Production acceptance tests (PAT)

All components were subjected to an acceptance test sequence prior to use. Thrust chambers and 3-way valves were individually tested. Summary results are shown in Table 10. Each REM underwent P_c calibration to limit torque unbalance to <6 by changing thruster trim orifices at the inlet to the thrust chambers, as required. The average P_c's of opposing coupled thrusters were matched to within 10 psia (Table 11). Thrust chamber hot response in REM S/N 02 and S/N 03 from fluidic signal to 90% of P_c fell between 24 and 31 msec for all thrusters during all pulsing operations. When corrected for use of the modified 3-way valve spring these response times become 27 and 34 msec. Corrected off response times of 27 to 39 msec were observed.

Conclusions

1) A closed-loop roll-rate control system was designed and fabricated using fluidic sensing and logic to command monopropellant hydrazine REMs. 2) The REM, as designed, fabricated and tested successfully demonstrated compliance with the performance specification. 3) A common pneumatic power supply is feasible for sensing and logic power, control valve actuation, and propellant expulsion. 4) The conventional electrical solenoid valve can be replaced with a pneumatically actuted 3-way control value to eliminate electrical interfaces. Pneumatic valve responses of less than 20 msec are practical. 5) Response times of less than 50 msec were demonstrated with REM using the 3-way valve. 6) Steady-state specific impulse values greater than 200 sec were demonstrated at 2.0 psia amb pressure with a nozzle area ratio of 10. 7) Proven components in the REM reduced technical risk for the flight feasibility program, but significant reductions in size and weight can still be achieved by integrating functions and optimizing components.

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Lunar Flying Vehicle Propulsion System

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The optimization of a propulsion system for a Lunar Exploration Flying Vehicle (LFV) is discussed in order to report a special approach to extended system life, reliability, and flight safety. Pressure-fed engine and system design parameters are simultaneously optimized for minimum weight delivered to the moon and maximum LFV operational range on the moon. The resulting optimum engine chamber pressure and nozzle expansion ratio are 80 psia and 40:1, respectively. Refractory metal engine life is assessed in terms of time/temperature history on the basis of test data, concluding that a 3.5 hr burn time requirement can be met with a maximum operating temperature of 2250°F. Redundant engines are compared with derated (low operating temperature) engines for equivalent flight safety and reliability. Propulsion system design for derated engines with preflight check capability is proposed to reduce operational risks to the level of light aircraft operation on Earth.

Introduction

THE Lunar Flying Vehicle (LFV) is a one-man rocket propelled flight system for lunar exploration away from the Lunar Module (LM) landing site. Operations analysis and vehicle studies (Ref. 1) produced the optimized LFV configuration shown in Fig. 1. This configuration incorporates a pressure-fed propulsion system sized for 300 lb of useable N_2O_4-0.5 N_2H_4/0.5 UDMH propellants and two 150-lb throttleable engines to lift a payload of 520 lb (including the astronaut). The outboard engine arrangement incorporates articulated single-axis gimballing for pitch and yaw control. Roll control is provided by differential throttling of the engines. The LFV is flown, in flight attitudes similar to a helicopter, by continuous thrusting. High thrust is used to lift off, climb to altitude, and pitch over to accelerate along the flight path. The LFV then cruises inertially by rotating to vertical and reducing thrust to the level required to balance lunar gravity. High thrust is again required (in a rotated attitude) for deceleration followed by continuously lower thrust for descent and soft landing. To provide for 60 to 150 such flights, a total engine service life of 3.5 operating hr is required.

Similar one-man jet and rocket powered vehicles have successfully demonstrated free flight on Earth. Bell Aerospace has developed the rocket belt and jet belt (Ref. 2); North American Rockwell has experimented with a one-man flying vehicle tethered to simulate Lunar gravity (Ref. 3); and flight simulator investigations at NASA Manned Space Flight Center and Langley Research Center have demonstrated the feasibility of a rocket powered one-man flying vehicle in the lunar environment.

The design of an optimum propulsion system for this LFV configuration demanded identification of the design features for the best combination of weight, performance, extended system life, reliability, flight safety, and operational techniques. This paper reports the results of this portion of a one-man lunar flying vehicle study conducted for the NASA Manned Spacecraft Center under Contract NAS9-9044.

Basic System Model

The basic propulsion system under study is shown schematically in Fig. 2. The arrangement of components corre-

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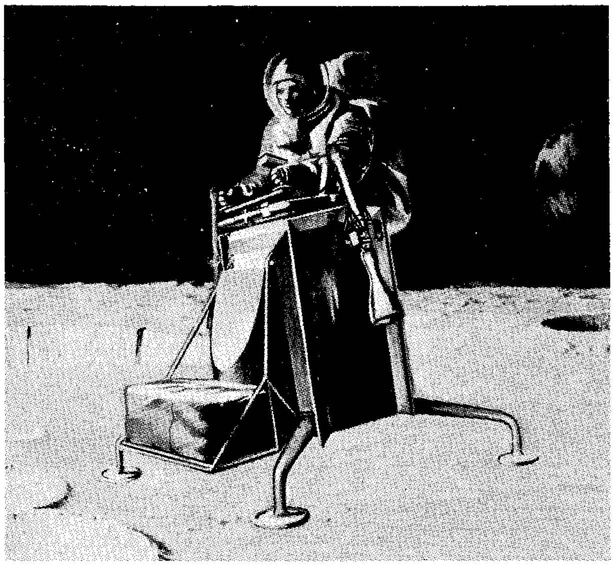

Fig. 1 Optimized lunar flying vehicle (LFV) configuration.

sponds to their locations in the vehicle. It is a classic Apollo type pressure-fed propulsion system with components based on current technology. To model this propulsion system for weight and performance optimization, the following working assumptions were employed:

1) The pressurant gas was assumed to be helium with maximum pressure of 4500 psia and a minimum pressure of 1.7 times engine feed pressure. In determining quantity required, compressibility was considered, and a polytropic expansion coefficient of 1.2 was assumed.

2) The gas storage bottles were assumed to be made of 6A1-4V titanium alloy with an overall safety factor of 2.8, giving a bottle weight of 0.0182 $lb/in.^3$ of volume for the required working pressure of 4500 psia. Spherical tanks were used.

3) The propellant tanks also were made of titanium alloy of 130,000 psi yield strength using a safety factor of 1.33 on yield strength, plus a scratch allowance of 0.005 in. on skin thickness. A minimum fabrication thickness of 0.017 in. was assumed. Safety factors and minimum fabrication thickness are those used for the Apollo LM, Service Module, and Command Module attitude control positive expulsion tank shells. The design of the LFV tanks was based on these flight proven propellant tanks.

The tanks consist only of titanium shells with slosh and vortex baffles. No positive expulsion devices are needed since, continuous thrust provides propellant orientation.

Tank volumes were based on a total propellant load of 306 lb at an oxidizer/fuel (O/F) ratio of 1.6 and densities at 120°F. The optimization study mixture ratio of 1.6 was

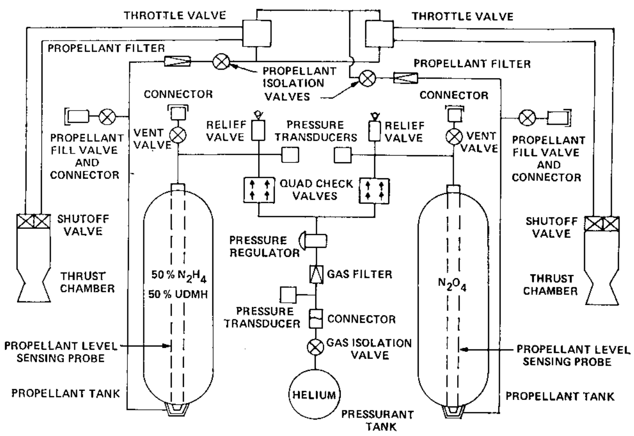

Fig. 2 Schematic diagram—LFV propulsion system.

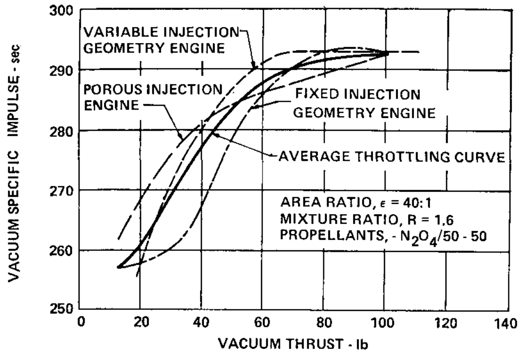

Fig. 3 Throttleable engine performance.

selected for maximum engine performance to identify the maximum LFV range or velocity increment, ΔV. However, dry weight optimization is not sensitive to O/F ratio in the range of interest (1.3-2.0) and the selection of 1.6 was made as a midrange value. The optimum design points and system comparisons are essentially the same throughout this O/F ratio range. For stressing purposes, the tank working pressure was taken to be 50 psi above nominal engine feed pressure. (The engine feed pressure is dependent on injector design and rated chamber pressure.)

4) Radiation cooled engines with fixed injection geometry were selected for minimum weight and maximum reliability within the current state-of-the-art. The weights of engine components were curved through several point designs in the design range of interest. Throttle and propellant valve weights are dependent upon thrust level, and thrust chamber weight is a function of thrust, chamber pressure, and nozzle area ratio.

5) It was assumed that the weight of the remaining system components, gas valves, lines, propellant lines, fill valves, etc., would be constant, and 16.6 lb was allowed for these items for the twin-engine system.

6) To establish the vehicle velocity increment, the vehicle weight breakdown in Table 1 was assumed.

7) The variation of specific impulse with design chamber pressure and nozzle area ratio was based on theoretical performance calculations that included chemical reaction rate variation with chamber pressure. The basic computer program employed a search subroutine to identify the nozzle station (or break point) where frozen composition is fully developed by local pressure, temperature, reaction rates, and velocity. The basic computer program then calculated theoretical performance with the simplifying assumption of shifting equilibrium flow up to this break point and frozen composition in the remainder of the nozzle. This data was then corrected by exact one-dimensional kinetic performance calculations at several points in the range of interest, and an assumed engine efficiency of 92.2% was applied to obtain rated engine specific impulse.

Variations of specific impulse (I_{sp}) with thrust are given in Fig. 3 for one variable injection geometry engine and two fixed geometry engines operating at the maximum performance mixture ratio of 1.6. These data were reported from firing tests by three different engine vendors (Refs .4–7). The throttling data show little difference (~2%) in performance of the three injector types at thrust levels above 50% where

Table 1 Assumed weight breakdown

Structure weight	100 lb
Payload	150 lb
Astronaut with environmental backpack	370 lb
Residual propellant	6 lb
Usable propellant	300 lb
Total (less propulsion system)	926 lb

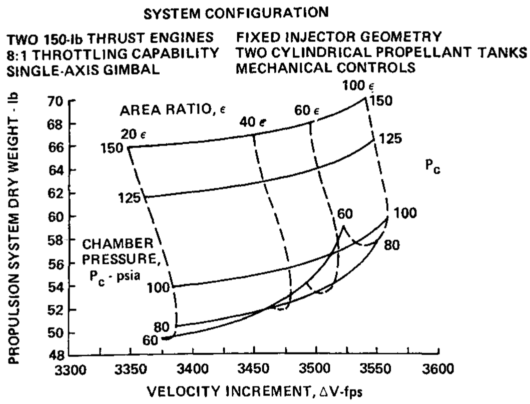

Fig. 4 Propulsion system optimization.

mission duty cycle (MDC) evaluation indicates that most of the propellant will be consumed. An average nominal curve for throttling performance was therefore assumed.

Chamber Pressure and Expansion Ratio

Tradeoff data were generated by calculating the propulsion system dry weight and vehicle ΔV capability for ranges of engine design chamber pressure (P_c) and nozzle expansion ratio (ϵ). Figure 4 shows that the minimum-weight pressure is near 80 psia, and that weight begins to increase appreciably for $\epsilon > 40$ ($\Delta V > 3480$ fps). Therefore, we select $P_c = 80$ psia and $\epsilon = 40$.

The inertial ΔVs for Fig. 4 were calculated for engine operation at full-thrust I_{sp}. Since ΔV is directly proportional to I_{sp}, the curves may be adjusted to the effective I_{sp} of any MDC by referring to Fig. 3 and correcting the abscissa of Fig. 4. However, such adjustments will not affect either the character of the curves or the selection of P_c near 80 psia and ϵ near 40.

Engine Life and Reliability

The redundancy approach to high reliability is feasible for this mission, but it is compromised by the resulting requirement to sense and diagnose failures and initiate corrective action before critical LFV attitudes or flight paths are reached. A better approach is to design the engine for high intrinsic reliability by minimizing the number of components and providing large design margins. The number of components can be minimized by use of manually operated propellant and throttle valves, fixed injection geometry, and radiation cooling. The valves and injector can be overdesigned to provide large operating margins. However, the flight safety and reliability of the radiation-cooled thrust chamber is dependent on the thermal margin.

The radiation-cooled engine maximum operating temperature experience of Bell Aerospace is given in Fig. 5 as a function of engine performance (c^*). This family characteristic data is based on six different engine designs operating on N_2O_4-MMH and N_2O_4-0.5 N_2H_4/0.5 UDMH propellants. The observed spread in maximum operating temperature at any given performance level is attributed to design and operating variations in mixture ratio, chamber pressure, engine geometry (e.g., L^*), and extent of barrier or film cooling. Published data from other engine vendors (TRW and Marquardt) also appear to fall within the envelope shown in Fig. 5 (Refs. 6, 8, 9).

The operating temperature limit of refractory metal (i.e., Cb and Mo) radiation-cooled thrust chambers is defined by the 3100°F capability of the silicide coatings required for oxidation protection. A 600°F margin on this capability has proven satisfactory for man-rated applications. This is

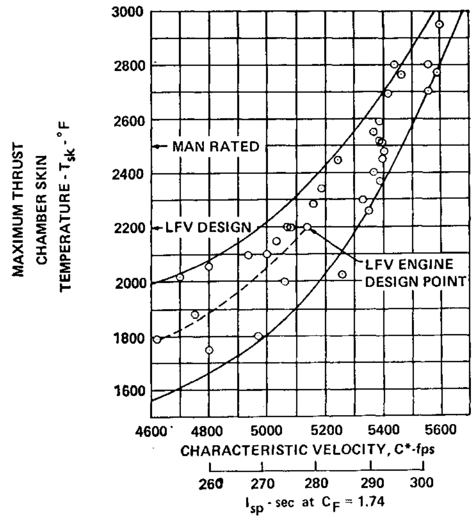

Fig. 5 Engine operating temperature vs performance.

substantiated by Bell experience with the fabrication and test of over 500 radiation-cooled engines (Cb). No failure was ever observed during operation at or below 2500°F.

Increasing this man-rated temperature margin by 50% to 900°F will, in the experience and judgment of Bell, provide equal safety, longer life, and lower probability of engine failure, than will redundant thrust chambers. Bell engine operation at and near 2200°F indicated unlimited life to the extent tested (perhaps due in part to the fact that this is below the 2400°F melting temperature of any columbium oxide that may form). Published data for TRW and Marquardt engines appears to confirm this Bell data as shown in Fig. 6. The silicide coating curve defines the potential life of these radiation-cooled engines as a function of operating temperature. The line of engine data points defines the portion of this potential life that has been demonstrated by engine test without failure. The boundary defined by the cross hatched area in Fig. 6 represents the operating time of the LFV engines at each wall temperature. The steps in this boundary reflect reduced temperatures at reduced thrust settings. The figure shows that the time-temperature life required of an LFV engine designed for 2250°F maximum operating temperature lies well within currently demonstrated technology.

Low engine operating temperatures can be achieved by reducing chamber pressure or mixture ratio and film cooling the combustion chamber wall. The propulsion system optimized at the low rated chamber pressure of 80 psia provides most of the benefits of this effect without penalty. Film cooling can be very effective without excessive engine

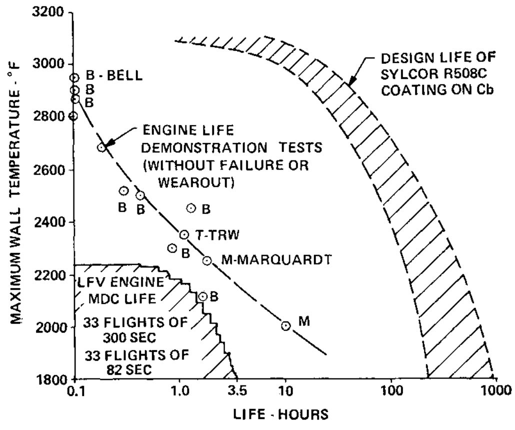

Fig. 6 Engine life vs operating temperature.

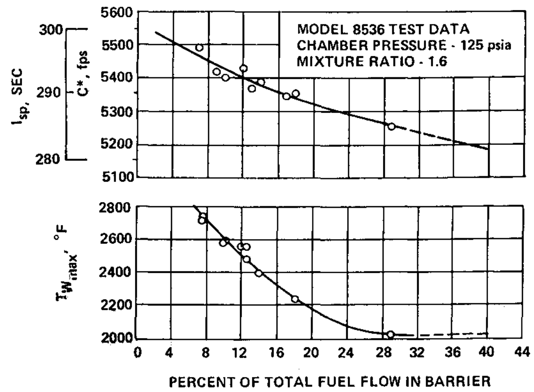

Fig. 7 Fuel film cooling—300-lb engine.

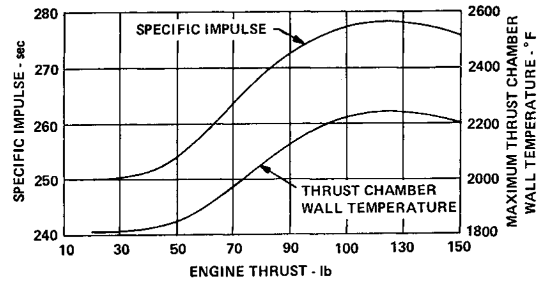

Fig. 9 150-lb thrust engine performance and maximum wall temperature.

performance derating as shown in Figure 7. However, mixture ratio reduction is more straightforward, so a combination of film cooling and O/F reduction was considered.

The calculated change in maximum engine operating temperature with mixture ratio is shown in Fig. 8 for the existing Bell 100-lb throttleable engine. This curve is indexed to the measured current operating temperature of this engine at a mixture ratio of 1.6 (where performance is maximum) and confirmed at a mixture ratio of 1.31 by test data from the TRW 100-lb throttleable engine. This figure indicates that the maximum operating temperature can be reduced to 2420°F by reducing the operating mixture ratio to 1.3, with a loss of 4 sec of specific impulse.

With this change to O/F = 1.3, existing film cooled engine test data indicates that the maximum film cooling required will be ~20%. The predicted performance and operating temperature of this engine design are shown in Fig. 9.

The selection of an operating mixture ratio of 1.3 also provides the significant advantage of maximizing the propellants available from the residuals of the LM Descent Stage, as shown by recent studies conducted by TRW in support of the Bell LFV study. The TRW analysis was based on the landing engines for LM-6 through LM-10, and it considered the limit cases of 6688 fps mission ΔV for direct flight touchdown (no hovering) and 7180 fps mission ΔV for touchdown after maximum hover. For these limits, the residual fuel loads were 1442 lb at O/F = 1.48, and 597 lb at O/F = 1.33, respectively. Since the impact of LM residual mixture ratio on LFV operation is maximized when the LM residuals are minimum (i.e., at 7180 fps ΔV) the optimum O/F for propellant utilization is 1.33.

Final Selection

This discussion, thus far, has shown that derated engines designed for O/F = 1.3 and operating at a maximum temperature of 2200°F are comparable in flight safety and reliability to redundant engines operating at a mixture ratio of 1.6 and a temperature of 2500°F to develop maximum performance.

Based on this common denominator of equivalent safety and reliability, a tradeoff analysis was conducted to compare the redundant and derated engine designs in weight and range capability. The results are shown in Fig. 10. The same techniques and assumptions as previously stated were employed here. However, the work was limited to P_c = 80 psia, since that was previously shown to be near optimum. Figure 10 shows that, at the equal range or ΔV point, the derated engine design is 12 lb lighter than the redundant engine design. Compared at ϵ = 40, the derated engine design is 19 lb lighter, but 134 fps (4%) lower in ΔV capability.

The refinement of these weights, with detailed estimates of fixed weight elements such as tank baffles, mounts, lines, and malfunction detection equipment, yields the comparison in Table 2, which shows that the derated engine system is 18.2 lb (23%) lighter than the redundant engine systems for a ΔV penalty of only 4%. The development risk of the derated engine is significantly reduced, and the development and operational cost of the derated engine is estimated to be approximately half the cost of the redundant engine case, which includes double the number of engines per system, increased development testing for higher performance, and development of a malfunction sensing and correcting subsystem. The derated engine system was therefore selected for the LFV application.

Engine Control Considerations

The selected LFV configuration requires propellant flow control located on a control quadrant that is relatively remote from the outboard engines. For differential throttling for roll control, the best location of manually operated engine throttle valves is on the control quadrant. However, for reliable and reproducible engine start and shut-down operations, the propellant "manifold" volume between the shutoff valve and the injection face should be held to a minimum. Therefore, the best location of the shutoff valves is at the thrust chamber. It was therefore concluded that the best control arrangement for the LFV outboard engines is separate throttling and shutoff valves.

The most reliable propellant flow control is a simple bipropellant variable-area cavitating-venturi valve. Cavitation decouples flow control from engine operating variables for simple open loop calibrated thrust control. Throttle valve design simplicity, large design margins, and the follow-

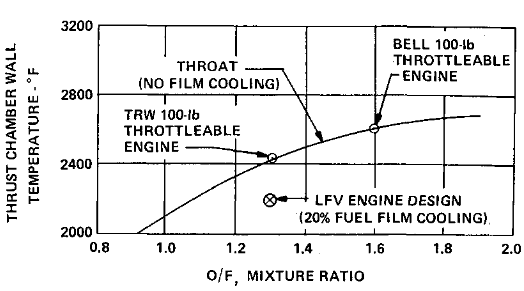

Fig. 8 Steady-state wall temperature vs mixture ratio.

Table 2 Propulsion system comparison

Comparison element	2 derated engines	4 redundant engines
Dry weight, lb	62.9	81.1
Velocity increment, fps	3240	3374
Required throttle ratio	$\frac{6}{1}$	$\frac{9}{1}$
Specific impulse, sec	275 → 250	293 → 256
Approximate cost ratio	1	2

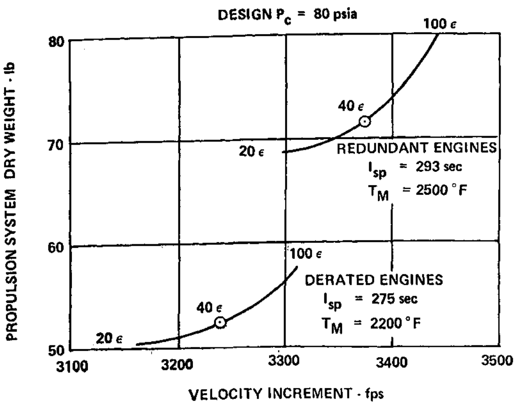

Fig. 10 Engine reliability options.

ing operational considerations indicate throttle valve redundancy will not be required for flight safety and reliability in the LFV application: 1) throttle valve and actuation system exercised under operating pressures in preflight checks-abort ignition of binding is detected; 2) any comtamination cleared by wide open throttle operation during ascent; 3) in-flight binding precluded by preflight check; 4) degraded flow modulation due to misalignment or hard contamination compensated by trim to match second throttle valve output; and 5) Internal leakage—not applicable.

The propellant shutoff valve should be a normally open device to preclude thrust loss due to valve failure in flight. Since post-flight shutoff redundancy is provided by the master propellant isolation valves, design for simplicity with large margins, together with the operational considerations summarized in Table 3, indicates that LFV flight safety and reliability can be provided without propellant shutoff valve redundancy.

Pressurization Options

The number of possible pressurization options to maximize flight safety and reliability is quite large—including complete pressurization subsystem redundancy, quad-redundant regulator, and large propellant tank design ullage for unregulated blow-down operation capability. However, all of these options penalize the system on a major measure of merit propulsion system dry weight.

The pressurization system, shown in Fig. 2, provides the required safety in the LFV application. Each flight will be preceded by the pressurization system checkout (inherent in system activation) by opening the gas isolation hand valve and readout of resulting gas and propellant tank pressures. All takeoffs will therefore be with normal pressurization system operation.

The only moving parts during flight are the check valves and the regulator. The check valves are quad-redundant and the regulator is backed up by the relief valves and the tank ullage volume. The immediate effect of a regulator-open failure is over-pressurization of the propellant tanks to the relief valve setting and venting of the gas tank down to propellant tank pressure (through relief valves). The propellant tank pressure will then begin to decay by blowdown of both the gas tank and propellant tank gas as propellant is consumed. The resulting excursion in propellant tank pressures can be compensated by throttle settings for a safe landing. In the case of a regulator-closed failure, propellant tank pressure blows down from the regulator setting as propellant continues to be consumed. The resulting tank pressure excursion can be compensated by throttle setting changes to produce the thrust sequence for a safe emergency landing.

There is intrinsic compensation in this scheme of backing up the regulator with blowdown operation. As the burn time required for a safe emergency descent and landing increases with altitude and velocity, the propellant consumed to reach increasing altitudes and velocities has left increasing tank ullage volumes of pressurized gas for increased blowdown operation capability. However, the exact value of the design ullage volume required to provide safe emergency landing in all portions of all possible flights must be determined by operations analysis.

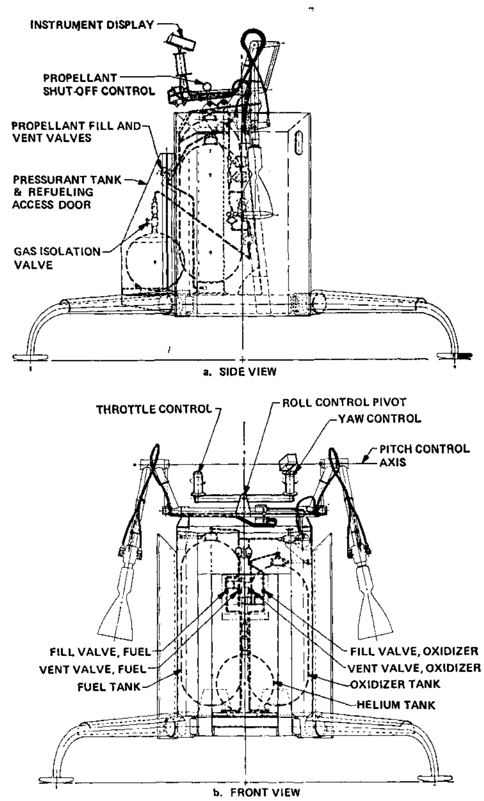

Fig. 11 LFV inboard profile.

System Operation

Figure 11 shows propulsion system installation drawings. The Bell Model 8414 engine was employed in this preliminary design effort because it is currently configured for the LFV. However, reconfigured versions of other engine candidates also apply, without change in system design.

Of particular importance in selecting the arrangement and mounting of components are the considerations of system operation during flight and refueling. The design allows all refueling operations to be done from the front of the vehicle. The front panel may be opened to give access to the helium bottle and gas isolation valve and to the propellant fill and vent valves. As the first step in refueling, the helium isolation valve is closed, and the helium tank removed by

Table 3 Engine shutoff valve considerations

Failure mode	Cause	Effect
Fails to open for start-up in flight	Stuck closed, actuator jammed or broken actuator	Safe-abort flight
Activation failure in flight	Broken actuator	Safe-valve normally open, continue flight
Fails to close on shutdown	Stuck open, actuator jammed or broken	Safe—close isolation valve or run to propellant depletion
Internal leakage	Seat scratch or contamination	Safe—close isolation valves

unlatching the tank straps and disconnecting the gas supply line. The refueling system and procedure must be designed to prevent propellant spills which mght contact the astronauts. The dangers from spills are: 1) the astronauts pressure suit might be damaged by the oxidizer, and 2) propellants on the suit might be carried into the LM and contaminate the cabin atmosphere. Use would be made of quick disconnects with integral shutoff valves to keep propellant leakage to a minimum at all connections necessary during refueling. Available data on such connectors indicates that the total propellant leakage during the refueling operation would be limited to a few drops. To further ensure against contamination by the propellants during refueling, the tanks are vented through long lines which lead away from the LFV. The fuel tank is refilled from the LM descent stage by connecting the resupply line to the LFV fuel-fill connector. The tank is filled by opening the fill valve and venting the tank. The oxidizer tank is filled in the same manner. The last step in the refueling procedure is replacement of the helium bottle with a charged bottle.

The LFV is operated with a basic philosophy of runup and checkout before commitment to flight. This approach is best illustrated by operational procedures. After a visual inspection the LFV is approached from the front to accomplish pressurization before mounting. The astronaut initiates pressurization by opening the helium isolation valve and confirms satisfactory regulation and tank pressurization by readout on the oxidizer and fuel tanks pressure transducers. The astronaut then mounts the LFV from the rear (see Fig. 11) and straps in. From the flying position he then opens the oxidizer and fuel isolation valves to admit pressurized propellants to the throttle valves and into the flex lines to the shutoff valves located at each engine.

With the propulsion system pressurized, the pitch, yaw, and roll controls can be fully excerised without engine firing. The engine mount yoke is traversed fore and aft for pitch. Yaw motion is controlled by cables to engines differentially pivoted at the engine mount points, and roll control is performed by differential throttle-valve actuation. Thrust modulation control is obtained by full stroking the throttle valves. Satisfactory operation of all these flight controls is established by visual observation of control movements and feel of the correct level of control forces.

The throttle valves are then closed to minimum thrust for engine start, which is initiated by opening the propellant shutoff valves at the engines. Takeoff is then accomplished by simply opening the throttle. The propulsion system is secured after flight by closing all valves in the reverse order.

Conclusion

It was concluded that basic system design simplicity, large component design margins, derated engine design for high reliability and long service life, and an operational philosophy of run-up and checkout before commitment is optimum for the LFV.

On reflection, it appeared that this study had simply rediscovered, for the LFV case, the design and operational philosophy that has been successfully applied to light aircraft for countless numbers of aircraft and flights.

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