue tests.

Quality control on components will use ultrasonic, penetrant and x-ray, and helium leak tests (1×10^{-6} scc/sec at limit pressures). The Panel questioned the use of a leakage rate of less than 1×10^{-6} scc/sec at limit pressures noting that this leakage rate appeared excessive in determining the acceptability of the heat exchanger. This is being reevaluated at this time.

A modification being considered to the LOX pressurant control system which would interconnect the heat exchanger discharge upstream of the Orbiter flow control system, which would insure valve

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inlet pressure being above the hot-gas manifold pressure.

3.2.2.5 Manifold

System Design

The hot-gas manifold serves as the structural nucleus of the engine and provides gas passage interconnection for the preburners, high-pressure turbopumps, and the main injector. Hydrogen-rich hot gas (hydrogen and oxygen) flows through this manifold and then into the main injector. Cooling of the hot-gas manifold is accomplished by using double wall contruction (a structural outer wall and an inner liner). This provides a flow path for hydrogen gas coolant exhausting from the low-pressure hydrogen turbine. This configuration isolates the structural wall from the hot gases flowing within the inner liner.

Current Status

This hardware is fabricated with complex weld which has required considerable in-process rework at the fabrication location. Critical to achieving successful weld is the alignment of the joints and the materials and processes developed for such welds. Proper alignment reduces the stress concentrations and discontinuities that normally cause problems in welds. All manifolds are analyzed for weld adequacy. To further reduce induced stresses, prestraining and an annealing, heat treatments are utilized. Hydrogen-rich mixtures, particularly at high pressures (up to 6000 psi in part of the engine), leads to the possibility of metal embrittlement problems. The

possibility of cracks, warpage and structural failures obviously affect
the engine operation and performance from simple gas leakage to engine
shutdown, and in extreme cases potential aft compartment fire or ex-
plosion. Based on the material provided to the Panel, NASA and its
contractor are aware of these problems and continue to place very
heavy emphasis on eliminating the fabrication and material problems,
and on the test program to validate the design and manufacturing
processes.

3.2.2.6 POGO Suppression

The Problem

POGO is not only an SSME problem but also must be viewed from
a "systems" standpoint. The discussion here deals with the hardware
as currently designed and as attached to the SSME's themselves.
Systems integration aspects are covered in more detail in Section 6
of this volume. The Panel's concern with POGO effects goes back to
Saturn V launch vehicles in the Apollo program. Most large, pump-
fed rocket vehicles have had moderate to severe longitudinal oscillations
caused by POGO instability. Such oscillation can result in an environ-
ment severe enough to cause structural damage and affect crews physio-
logically. POGO is a closed-loop phenomenon involving fluid-feed-system
pressure oscillations which result in engine thrust perturbations and
structural motions. These may be visualized as beginning with small
vehicle accelerations that produce variations in propellant pressure

and flow rates, which in turn cause thrust variations, resulting in increasing vehicle oscillations.

Elements of the Space Shuttle Vehicle system involved in POGO are:

(a) Long liquid oxygen supply line.

(b) Asymmetric Shuttle structure and thrust vector couples, and coupling of flight control and POGO instabilities.

(c) Main propulsion system (SSME's, ET, etc.) which operates from liftoff to orbit with extreme changes in vehicle structural characteristics and turbo-pump inlet pressures.

(d) Space Shuttle's main engines themselves, with their LOX and LH_2 high and low pressure dual pump systems.

The depth of NASA and contractor efforts to assure that POGO does not become a Shuttle operational problem can be seen in planning, documentation, testing, and analytical work being performed to resolve this concern. This includes the "POGO Prevention Plan" JCS 08130, dated January 6, 1975, as well as studies to determine the need for POGO suppression, and to add the suppression system. Such groups as the POGO integration Panel and the independent MSFC POGO analysis team, have been working this challenge.

Suppressor design requirements have been defined as follows:

(a) Location as close as practical to the High Pressure Oxygen Turbo-pump.

(b) Volume about 0.6 cubic feet or equivalent with ability to increase to more than one cubic foot if test program indicates this to be necessary.

(c) Damping of fluid surges (frequency of pulses) over a broad frequency range; inertance less than 1.1×10^{-3} $\sec^2/\text{in}^2$.

(d) Minimal fluid pressure-drop in the suppressor.

Comparison between the Saturn V and Space Shuttle engine/fluid systems is shown in Figure 22 . The POGO suppression system and its components are shown in Figures 23 and 24 .

Current Status

POGO mechanisms are known to be complex, and a continuing analytical program is being pursued to understand the phenomenon and its implications. The suppressor has been baselined. An extensive ground-based program is being conducted to verify the design. Extensive use has been made of Saturn data in designing the test program. Tests are being conducted at MSFC, Martin Marietta Company, Rocketdyne, and NSTL sites. The location, type, size and inertance of the proposed system have been arrived at after a thorough design trade-off study. Analysis of abort situations and their impact on the design of the POGO suppressor have to be accomplished to assure maximum safety. But the proof-of-the-pudding can only be found during flight tests under actual environments.

It appears that the liquid hydrogen does not contribute to any

degree to the POGO problem, and there is no apparent need for a suppression device in the liquid hydrogen fuel system. Preliminary examinations indicate that the Solid Rocket Motors do not contribute to any degree to the POGO problem, but the analysis is continuing.

3.2.2.7 Ground Operations and Ground Support

SSME's are designed for automatic checkout and fault isolation, use of "line replaceable units" with good accessibility and long life, and to accommodate the so-called "condition monitored" concept. This concept has as its objective the ability to discover failures before they occur, using nondestructive evaluation methods, and to eliminate premature maintenance.

SSME controller assembly has automatic checkout capabilities for self-test and fault isolation to the line replaceable unit level. Working in conjunction with ground equipment, it conducts the following tests:

(a) Pneumatic

(b) Actuator

(c) Sensor

(d) Flight readiness tests

(e) Redundancy verification

Panel interest will continue in this area to assure that ground operations and equipment do not adversely affect the engines and associated hardware during maintenance and preparation for launch.

The following GSE status was presented to the Panel recently:

(a) There are no significant GSE problems known at this time.

(b) While economic problems have resulted in quantitative reductions of GSE, there have been no quantitative cutbacks that would affect safety.

(c) Major GSE units have completed design verification testing.

(d) Majority of GSE components are now in service.

3.2.2.8 Hydraulic Fluid

Introduction of the MIL-H-83282 hydraulic oil in place of the original "red oil" has been made at all locations working on the SSME: NSTL, MSFC, Hydraulic Research Company, and at Rocketdyne. To date there appear to to be no functional problems associated with the use of this fluid, and laboratory tests continue to be conducted to assure that the fluid when in operational use will meet requirements under all induced environments.

3.2.2.9 Lightning Protection

The requirement currently on contract for lightning protection is MIL-B-5087B, Amendment 2, 31 August 1970, "Bonding, Electrical and Lightning Protection for Aerospace System." Use of this standard is currently under review, with the probability that it will be replaced by the NASA publication JSC 07636, "Space Shuttle Lightning Protection Criteria." Assessments are being made during the May 1975 time-frame

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with regard to lightning field amperage components, direct strike
capability, launch constraints, cable shielding requirements and cost
and weight impacts. Results of these assessments will be examined by
the Panel during upcoming reviews. Lightning protection for the
Shuttle as a system is discussed in more detail in Section 6 of
this report.

3.3 External Tank Project

The External Tank is a part of the main propulsion system, along with the main engines and interconnecting portions of the Orbiter vehicle.

In this section the discussion will be devoted expressly to the external tank and peripherally to those significant interfaces with the Orbiter and Solid Rocket Booster that affect crew safety.

The External Tank is the only element of the Shuttle system that is discarded after depletion of its oxidizer and fuel resources. Because it is expendable, great emphasis has been placed on low cost production of this tank. The external tank is being designed, developed and manufactured by the Martin Marietta Corporation at the Government-owned Michoud Assembly Facility in Louisiana.

The External Tank consists of three major components: (1) a liquid oxygen tank, (2) an inter-tank, and (3) a liquid hydrogen tank. It is of aluminum construction utilizing a spray-on foam insulation and spray-on ablator for thermal protection. A configuration is shown in Figure 25 . In September 1974 a Preliminary Design Review of the tank was conducted; the Critical Design Review is scheduled for the fall of 1975. Fabrication and assembly of the LOX and liquid hydrogen tanks for the structural test article will begin in the summer of 1975.

3.3.1 Subsystems Critical to Crew Safety

The tank can be divided into the following subsystems:

(a) Structures

(b) Propulsion and mechanical

(c) Electrical

(d) Separation and dispersion

(e) Thermal Protection Subsystem

(f) Ground support equipment and logistics

Particular attention was given by the Panel to those components or situations most critical to crew safety. These were chosen on the basis of the criteria used on other elements of the program - potential problems utilizing experience on prior programs and components that could critically degrade the performance of the Orbiter or SRB if they were improperly designed, could not be tested or analyzed to the degree necessary for confidence in them, or failed to operate during critical mission sequences. To illustrate, the Panel in its review of structures gave particular attention to fracture control. A review of the propulsion system focused on the anti-geysering system. Review of the electrical system focused on controlled use of teflon wiring as well as on lightning protection.

Weight control is as important a management concern on the External Tank as on the other elements of the Shuttle program. The next control weight has been set at 72,360 pounds. With a current estimated weight of 71,445 pounds, the margin is 915 pounds. There-

fore, the Panel is sensitive to the impact of weight control on decisions affecting crew safety.

3.3.1.1. Structures

System Design

The structure must retain the liquid oxygen and hydrogen within their respective tanks and must serve as the structural backbone of the launch and ascent Shuttle vehicle as well. Material provided to the Panel indicates that the design and construction of the structural portions of the External Tank follow the large Saturn tank and Titan tank methods, as well as the use of current sophisticated design tools developed by NASA (NASTRAN).

In light of prior program experience, the Panel reviewed the actions taken by NASA and contractor management to insure that the initiation or propogation of cracks or cracklike defects in the External Tank will not cause structural failures or unacceptable leaks.

Current Status

Fracture control plans have been developed to cover the phases of design, fabrication, test, environmental control, inspection, maintenance, repair, and acceptance procedures. A Fracture Control Board has been established to assure the plans are implemented. The straight polarity TIG welding process has been selected. Vendors for critical formed parts, such as gores and caps, have also been selected.

Both NASA and the contractor feel that the initial processes provide a reasonable basis for confidence.

Some fracture mechanic limits for tank welds are shown in Figure 26.

3.3.1.2 Propulsion and Mechanical System Design

The External Tank propulsion/mechanical subsystem delivers LOX and liquid hydrogen to the Orbiter interface from the external tankage. The propulsion and mechanical subsystem is comprised of the liquid oxygen feed system, liquid hydrogen feed system, LOX tank pressurization and vent/relief system, intertank and tank environment control systems. The separation system, normally considered a part of the mechanical and/or structures' system, is discussed under a separate section later in this report. There are three separate mechanisms associated with the External Tank propulsion subsystem: (1) intertank umbilical disconnect, (2) right aft ET/Orbiter umbilical LOX disconnect, and (3) left aft ET/Orbiter umbilical liquid hydrogen disconnect. Only the intertank disconnect is discussed in this section since the other two are a part of the in-flight separation system.

One of the more significant design features of the external tank that should provide for greater hardware reliability and reduced mission risk is a dual flange seal with the capability of monitoring leakage through the primary seal. This seal is used at

all major structural tank connections. See Figure 27 .

The LOX pressurization line is supported by 29 sliding supports and three fixed supports. These supports are bolted to floating anchor nuts in brackets welded to structure on the LOX tank. A phenolic insulation block is placed between the support and the tank to reduce heat transfer. These same supports also serve the larger anti-geyser line and the electrical tray. Seven flexible joints accommodate thermal and dynamic deflections. Figure 28 shows not only these lines but the LOX propellant feed-system as a whole.

The vent/relief assembly serves two functions: (1) tank venting during propellant loading, which controls the boil-off rate, and (2) relief of the ullage pressure to protect the tank structure in the event that it exceeds a preset value.

The liquid hydrogen feed system is similar to the LOX system. The liquid hydrogen pressurization line assembly provides the means for transmitting adequate pressure and for the correct rate of flow of LH_2 to the Orbiter main engines. The LH_2 recirculation line is a 4-inch vacuum-jacketed line which provides a return path for the hydrogen recirculation flow that used to thermally precondition the SSME prior to initiation of engine start. The vent/relief assembly serves the same two functions as the similar system in the LOX feed system.

The tank environmental control or conditioning system includes LOX, liquid hydrogen and inter-tank purge hardware. Propellant tank purge is accomplished prior to propellant loading. The inter-tank purge uses dry gaseous nitrogen to remove contaminants from its area and to maintain the temperature of the inter-tank area at 80 $\pm$ 15 degrees F.

External tank-to-ground interface consists of an environmental control system disconnect, a gaseous hydrogen vent line disconnect, and LOX and liquid hydrogen vent valve pneumatic control line disconnects. See Figure 29 .

The Panel gave particular attention to the control of the possible hazard of geysering. Geysering is the rapid upwelling of LOX into the tank ullage area; this can cause a rapid reduction of the ullage temperature, reduce the ullage pressure and, in the worst case, result in the collapse of the LOX tank. This phenomenon has been found on prior large liquid rockets and occurs when a comparatively high density cryogenic fluid contained in a line or tank begins to heat up and bubbles form at a progressively increasing rate. As a bubble matures it begins to rise through the liquid, due to its reduced density. At the same time the liquid head (pressure) on the bubble is constantly being reduced. As the bubble moves upward it accelerates and pushes liquid ahead of it. When the bubble reaches the tank, the liquid above it is expelled upward through the liquid surface into

the open tank area with great force. It is not unusual for this slug of liquid to weigh several hundred pounds. Thus, in addition to the possible tank pressure reduction, resulting in conditions conducive to tank collapse, there is a danger of the slug itself hitting internal structure and damaging the structure and any lines or instrumentation therein. The return of this liquid can also result in "water hammer" effects.

The geysering action is shown schematically in Figure 30 .

Current Status

NASA/MSFC and Martin Marietta Corporation have baselined what appears to be an acceptable anti-geysering system and test program, all of which must be completed before the initiation of the main propulsion test program at NSTL. To prevent geysering it is necessary to agitate the liquid column to prevent stratification or layering during the ground fill sequence when lines and tank are relatively warm. Current design plans are to use helium injection system as shown schematically in Figure 31 . Actual design of the system is still under study and analysis because the initial design concept as proposed was considered less than optimum. Location of the function of the 4-inch LOX anti-geyser line with the 17-inch LOX main-feed-line can potentially cause unpredictable flow patterns as well as nullify the desired effect of the system. This could happen if there is a ground helium supply failure for any reason because the LOX vent

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