and together with the Apollo, injects into a low Earth orbit. When the Moon and the S-IVB payload combination are in the most advantageous relative position, the S-IVB engine is re-ignited to accelerate the vehicle out of Earth orbit and into Earth–Moon transit.

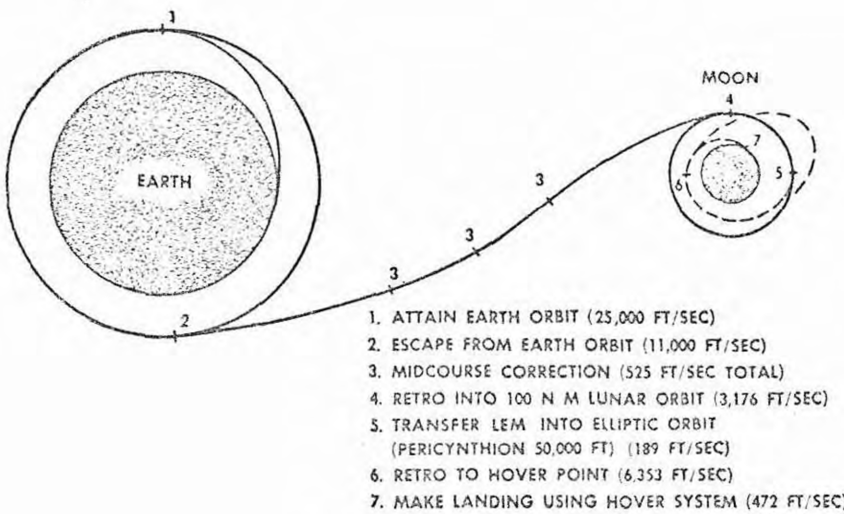

Fig. 2. Lunar rendezvous

Following the S-IVB stage cutoff, the command and service modules are separated (Fig. 3) from the vehicle and the Lunar Excursion Module (LEM) adapter covering the LEM, or bug, is detached. The Apollo capsule turns 180° to face the vehicle and mates with the LEM. After mating, the Apollo and the LEM pull away from the expended S-IVB stage.

After coasting to the moon, the Apollo injects the assembly into a low lunar orbit. After determining that all subsystems are functioning properly, two of the three astronauts will transfer from the Command Module (in Apollo) to the LEM. The excursion module is then placed on a trajectory which has the same period as the circumlunar orbit of the Command Module, but has a much lower perigee, about 50,000 ft. If the equipment is still functioning properly, a landing will be accomplished. The landing module, or bug, can hover for as much as a minute and translate as much as 1000 ft in any direction while hovering. After landing and after the crew has had a chance to explore the lunar surface, the bug will ascend and rendezvous with the Apollo in lunar orbit. The astronauts transfer to the larger craft and, leaving the bug in orbit, inject into an Earth-return trajectory.

The vehicle which accomplishes this mission (Fig. 4) has, for first stage propulsion, five 1.5 million-lb thrust Rocketdyne F-1 LOX/RP engines. The second stage has five 200,000-lb thrust Rocketdyne J-2 LOX/LH_2 engines, and the third stage has one similar Rocketdyne J-2 engine. The vehicle itself also has assorted storable propellant engines for roll and

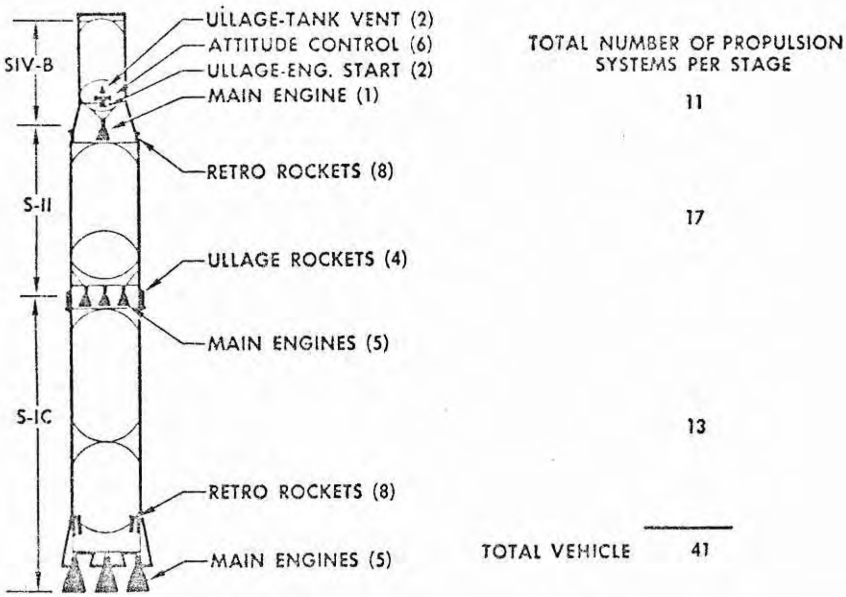

Fig. 4. Saturn V propulsion systems (LOR configuration)

attitude control, solid propellant ullage rockets for propellant settling at engine start, and retro action at stage separation.

In addition to these systems, the payload complex itself contains approximately 50 engines (Fig. 5). The service module contains a single Aerojet General engine employing storable propellants, with a thrust level of 21,000 lb. The LEM is really a two-stage vehicle. The landing stage employs a throttleable engine with a thrust level between 8 and 10,000 lb, using the same propellants as Apollo. The lunar take off system employs a 4000-lb thrust engine. Attitude control engines are employed in the S-IVB, Apollo, and LEM; these engines also employ storable propellants.

This mission, then, requires propulsion systems which perform in the following categories:

1. Primary thrust.
2. Attitude and roll control.
3. Ullage force for propellant settling.

4. Retro-action at stage separation.
5. Astronaut escape.

For each of the 98 rocket engines required to perform this mission, the space vehicle designer had to solve many critical problems to adapt the engine to the stage. Some of the most important engine parameters which

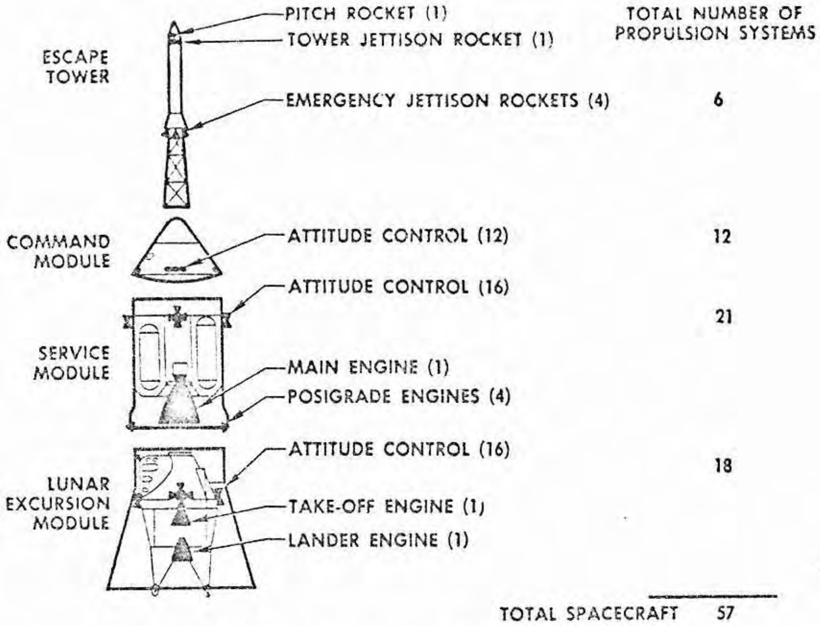

Fig. 5. Propulsion systems of Apollo spacecraft

affect vehicle performance are thrust, specific impulse, total impulse, and engine weight. On the left of Fig. 6, we see the effect on orbital payload of a pure change in thrust (without an accompanying change in specific impulse, engine weight, or total impulse) for the second and third stages of a hypothetical, large, three-stage rocket of the Saturn V class. On the right-hand portion of Fig. 6, all parameters are constant, except specific impulse. The conclusion to be drawn from this figure is: a second-stage designer is most interested in maximizing thrust and specific impulse, but to a third-stage designer, specific impulse is by far the more important parameter so long as thrust is reasonably high.

Where high thrust is desirable and single engines are not available, the approach has sometimes been to employ multiple engines. For example, in the S-II stage, five J-2 engines are used to produce 1 million lb of thrust. This approach involves additional vehicle complexity over a single engine of equivalent thrust, since more feed lines, control systems, pneumatic systems, and instrumentation are required. Furthermore, the engine empty weight is usually higher for a given thrust level when that thrust level is obtained through the use of multiple engines rather than a single engine.

The selection of propellants to be used in a given stage has major consequence on the design of that stage. The total impulse required, in part defines

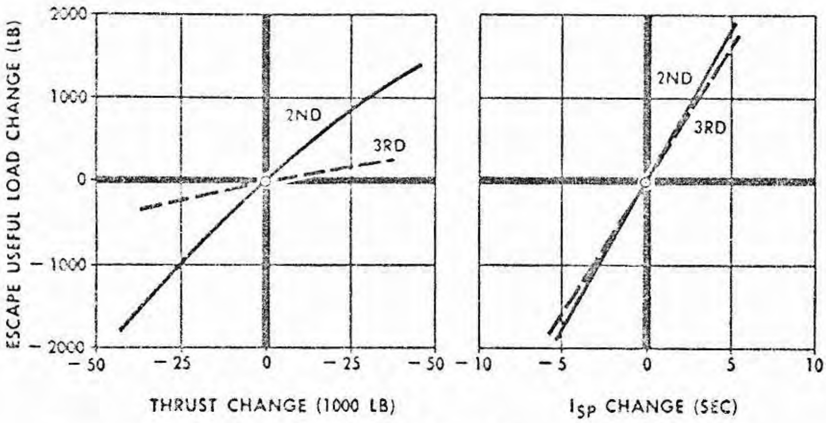

Fig. 6. Thrust and specific impulse effects. Saturn V LOR configuration

the size of the external propellant tank envelope. The division between oxidizer and fuel within that envelope is determined by the nominal engine mixture ratio requirements and propellant densities. Beyond this gross geometric consideration, the selection of propellants is important in other vehicle design areas such as: (1) structural design, (2) component development, (3) field testing, and (4) special subsystem requirements.

Structural Design

In the case of S-IVB, the use of LH_2 as a fuel caused several important structural innovations (Fig. 7). A significant problem in the design of this hydrogen-fueled vehicle was minimizing boiloff losses during ground standby and during the exit trajectory. In general, two approaches were possible: (1) external insulation, which could be attached directly to the tank walls or spaced off from the walls, or (2) internal insulation mounted to the hydrogen tank. In the case of the directly applied external insulation, the bonding agent would be forced to withstand liquid hydrogen temperatures of −423°F. In internal insulation, however, the insulation material itself protects the bond line, and the bonding agent is subjected to a temperature of only $-100°F$. In the case of the spaced-off insulation, a purge is necessary to prevent air condensation next to the tank. In this spaced-off method, a crack in the insulation invites cryopumping; i.e., the entering air liquifies, pressure drops, and more air is drawn into the crack. Internal insulation is not subject to this disadvantage. Furthermore, it is protected by the tank during transportation and in the exit trajectory. However, for long-term coast missions where the boost phase conductive heat input is of secondary importance compared to the radiative input during coast, spaced-off external insulation appears more attractive.

In the design of the basic structure, the temperature of the propellants must be considered as a major input into the design. Low carbon steels become exceedingly brittle and nearly useless at liquid hydrogen temperatures. Others, such as copper, stainless steel, and aluminium alloys have improved properties at these temperatures. Figure 8 shows the strength of

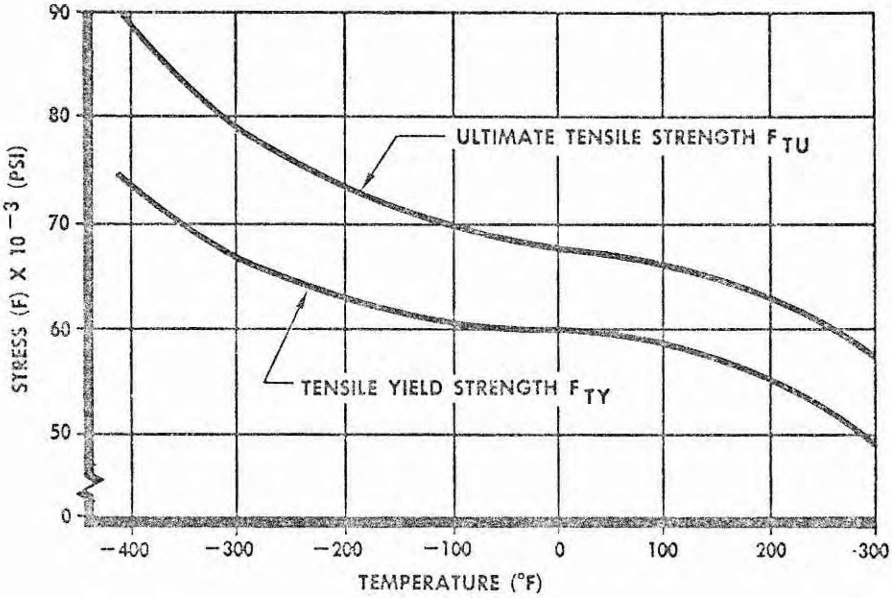

Fig. 8. 2014-T6 aluminium alloy

2014T-6 aluminium as a function of temperature. This improvement in lower temperature strength has been used to advantage in design of the Douglas stages, since the design of elements of the system is predicated on the low temperature strength rather than the room temperature strength. Although some materials exhibit this strength improvement, all properties need be investigated, since improvement in strength can be accompanied by increased notch sensitivity or susceptibility to fatigue.

Component Development

An example of specialized components dictated by the use of a particular engine system may be found in the S-IVB feed lines. These lines, delivering propellants to the J-2 engine, are vacuum jacketed aluminum ducting, rather than more conventional, uninsulated types (Fig. 9). This was included

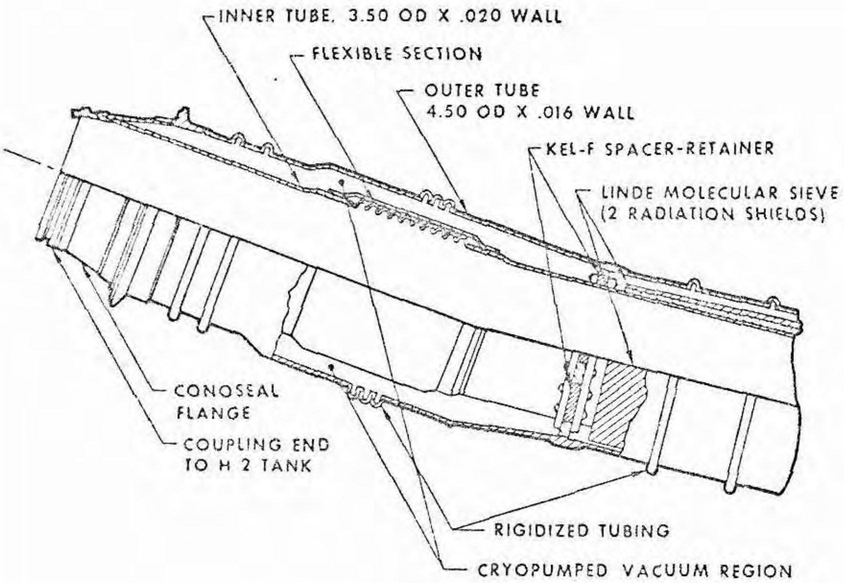

Fig. 9. S-IVB vacuum-jacketed hydrogen field line

in the design to minimize the amount of heat absorbed during propellant transfer and engine start. A vacuum is drawn in these all-welded lines at the time of assembly. This pressure can be continually monitored throughout the assembly and testing of the vehicle. When hydrogen flows through these lines, the cold temperature reduces the pressure in the vacuum jacket even further by condensing out the residual gas.

Throughout the vehicle, similar specialized components have been developed, particularly for hydrogen service, including new types of seals, valves, transducers, etc.

Field Testing

The technique used for field testing of the vehicle is, to a large measure, determined by the requirements of the propulsion system. For example, at Sacramento, California, Douglas and the National Aeronautics and Space Administration have constructed special test areas for the Saturn S-IV and S-IVB programs. The test area to be used in the S-IVB program is shown in Fig. 10. This area is composed of a blockhouse from which

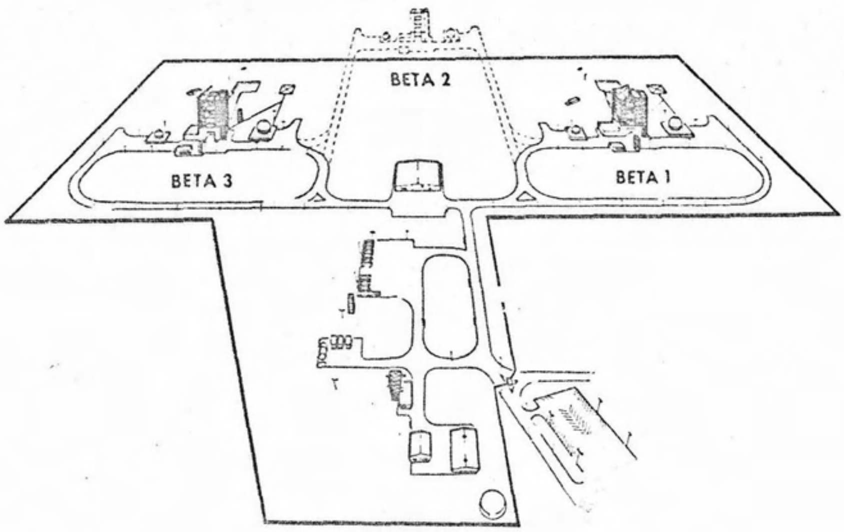

Fig. 10. Beta complex

two firing stands are controlled. One test stand will be used for early battleship testing, the other for vehicle systems testing. In the battleship phase, a heavy-walled steel tank of the same geometric configuration as the flight vehicle will be used to first test the marriage of the engine and vehicle. The question to be answered here is: does this engine, tested only as a component previously by the engine manufacturer, operate properly when fed by the vehicle systems? The other stand employs totally flight-weight systems. On this stand, the effect of engine operation on the real vehicle is investigated to determine that all systems can operate with the engine firing and the engine can be operated by real vehicle systems.

The disposal of hydrogen vent gases is an example of specialized vehicle testing requirements which arise from engine-dictated boundaries. One disposal technique is to dump the hydrogen through a burn stack, in which the hydrogen is bubbled through a column of water and the erupting bubbles are burned. At the Sacramento Test Site, other forms of nonburning stacks have been used. In one nonburning configuration, a flapper check valve was used to prevent air from entering the stack and flowing back to the tank, where it would condense and solidify. Condensation of air in hydrogen tanks can be a serious problem. Contaminants can enter hydrogen tanks through the venting system, the filling system, or, by virtue of contaminants in the hydrogen at the time of tank filling. Liquid air will collect and solidify at the bottom of a hydrogen tank, and in lines and filters, and represents an explosive potential. To preclude the possibility of building up large amounts of such contaminants, procedures have been instituted to drain all propellants and warm the hydrogen tanks at periodic intervals. Then, the tanks are completely purged with helium before refilling.

The disposal of hydrogen gas is sometimes accompanied by spontaneous ignition. For example, in the vent stack just described, there have been several occasions when static discharges have ignited the venting hydrogen. This condition is not especially serious, and was anticipated. It can be detected in the blockhouse with a thermocouple mounted in the stack. The thermocouple is necessary, since the fires themselves are almost invisible. The fires are extinguished by the introduction of an inert gas into the stack to reduce the concentration of the combustible gas.

The extreme volatility of hydrogen and its susceptibility to ignition in the atmosphere has led to the employment of sensitive fire detection systems near critical components on the firing stand. Figure 11 shows the installation of such a system around a hydrogen joint. This continuous fire indicator is similar to the type used on aircraft. If a temperature rise is experienced, an alarm system is activated in the blockhouse. Also employed on our test stand are thermocouples which sample discrete points for unexpected temperature rises which could be indicative of fire. Douglas, and other agencies, have done some experimentation with ultraviolet detectors, which can scan the area remotely. Although such a system is not yet installed, it is believed to be feasible and is currently under active investigation.

Some engines designed for high-altitude operation are faced with nozzle separation problems in sea level test conditions. This necessitates adding an altitude simulation system to the field test site so that the engine operates with lower than atmospheric back pressure. An altitude simulation system was installed in the Sacramento Test Site for testing of the engines associated with S-IV. (This vehicle is to be on the second stage of Saturn I, the first version of Saturn, and employs six Pratt and Whitney RL10A 15,000-lb thrust engines.) The altitude simulation system is based on the use of diffusers. The diffusers are evacuated by a two-stage steam ejector, shown in Fig. 12. With this system, the pressure adjacent to the engine is reduced to approximately 1 psi at time of ignition. The flow of engine gases through the diffusers after ignition is sufficient to keep the pressure reduced.

Special Vehicle System Requirements

Certain ancillary systems were added to the configuration of S-IVB as a result of engine-imposed requirements. Most important among these were the tank pressurization systems which provided propellants to the engine inlets at the proper feed pressures and temperatures. Maintaining proper propellant quality is important, of course, to prevent pump cavitation. The hydrogen tank is pressurized by helium, which is heated in an engine heat exchanger. This helium is stored in titanium bottles submerged in the liquid hydrogen tank. Because of the requirement for restart of the S-IVB J-2 engine, auxiliary systems had to be provided to boost the tank pressure above saturation conditions so that propellants of adequate quality could be provided after the orbital coast period.

Without fully describing the other systems involved, consideration must be given in the installation of an engine to items such as: (1) propellant utilization, (2) gimballing systems and control, (3) guidance interface, particularly as related to thrust build-up and cut-off transients, (4) instrumentation, (5) dynamic loads introduced into the vehicle structure by release after holddown, (6) coupling between engine dynamic combustion modes and vehicle structural or control modes, and (7) inflight disposal of engine prestart chilldown gases.

System Modular Concept

One of the special vehicle systems on S-IVB is the auxiliary propulsion system, which provides slight forward acceleration at main engine start to position the propellants in the tank and maintain roll and attitude control of the stage. Two 1750-lb thrust ullage engines furnish axial thrust before initial start (S-IVB stage separation) and restart of the main engine. During orbital flight, two 150-lb thrust ullage engines furnish axial thrust to assure that the liquid hydrogen propellant is properly settled in the stage tank prior to venting. Control of pitch, yaw, and roll is maintained by two sets of three 150-lb thrust attitude engines. These engines are mounted in modules; each module contains one 1750-lb thrust engine and four 150-lb thrust engines. Figure 13 is a view of the S-IVB stage with the auxiliary propulsion system modules attached.

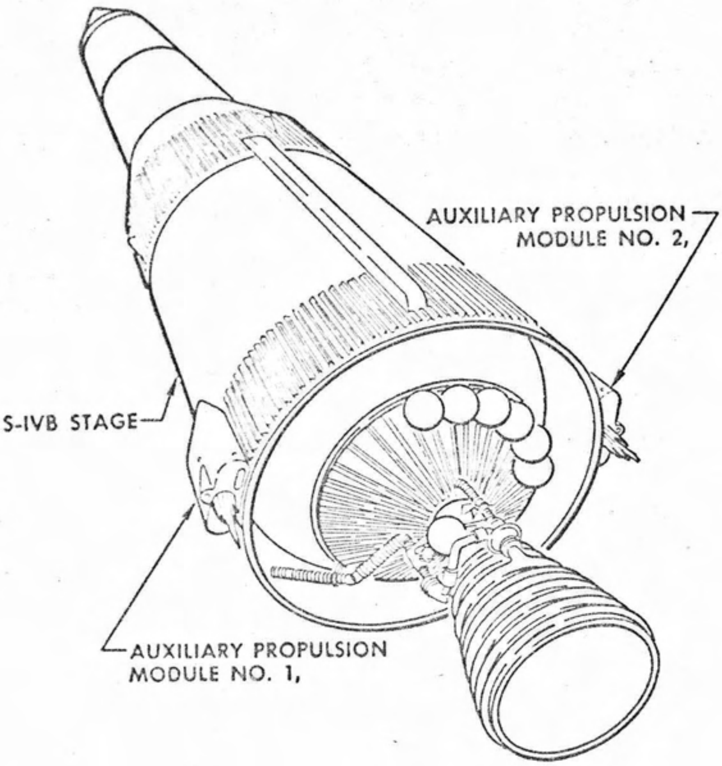

Fig. 13. Saturn V propulsion control system Saturn S-IVB stage

A positive expulsion system is used to assure that the hypergolic propellants will be supplied to the engines under zero *g* conditions. This system, shown in Fig. 14, consists of welded stainless steel wafer bellows in the propellant tanks, which expel the propellants into a manifold. The upper end of the tanks is closed by the high pressure sphere, which is used for storage of the 3000-psia helium for the pressurization system.

The 150-lb thrust, ablatively cooled rocket engines are being developed for Douglas by the Tapco Division of Thompson Ramo-Wooldridge Corporation. The design firing time for these engines is 30 min for steady-state operations, or pulse firing of up to 10 pulses/sec.

The 1750-lb thrust rocket engines are being developed for Douglas by the Marquardt Corporation. These engines, which are similar in design concept to the 150-lb thrust engines, are ablatively cooled and have quadruple propellant valves. The design firing time for these engines is 250 sec for steady-state operations.

Just as the engine has given the vehicle designer problems which must be solved, the vehicle design provides many difficult boundaries to the engine designer. For example, the vehicle geometry constrains the physical engine parameters, such as size, weight, thrust, specific impulse, materials, etc. Perhaps of equal importance are the environmental constraints which the vehicle imposes on the engine. For example, components of the J-2 engine, when installed on the S-IVB, will experience temperature extremes of

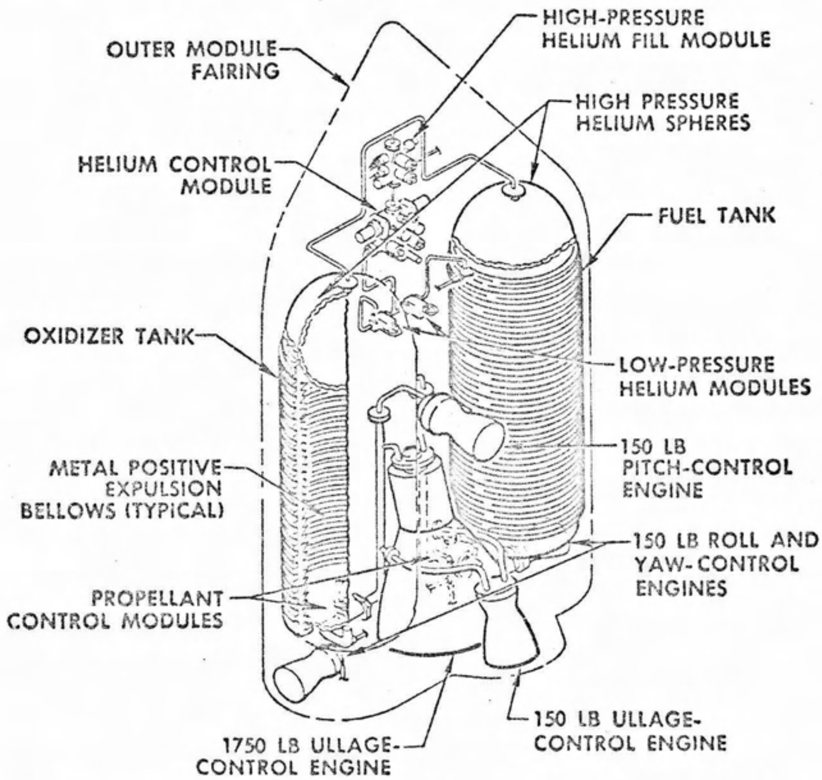

Fig. 14. Saturn V auxiliary propulsion control system module Saturn S-IVB stage

from $-423°$ to $5400°$F, vibration levels of 24 g's in the frequency range from 35 cps to 370 cps, vibration levels of 60 g's at higher frequencies, and shock levels of 165 g's.

This environment arises in part from aerodynamic buffeting and dynamic inputs from the burning phase of the lower stages. Also, consider the environment of a single engine which operates as a member of a cluster. It must withstand the loads imposed by its adjacent brothers while it contributes to the racket that they must endure.

Among the general requirements which a space vehicle imposes on the engine design are: (1) tailoring of the thrust time curve, (2) mixture ratio control, and (3) reliability and maintainability.

Tailoring of the thrust-time curve is important in several areas. For example, some engines, during their start cycle, may overshoot nominal thrust by 20%. If an engine were to perform this way, the vehicle's thrust structure would be designed around the overshoot condition rather than the burning period, when the thrust presumably would be nominal. This could result in heavy and inefficient structure. Therefore, the vehicle designer tries to impose a requirement for minimum overshoot. Similarly, at the other end of the thrust curve, the vehicle designer wants an engine that yields the same thrust tailoff curve every time a particular engine is cut off and one which is identical between different engines. Minimum engine-to-engine variation is important in the clustered configuration so that turning transients are not generated at cutoff. Such transients could make separation of a succeeding stage very difficult. Repeatability of thrust tailoff in successive firings of a single engine is important in the prediction of the total impulse gained after cutoff command. This figure is used in setting the flight guidance equations so that the cutoff command can be generated ahead of the desired final velocity. Without good cutoff transient repeatability between successive firings, this "velocity-gained-after-cutoff" figure might have a large uncertainty.

Thrust and performance repeatability is also important to the vehicle designer. Calibration firings at the engine manufacturing plant determine the particular propellant flow requirements for a given engine. These calibration data are used in preflight predictions to generate information required for accurate preflight propellant loading and trajectory computations. If, on subsequent firings, engine performance varies from that predicted for the flight conditions, trajectory perturbations and path uncertainties will result.

Velocity gained in a stage is a strong function of the stage empty weight. Therefore, where velocity gained is important to a vehicle designer, he strives to minimize empty weight. Unburned propellants can add to the empty weight and seriously limit the velocity potential of the stage. Therefore, the vehicle designer usually wants to run out of both propellants simultaneously. This is approached in one of two ways: (1) open loop, by accurately predicting propellant flow rates and pre-loading precisely the correct amount of propellants, or (2) closed loop, by including sensing elements which detect the amount of each propellant remaining and adjust the flow of one or the other to guarantee simultaneous exhaustion. In either case, special requirements are imposed on the engine. Open loop systems demand very precise calibration data; close loop systems demand a means for controlling propellant flow without adversely affecting other engine operating parameters.

Reliability and maintainability are words which take on new meaning in a system which has 98 engines. In almost all cases, the engines must function properly on command or the mission will be aborted. The vehicle designer expects to get an engine which will perform on demand. Reliability usually comes with experience. Rocket engines are better today than we might otherwise have a right to expect from a simple analysis of our experience with them. Figure 15 compares the total flight hours of the

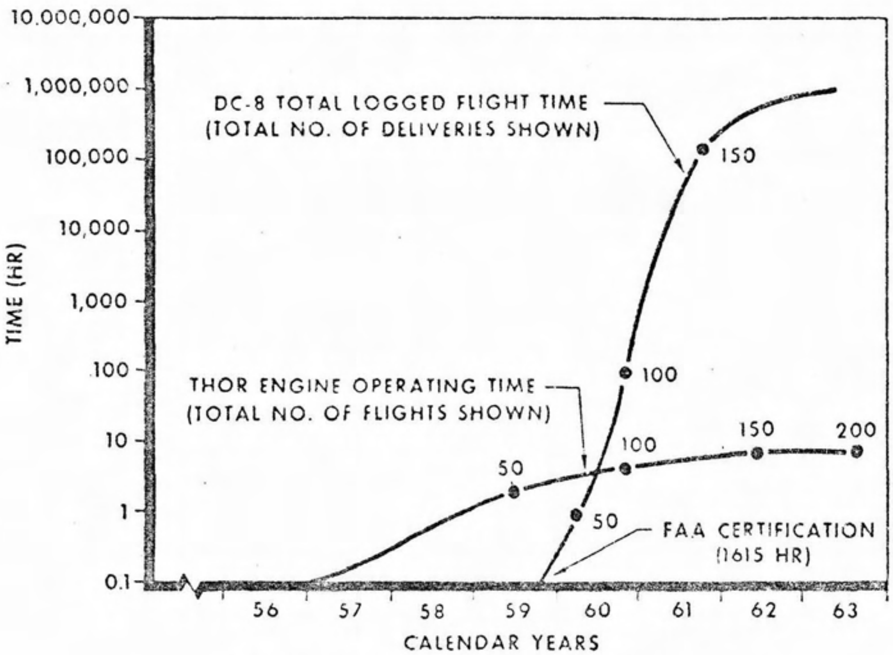

Fig. 15. Comparative engine operating times

DC-8 to the Douglas-built Thor. It is interesting to note that if all rocket engine operating times were to be considered, there are still differences of orders of magnitude in total flight time. From the standpoint of operating time alone, the rocket is still back in the hand crank, manual spark advance days.

Proper engine design considers preflight maintainability and checkout. These considerations may be even more important when long term operation in space is considered. Then, the vehicle designer will ask the engine manufacturer to provide an engine which can be checked in space, repaired in space, and perhaps tested in space. Modules will be external for access; systems will be plug-in; instrumentation will be self-checking and foolproof.

Fitting the engines to the vehicle and vehicles to the engine are design challenges today. The problems involved are recognized and are being solved today through cooperation between engine manufacturers and engine users.

Solving these interface problems today, determining how best to use an engine to accomplish a particular mission, will provide answers to questions which will be raised in the design of future planetary spaceships and their future engines.

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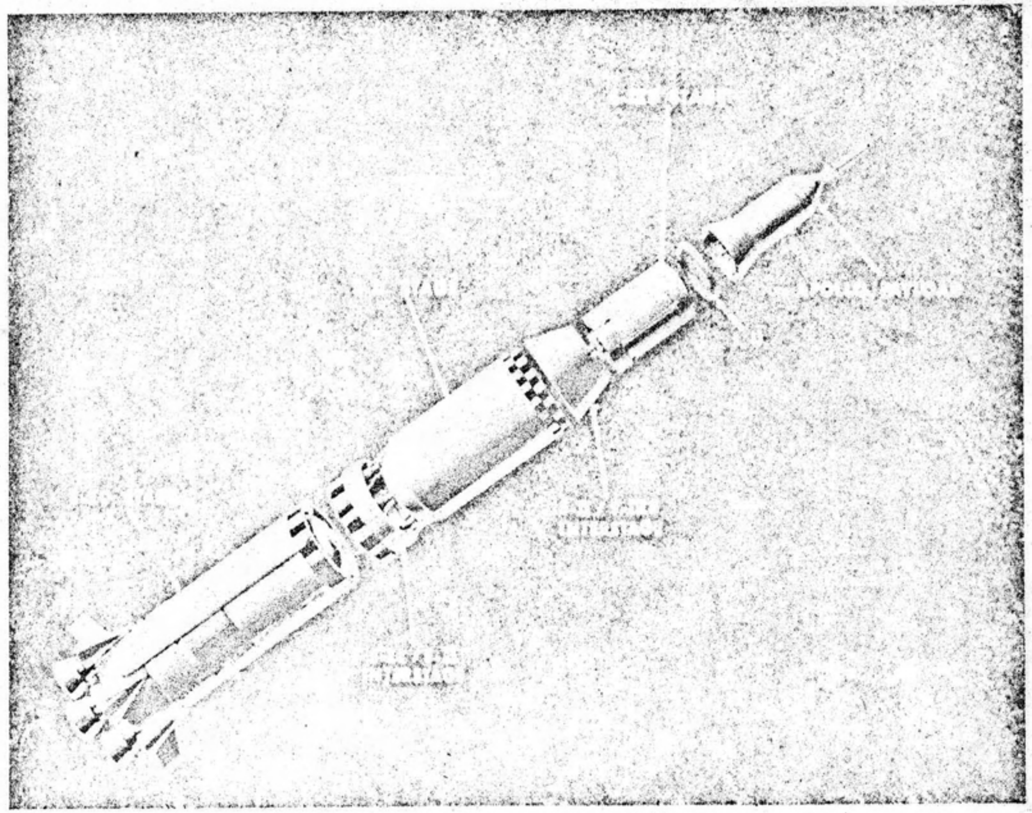

Fig. 1.

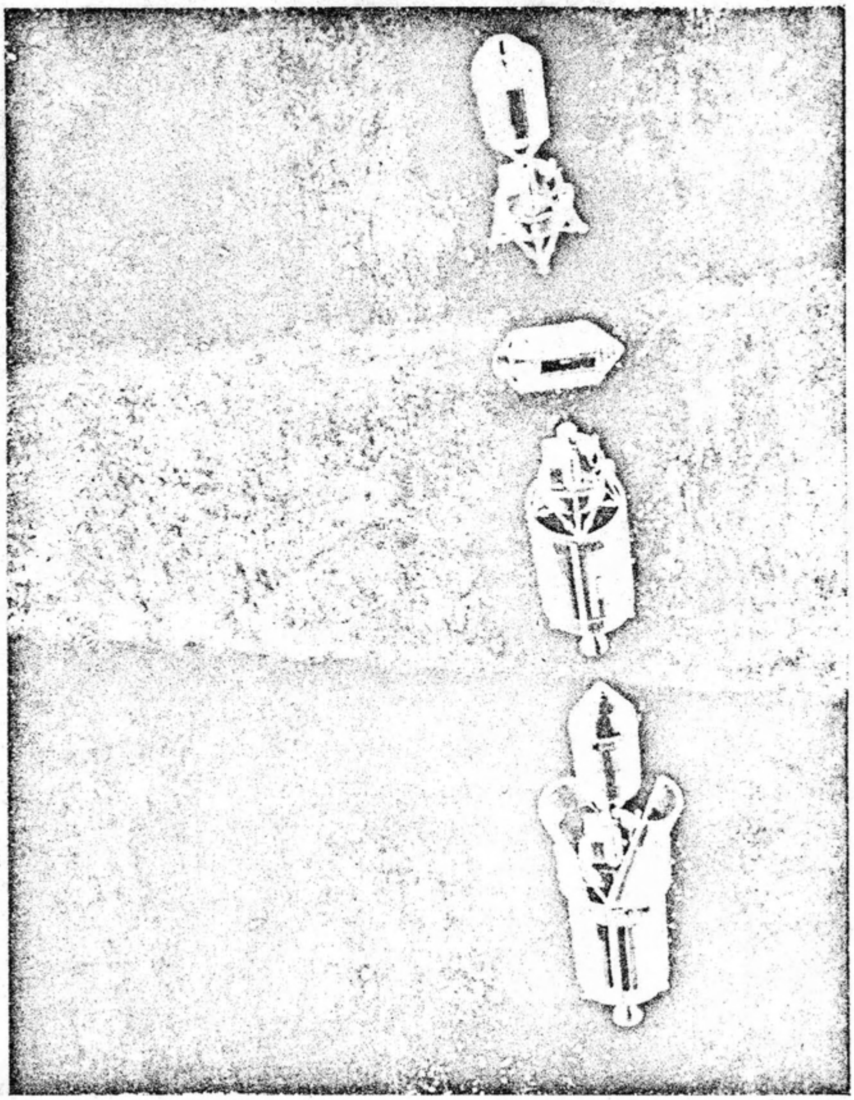

Fig. 3.

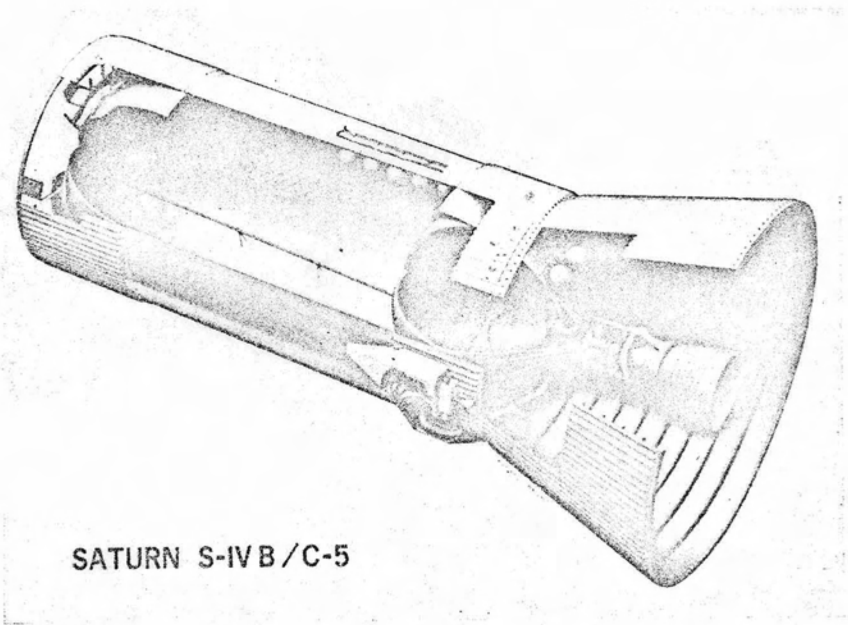

Fig. 7.

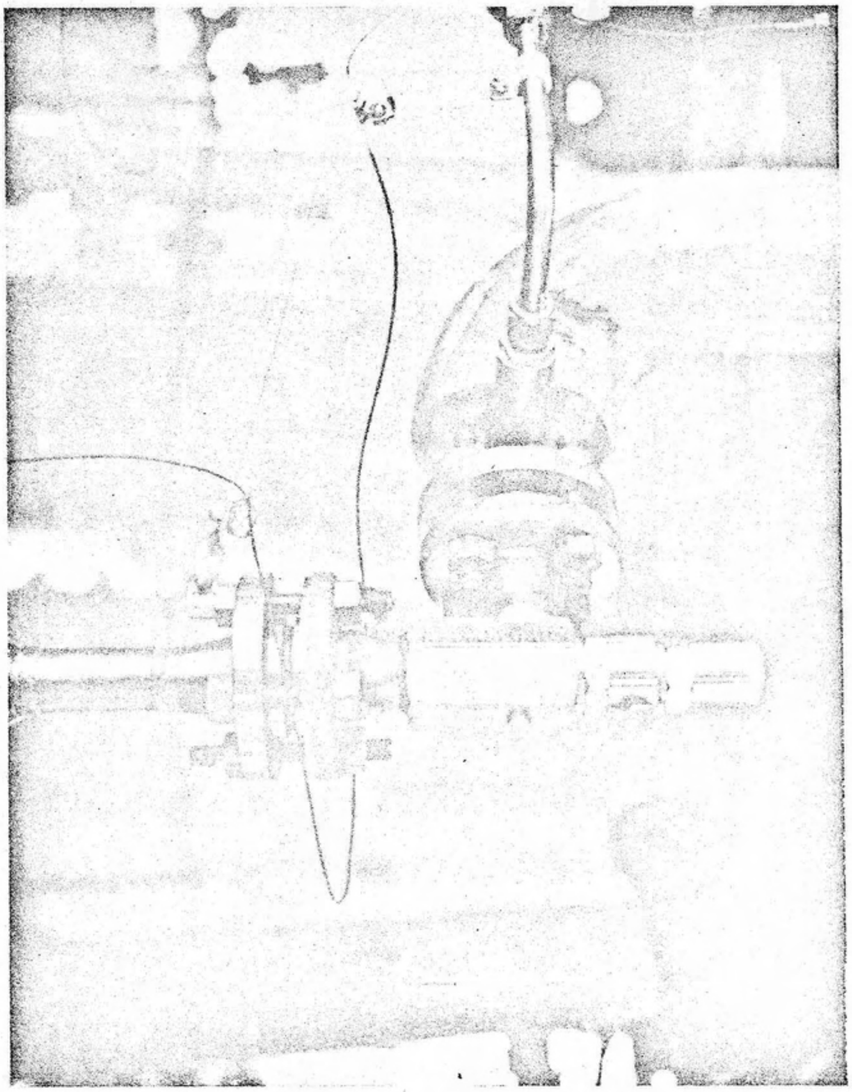

Fig. 11

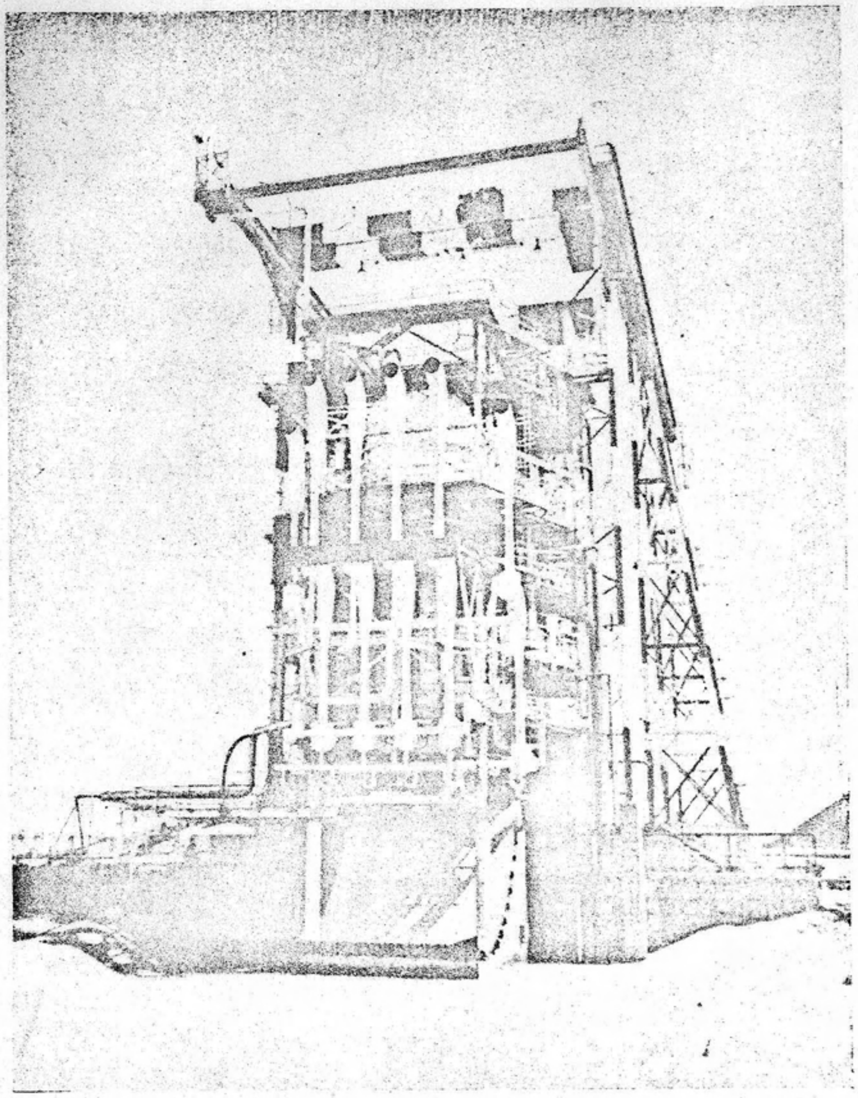

Fig. 12