A Three-Man Space Escape System

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The two general methods of returning astronauts from a disabled vehicle are space rescue through rendezvous by another vehicle and space escape involving use of an onboard emergency system to abandon the spacecraft and return to earth. The three-man escape system described here will function and permit return to earth from orbits of 100 to 300 naut miles altitude at inclination angles from 30° to 90° (posigrade). The rigid space escape system possesses buddy capability, land or water landing of astronauts by bailout on individual parachute after re-entry, storage capacity of 6 month prior to launch and 1 yr in orbit, independent of spacecraft power and expendables, and minimal weight. It was designed for minimal c.g. travel when carrying 1, 2, or 3 astronauts. The escape craft weighs 809 lb, with a 9.5-ft diam and a length of 6.5 ft. The re-entry ballistic coefficient is 17 lb/ft². The foamed elastomeric heat shield thickness ranges between 1.03 and 1.29 in. The escape vehicle includes self-contained life support and automatic control systems. Two nonconventional subsystems used are: 1) the polyurethane foam structure and 2) the eyeball attitude reference system for deorbit. Feasibility tests have already verified many design features of the space escape vehicle.

Nomenclature

A_x = axial deceleration, g
CM = pitching moment coefficient
ESM = elastomeric shield material
EVA = extravehicular activity
H = altitude, ft
K = stability factor
LSS = life support system; PLSS portable LSS
MOOSE = manned orbital operations safety equipment
PGA = pressure garment assembly
q = total integrated heat flux, Btu/ft^2
$\dot{q}$ = heat flux, Btu/ft^2-sec
Q = dynamic pressure, psf
Res = Reynolds number based on distance along vehicle surface
SEV = space escape vehicle
T_{back} = backface temperature, °F
T_W = wall temperature, °F
V = velocity, fps; V_E = initial entry value
ΔV = deorbit velocity, fps
α = angle of attack, deg
β = pitchdown angle along which ΔV is applied, deg
γ = flight path angle, deg; γ_E = initial entry value
ϕ = angle between solar and orbital planes, deg

Introduction

SPACE escape, i.e., use of an onboard system to effect emergency abandonment of a spacecraft to return to earth, appears, from a number of analyses,[1–3] to offer considerable advantages over space rescue for the near future. (We define space rescue as an emergency mode which calls for vehicles launched from an alerted base to rendezvous with the threatened astronauts, and, after crew transfer, to return to earth.) The General Electric Company has carried on an 11 yr research and development program in space escape and rescue. This program has included re-enterable and non-re-enterable, erectible and nonerectible, and single- and multi-man space escape-rescue systems. A brief summary and comparison of various escape systems has been presented previously,[4] and design and test data on the one-man, erectible system, Manned Orbital Operations Safety Equipment (MOOSE), have been presented.[4–6]

While one-man escape vehicles offer greater flexibility in location and stowage of escape equipment, multi-man vehicles offer buddy-system mutual aid benefits for injured crew members. The erectible systems offer the advantages of minimized stowage volume and minimized spacecraft design penalties incurred for protection of the escape system from fire, meteoroid shrapnel, and other emergency-induced hazards. On the other hand, rigid systems offer the considerable advantage of minimizing in-space operations during escape situations.

This paper summarizes an investigation of a rigid, three-man space escape vehicle (SEV). The aim is to define the design and characteristics of such a system and to minimize its disadvantages, when compared to the one-man erectible system, through the incorporation of the following features developed in past studies: 1) *A structure of rigid polyurethane foam.* Ten years of material testing, design study, laboratory model testing, full-scale vehicle structure fabrication and full-scale vehicle structural testing have shown that rigid polyurethane foam structures meet all the operating loads of this space escape vehicle with adequate design margins. The structure is relatively simple, easy to manufacture, inexpensive, compatible with a wide variety of environmental conditions, highly resistant to damage and catastrophic failure, and lightweight. This foam structure plays a multiple role as seat structure, flotation system, and component mounting structure for no additional weight penalty. 2) *An ablative elastomeric heat shield.* Eight years of development have produced ESM 1030, a self-healing shield, compatible with orbital thermal cycling, which has a low glassification temperature, a high heat of degradation, and low thermal conductivity. 3) *The eyeball orientation system.* This man-in-the-loop attitude control system stabilizes and orients the escape vehicle for

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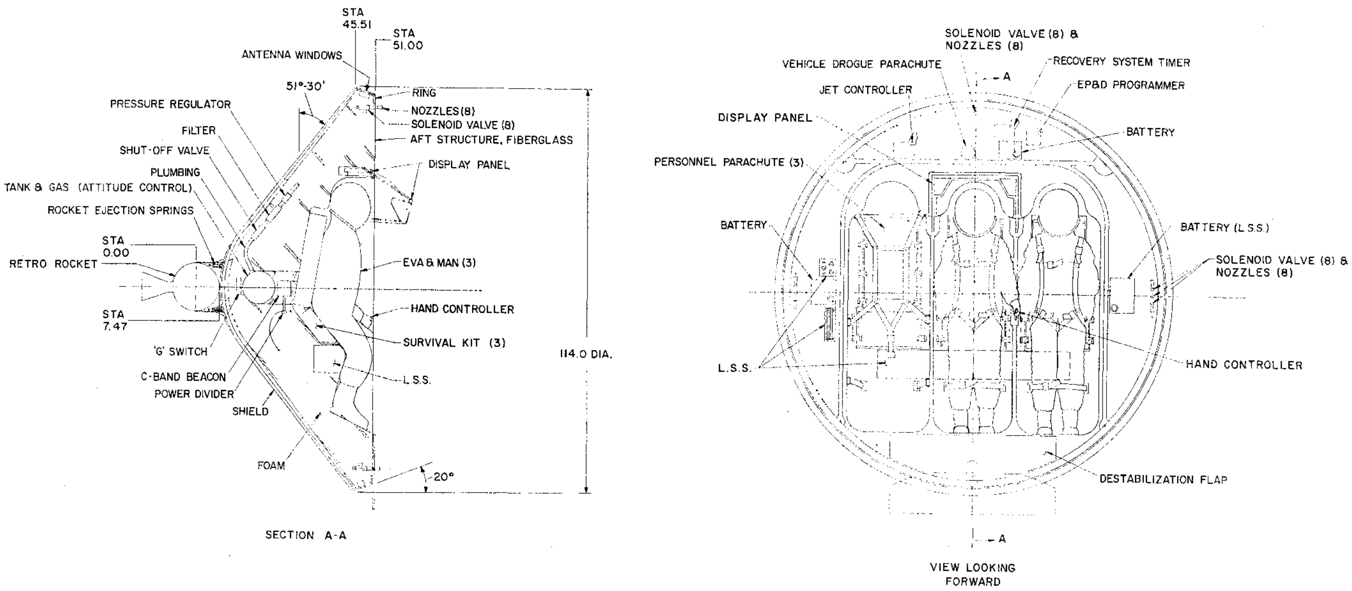


Fig. 1 Space escape vehicle (SEV).

deorbit firing. Utilizing a simple sighting reticle, the pilot operates attitude control jets to stabilize and point the vehicle. While accuracy is not high, it is adequate for safe re-entry.

Deorbit propulsion is to be provided by a rocket engine sized for direct retrograde thrust with allowable misorientation of up to ±25° in pitch, yaw, or both combined. A built-in life support system, already under development, is provided. The astronauts first enter the SEV while utilizing the standard Portable Life Support System (PLSS), and then they plug into the built-in LSS. The 4-hr capability of the built-in system may be extended to 8 hr by retaining the PLSS. Electrical power supply, tracking beacon, survival gear, and recovery system for the proposed design concept are conventional.

The 809-lb vehicle presented in Fig. 1 represents a basic configuration capable of performing the required mission. The 9.5-ft diam and 6.5-ft length will present some location and stowage problems within a parent spacecraft. However, the ruggedness of the foam structure, its ability to insulate the embedded subsystems, and the self-healing characteristic of the heat shield make the vehicle relatively insensitive to storage conditions and to secondary damage from the escape inducing emergency. Such characteristics allow location and stowage of the SEV in positions, and under conditions, not tolerable to standard designs. Many of the basic SEV features have been reduced to hardware or models which have been subjected to key tests, as discussed herein. Weight penalties have been estimated for additional capabilities: two-way communications, impact attenuation for in-vehicle landings, and a post-impact righting system.

System Requirements

The SEV is conceived as a back-up in conjunction with a manned re-entry vehicle. For purposes of this study, the spacecraft has been considered in an orbit of 100–300 naut

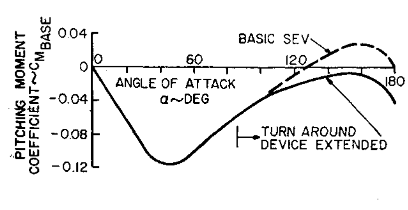


Fig. 2 Hypersonic pitching moment coefficient vs angle of attack.

miles (nominal conditions for minimized exposure to trapped radiation and reasonable orbit degradation due to drag) with an inclination of 30°–90°. While the SEV is nominally a three-man vehicle, it must alternatively provide capability for one or two men with the minimum detrimental effects on its control and its re-entry trajectory. Consistent with reasonable logistics and astronaut tours of duty expected for the early 1970's, the minimum storage capability will be 6 months prior to launch and 1 yr in orbit. Furthermore, during storage and operation the SEV will not draw upon the power or expendables of the spacecraft. A goal of this study has been minimal monitoring of the SEV while in storage.

The SEV may be stored inside, mounted on the outside surface of, or rigidly tethered to the spacecraft. When an in-orbit emergency requires escape, each astronaut suits up, attaches the PLSS, and transfers to the SEV. After check-out of the SEV the crew has the option of switching to the vehicle LSS and ejecting the PLSS, or retaining the PLSS. After separation from the spacecraft and achieving a safe separation distance, the SEV is manually stabilized. It is then put into the cruise mode with inactive attitude control in order to conserve attitude control fuel. The deorbit point is determined by ground observation and previous orbital information of the space station. If the astronauts have elected to retain the PLSS for added orbit stay time, that system must be ejected prior to alignment for deorbit. The SEV is then aligned for deorbit using the eyeball orientation system and the attitude control system. The deorbit rocket is fired at the deorbit point and then jettisoned. After entry and deceleration to subsonic speed the astronauts leave the SEV on individual parachutes. If they land on water they use their individual life rafts. In any case, they activate their recovery beacons and other recovery aids and await recovery using their survival gear until rescued.

Operating Requirements

The GE Mark II forebody shape was selected. This configuration (Fig. 1) is well-proven in both ground and flight tests. Because it has stable trim points at angles of attack of 0° and 180° (Fig. 2), a requirement for a flap-type turnaround device has been specified. Such a device will destabilize the vehicle should it begin rearward re-entry. The flap provides the vehicle with single-point stability. The extended flap will not interfere with crew ingress, and it is small and lightweight.

The requirement for crew bail-out prior to landing requires stable flight throughout the transonic regime. Historically,

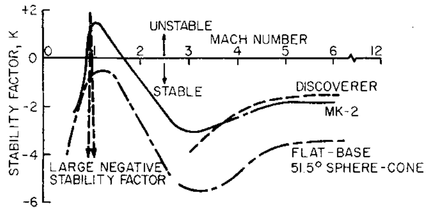


Fig. 3 Computed dynamic stability of blunt sphere cones.

blunt sphere-cone shapes have had transonic dynamic stability limitations (Fig. 3). However, the plotted dynamic stability factor for the selected configuration indicates transonic stability.

Only lightweight, passive thermal control measures should be required to maintain subsystem heating rates and temperatures at acceptable levels when the SEV is not within the thermal control of the spacecraft. When the SEV is on the shadeside of the space station, without thermal control, the temperature varies from $-60°$ to $-220°F$; when on the sunside, the maximum temperature of the heat shield is 237°F. To provide for such environmental extremes, the design concept is configured with passive thermal controls such as insulation, thermal barriers, and thermal control coatings.

The in-orbit heat fluxes on the SEV base surface (crew location) are given in Fig. 4 for the 300-naut-mile orbit inclined 30°. The nominal re-entry trajectory is presented in Fig. 5. Heat-shield heating rates are shown in Fig. 6. To minimize the SEV weight, deorbit should be accomplished using a minimum ΔV. The pitch-down angle β, for minimum ΔV, is nearly zero. To accommodate possible worst case deorbit orientation misalignments of up to 25°, a ΔV of 470 fps is required to assure that the space escape vehicle re-enters from the 300-naut-mile circular orbit. The maximum range dispersion associated with this case is approximately ± 400 naut miles, due primarily to tolerances in ΔV and β.

Figure 7 illustrates the re-entry velocity and path combinations resulting from minimum ΔV deorbit from orbits between 100 and 300 naut miles. In general, small re-entry path angles produce large values of total heating, whereas larger re-entry path angles cause higher deceleration levels and higher heating rates.

Subsystem Design

Propulsion and Power

The deorbit rocket selection was based on system-level requirements of 20,000 lb/sec of impulse, which has been converted to 40-sec burn time and 400-lb thrust, to supply a nominal ΔV of 470 fps. A solid-propellant motor was selected for storage, reliability, and physical size considerations. The Thiokol TE-444 rocket motor was selected from several motors available. For the SEV concept, the motor will be modified by changing the fuel from PDAA to polyurethane to provide the longer burning time. To satisfy the in-orbit storage requirements the unit will be sealed. The modified TE-444

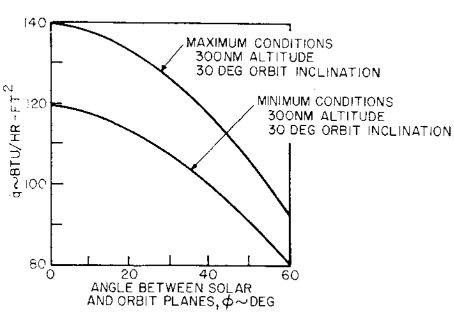


Fig. 4 Aft surface heating rate, in-orbit.

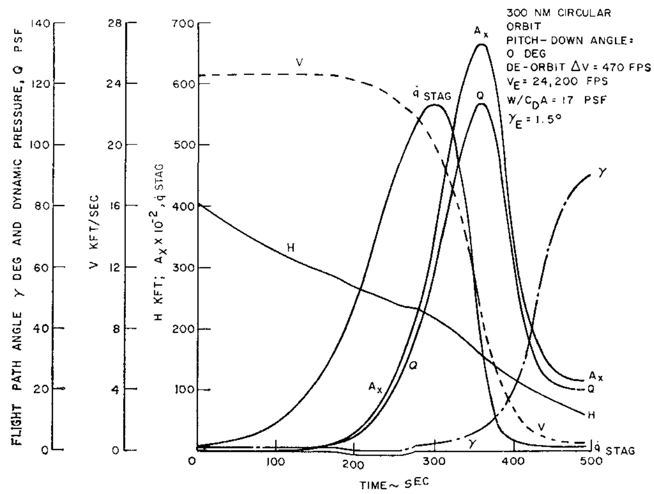


Fig. 5 Re-entry performance of proposed SEV design concept.

motor is a 13-in.-diam sphere with a 22.8-in.-long nozzle. Postburn jettison is accomplished by the actuation of four hot-wire tension links. A separation velocity of 1 ft/sec is achieved through the ejection of a compressed helical spring.

The electrical system comprises two AgO-Zn batteries with heaters, a programer, a control panel, and electrical harness. The assembly is designed for a 4-hr power life. Percussion-activated chemical heaters provide initial heating to the separately storaged battery electrolyte. Once the batteries have been heated and activated, bootstrap electrical heaters will maintain the required temperature. Application of power to the subsystems is carried out either by the programer or through the control panel. In the manual mode, signal sequencing is interlocked to prevent inadvertent premature signals. The control panel includes a mechanical timer.

Life Support Subsystem (LSS)

The LSS will provide adequate environment for three crew men suited in EVA garments. As a minimum, it provides

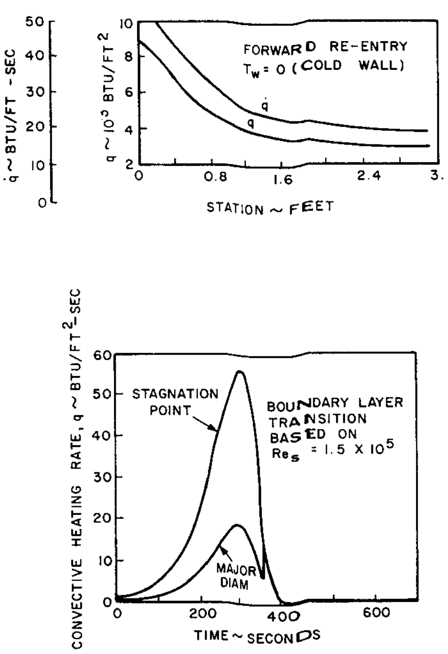


Fig. 6 Convective heating rate $\dot{q}$ and total integrated heat flux q vs vehicle station, and $\dot{q}$ vs time at the stagnation point and the major diameter.

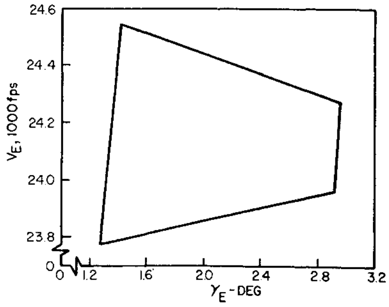


Fig. 7 V_E vs ν_E ($100 \leq H_p \leq 300$ naut miles; $100 \leq H_a \leq 300$ naut miles; $\beta = 0$).

oxygen and cooling based on 600 Btu/hr-man for 4 hr. It is estimated that minimum escape operation takes 2.6 hr of LSS capability; a 50% contingency is added for possible decision on an alternate recovery zone.

In the eartly 1970's, the pressure garment assembly (PGA) will be of the liquid cooling type; therefore, the LSS must be capable of supporting it. Preliminary investigation has shown that the selected system can be constructed with the same components as those used in the PLSS. The system can furnish oxygen at 7 lb/hr and water at 4 lb/hr. The O_2 is provided at 5.0 psi, rather than 3.5 psi, to avoid denitrogenation problems. The expendables in the LSS consist of 3.12 lb of O_2, 13.0 lb of water for the reservoir, and 2.0 lb of water for the transport loop. The LSS, as previously described, will weigh approximately 118 lb and have a volume of 5300 in.^{3}

As an alternative, the possible use of three PLSS's as the LSS was investigated. Since a larger SEV would have resulted from the three PLSS's, which require a volume of 12,344 in.^{3} and weigh 210 lb, the built-in LSS was selected. One disadvantage of the integral three-man LSS is that it would require the crew to make systems change (PLSS to LSS) while in a vacuum. However, it appears that by 1970, the reliability of the umbilical connectors (and crew confidence in them) will be established. The three-man LSS offers the crew the option of extending SEV in-orbit time through retention and use of the PLSS before switching to the onboard system.

Recovery Subsystem and Survival Kit

A timer initiated by a re-entry g sensor will initiate drogue parachute deployment at 60,000 ft. A 7-ft-diam drogue parachute will be required for stabilizing the vehicle, which attains a terminal velocity of 171 ft/sec. Three personnel parachutes are provided for bail-out at 20000 ft. This drogue-chute-plus-bail-out concept is aimed at personnel survival of land or water landings. Alternative approaches for total manned vehicle survival are being evaluated.

Postlanding crew support consists of three individual NASA Apollo Block I Survival Kits, each stored in the crew couch. The Apollo Kit has been qualified during astronaut training. It includes a one-man life raft, drinking water, survival light, desalting kit, radio beacon, sunglasses, machete and sheath, and a life vest.

Tracking Subsystem

Onboard tracking equipment for recovery force targeting will be compatible with existing ground equipment. An onboard C-band beacon was chosen for use with FPS-16 ground-based radars, which are widely used on existing test ranges and tracking networks. The antenna consists of a circumferential slot and a longitudinal slot located 180° apart on the vehicle to provide an adequate ground radiation pattern.

Structure Subsystem

Basic members of the SEV structure are an aluminum ring at the periphery of the cone base, a molded fiberglass skin, and a rigid polyurethane foam internal structure. The aluminum ring provides a surface for ground handling, attachment to the space station, and attachment points for the parachute and the deorbit rocket. The molded fiberglass skin serves as attachment surface for the crew and a number of the subsystems. The foam functions as: 1) *A good thermal insulator.* 2) *An ablator in case of heat shield penetration.* Several experimental flight test vehicles of other General Electric programs have deliberately experienced such situations and have performed satisfactorily due to the foam inner structure. 3) *A shock mitigator.* While, in the present concept, the foam structure is not designed or required to provide shock mitigation, growth versions requiring man-in-vehicle landings may utilize such characteristics through foam crushing in the presence of point loads. 4) *An integral vehicle of near-uniform density* (as opposed to a metal structure that must use internal ribs and struts). The uniform density permits distribution of loads throughout the structure and uniform support of the crew. The structural loads encountered during an escape mission are well below the 20 psi strength of 2.0-lb/ft^3 polyurethane foam.

Selection of the specific foam is based on foam producibility rather than input loads. Today, polyurethane foam is a well-established industrial material with known properties. A 2-lb/ft^3 foam (Table 1) was selected because foams with lower density have much poorer structural properties.

The design operating temperature during re-entry is less than the 300°F capability for a polyurethane structure. The structure will experience static and dynamic loads during retro-rocket firing, re-entry, parachute opening, and water impact. Retro-rocket thrust load is transmitted into the vehicle through a support pad on the spherical nose of the

Table 1 Properties of representative commercial polyurethane foams

Foam	Density, lb/ft^3	Compressive strength,[a] psi: Room temp.	Compressive strength,[a] psi: After 24 hr at 300°F	Relative efficiency[b]: Room temp.	Relative efficiency[b]: 24 hr at 300°F	Weight loss after thermal cycle, %
Callery 207	2.3	22.7	17.0	9.9	7.5	5.1
Flexible products 200G20	2.7	32.5	28.6	12.0	10.6	4.0
General Latex RVR-122-23	1.7	14.2	11.0	8.4	6.5	15.0
Chase NAFIL-CP2	2.0	18.3	20.6	9.1	10.3	14.1
Isocyanate products RC-3A/B	3.1	31.9	40.6	10.3	13.1	2.6
Pelron 97152 A/B	2.7	23.6	24.0	8.7	8.9	20.5

[a] Measure perpendicular to rise of foam; provides minimum value.
[b] Strength/density ratio (lb/in^2:lb/ft^3).

vehicle heat shield. The rocket thrust of 500 lb causes a peak compression in the shield of 0.82 psi.

The trajectory parameters, for 200-naut-mile and 300-naut-mile circular orbit, at the time of peak acceleration and maximum heating rate for each orbit are shown in Table 2. Due to a large static stability margin and aerodynamic damping, the angle of attack during peak loads is zero. The maximum stress in the vehicle is 1.25 psi for a deorbit from 200 naut miles at the maximum g point in the trajectory. The maximum surface pressure of 0.148 psi on the forebody occurs at the same point. The strength-temperature performance of the foam is given in Fig. 8, and the bond line temperature history for the worst case, a 300-naut-mile deorbit, is given in Fig. 8b. The strength of the polyurethane foam is 19 psi at the maximum bond line temperature of 200°F. This is more than adequate to sustain the maximum stress of 1.25 psi or the surface pressure on the forebody of 0.148 psi. The cross-sectional area variation in the vehicle is shown in Fig. 8c.

The primary foam structure must also insulate the astronauts from aerodynamic heating. The maximum temperature rise during re-entry across the minimum foam thickness of 2 in. is no more than 20°F. The parachute deployment load is a shock load of 6 g applied at the four attachment points on the aluminum mounting ring.

Attitude Reference Subsystem

The attitude control system must stabilize the vehicle and maintain alignment attitude in pitch, roll, and yaw prior to, and during, retrofire. For simplicity and reliability, automation has been minimized, requiring astronaut capability to a level demonstrated in NASA's Mercury and Gemini programs. After separation of the SEV from the space station, stabilization can be achieved by the pilot using earth features, horizon, and stars as motion references. Prior to retrofire, the pilot will align the vehicle in pitch and roll using the earth's limb on the dayside and either the limb or nightglow layer on the nightside by means of a simple nonoptical gunsight-type display.[7,8] Data from the Mercury program have demonstrated that with a similar display, pitch/roll alignment very probably can be maintained to within $\pm 5°$ prior to retrofire.[9] Once alignment is established in pitch and roll, the astronaut will align the vehicle in yaw while tracking earth surface features. No correction need be made for earth rotation since drift could not exceed a maximum of 0.25°/min (polar orbit). With no moon on the nightside, the position of stars known to lie either along the orbital plane or at a specified number of degrees from the orbital plane can be used to provide both present heading and rate information.[10] Mercury program studies indicate that yaw alignment can probably be maintained manually within $\pm 10°$ on the day or nightside prior to retro.[10] When stable alignment is completed in all axes, the pilot can initiate retrofire.

The yaw alignments described were selected as the most feasible, simplest, lightest, and smallest compared to other methods (i.e., sighting on the disabled spacecraft, release of balloon, ion detection, gravity gradient, magnetometer, and gyrocompassing).

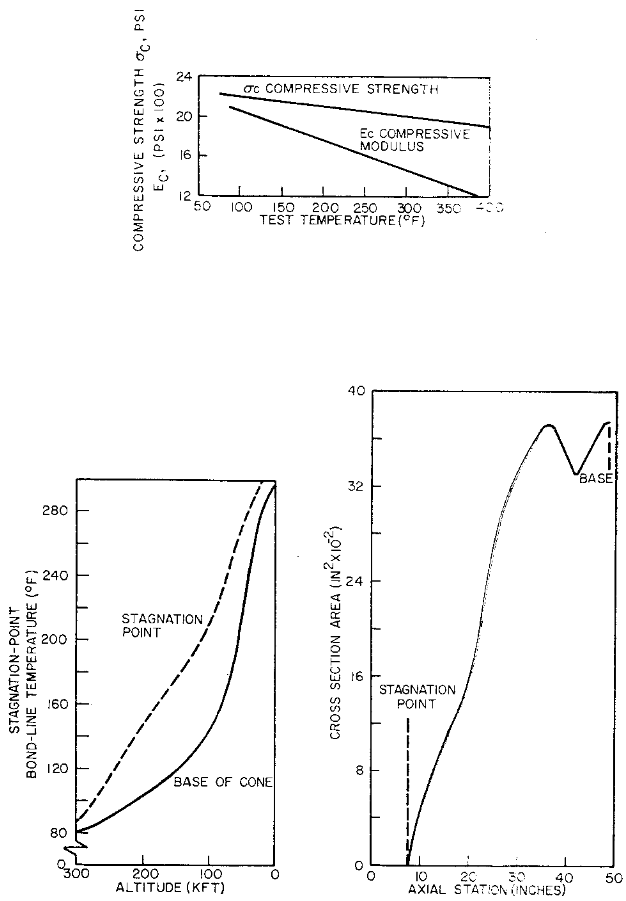
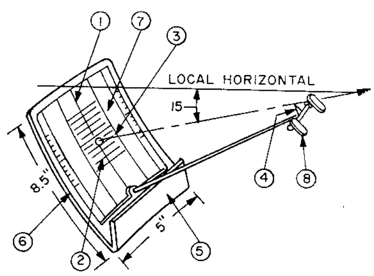


Fig. 8 Design data and required cross-sectional area of polyurethane foam structure (density, 2 lb/ft^3).

Figure 9 shows the gunsight attitude display, designed specifically for circular orbit use where the velocity vector lies along the local horizontal and altitude is known and constant. It is also suitable for elliptical orbit if the proper pitch/roll display reference mark is provided or can be estimated. An analysis was made to determine design values of thrust, total impulse, angular acceleration and system weight as functions of c.g. offset for several attitude-control-system configura-



DESIGN FEATURES

1. GRID WORK OF PLASTIC RODS. RODS APPEAR BLACK WHEN VIEWED AGAINST THE DAYSIDE LIMB
2. PITCH,ROLL REFERENCE MARKS.
3. YAW REFERENCE MARKS.
4. PIP TO ENSURE EYE-RETICLE ALIGNMENT.
5. PLANE MIRROR TO REFLECT SURFACE MOTION AT THE NADIR.
6. LUMINAIRE PROVIDES RED ILLUMINATION
7. RETICLE SHAPED AS PORTION OF A SPHERE WITH EYE AS ITS CENTER. EQUALIZES EYE TO RETICLE DISTANCE AT ALL POINTS ON THE RETICLE. YAW REFERENCE LINES THEN APPEAR STRAIGHT.
8. VISOR CUSHION.

Fig. 9 Gunsight attitude display.

Table 2 Re-entry vehicle loads

Case	A[a]	B[b]	C[a]	D[b]
Orbit altitude, naut miles	300	300	200	200
ΔV, fps	470	470	470	470
Altitude, ft	158,000	215,000	163,000	215,000
Velocity, fps	9,490	19,690	11,010	20,840
Mach no.	8.8	19.5	10.2	20.6
P_L (max), psi	0.135	0.068	0.148	0.077
Axial acceleration, g	6.64	3.6	7.5	4.1
Stress (max), psi	1.18	0.62	1.25	0.70

[a] Maximum deceleration.
[b] Peak heating; P_L (max)—maximum local pressure on forebody of vehicle.

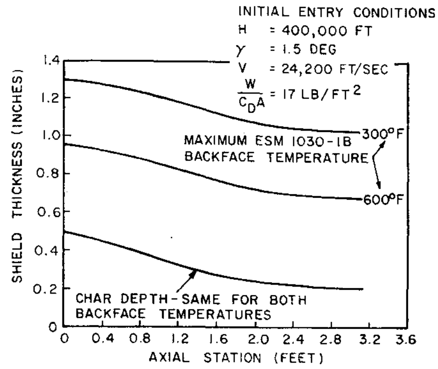


Fig. 10 Axial distribution of shield thickness requirements.

tions. A system was selected which employs pitch and yaw nozzles parallel to, and roll nozzles perpendicular to, the roll axis of the vehicle with Freon 14 as the working fluid.

In synthesizing this system the following operational sequence was considered: 1) *SEV separation.* The magnitude of the separation rates is 5°/sec about each axis. This rate will be in effect until removed by the controls and will result in a time-increasing variation from initial separation attitude. 2) *Remove separation errors.* 3) *Align SRV.* If immediate alignment to retrofire is desired, the controls can be operated as soon as the SEV clears the spacecraft. Control torques will be low compared to retrofire attitude control. For a manual-proportional control system, the minimum torques will be about 5–10% of maximum. By using additional time, it is possible to allow the SEV to rotate until one or more axes are nearly aligned and then operate the controls. 4) *Hold to retrofire.* Although the external disturbance torques will be operating continuously (aerodrag and gravity gradient) their effects will be felt only during a long hold period. Control will be used in a bang-bang method whenever the SEV rotates sufficiently out of alignment to be sensed by the astronaut. Rates will be small, resulting in primarily position error control. 5) *Initiate retrofire.* 6) *Control during retrofire.* When retrofire is initiated, disturbance torques will be applied to the SEV. Position errors will increase less rapidly than rate errors. Once the astronaut senses the errors he will operate the controls and the inevitable sensing and operating lag will cause rate cross coupling (primarily into roll) which will be removed by the controls. 7) *Retrofire ceases.* The lag in pilot reaction will cause some overcontrolling. The astronaut will reduce his control torque accordingly to reorient the SEV.

Analysis based on SEV parameters, space effects, and operational considerations showed control system requirements of fuel and tankage totalling 80 lb. The major factors

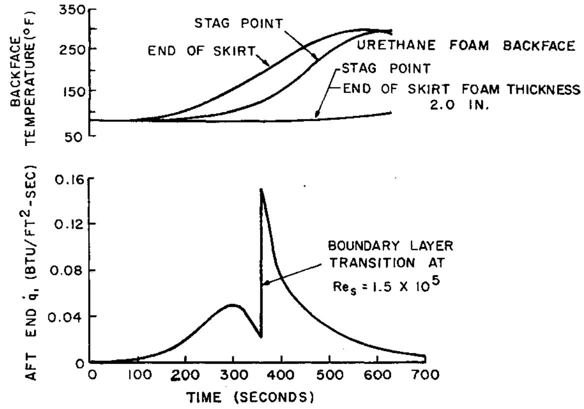


Fig. 11 Aft end convective heat flux history.

Table 3 Properties of candidate heat shield materials

Properties	ESM 1030-1B	ESM 1004X
Density, lb/ft^3	18.0 ± 2.0	15.0 ± 2.0
Specific heat, Btu/lb-°F		
0°F	0.30	0.26
100°F	0.35	0.31
300°F	0.44	0.35
500°F	0.49	0.38
Dynamic enthalpy	Large endotherm (air, 0.01 atm, 750°F)	Exotherm
Thermal conductivity $\times 10^{-5}$ Btu/ft-sec-°F		
1 atm air 0°F	1.12	1.00
100°F	1.24	1.14
300°F	1.48	1.46
Vacuum 0°F	0.79	...
100°F	0.84	...
300°F	0.95	...
TGA–5% weight loss		
In air	700°F	650°F
Onset of degradation	700–800°F	750–800°F
50% weight loss	975°F	1150°F
Residual weight	15%	35%
Weight loss in vacuum, 10^{-5} mmHg	0.62%	0.17%
Room temperature, 5 days		
Ablation time to T_{back} = 500°F, sec		
7 Btu/ft^2-sec—1700 Btu/lbm	276, 372 sec	162 sec
12 Btu/ft^2-sec—5800 Btu/lbm	405, 405 sec	425, 530 sec
24 Btu/ft^2-sec—5700 Btu/lbm	134 sec	...
Optical properties		
Total solar absorptance	0.376	0.755
Total hemispherical emittance	0.85	0.85
Electrical properties		
Room-temperature dielectric constant	1.47–10.011	1.70–10.016
Loss tangent	0.007	0.009
Attenuation coefficient, cm^{-1}	0.00889	0.01229
Maximum change in signal—during ablation—transmission	−1.4 db	−0.4 db
Maximum change in signal—during ablation—reflection	−3.5 db	

	Tens. str., psi	Elong., %	Mod., psi	Tens. str., psi	Elong., %	Mod., psi
Tensile properties						
−150°F	199	(2)	22.300	37.4	41	95
−100°F	125	(2)	...	...	...	...
− 50°F	33.6	56	60	12.6	22	58
75°F	20.5	46	45	9.4	22	43
300°F	10.1	35	29	7.2	17	43
Coefficient of thermal experiment, in./in./°F						
+75 to −75°F		1.06×10^{-4}			...	
+75 to 150°F		...			5.5×10^{-5}	
−200 to −250°F		3.3×10^{-5}			...	

in this large weight are the thrust and fuel requirements to control the 500-lb retro-thrust for 50 sec on a vehicle with a 1.0-in. c.g. offset. This c.g. offset can result from a 10 and a 90% crewman in the outboard couches.

To meet the foregoing requirements, a manual-proportional-plus-dead-band control system was selected. The pilot will use the gunsight display and hand controller to yield proportional torque through servo valves. The pilot reduces vehicle rates manually, acquires retrofire attitude, and maintains three-axis control during retrofire. The control stick employs an electrical potentiometer for positive and negative directions about three mutually perpendicular axes. The displacement of the control stick from null is proportional to the output of the thrust level.

Table 4 Weight statement for SEV design concept

Item	Description	Weight, lb
1	Structure	165
2	Heat shield	74
3	Retro rocket	100
4	Attitude control	120
5	Electrical and display	80
6	Life support	85
7	Recovery/tracking	110
8	Survival kit	75
	Total	809

Table 5 Key tests which support design and operations of three-man space life raft

Test	Purpose	Status	Key results	Planned follow-on testing
Heat shield arc tests	Determine heat shield response to re-entry heating	ESM 1030-1B selected; preliminary data acquired	Backface temperature and ablation rates determined; other elastomers tested and comparison data measured	Additional materials and composites to be tested; statistical sample tests to be run
Heat shield mechanical and physical properties tests	Determine heat shield response to handling, storage, air loads, etc.	Preliminary data acquired	ESM 1030-1B acceptable; also other elastomers	Statistical sample tests to be run
Foam structure properties tests	Establish proper foam formulation and determine its properties	GE custom-formulated polyurethane foam selected	Evaluated and key properties determined for several environments	Evaluation of commercially available foams; statistical sample tests of best foam
Large foam structure fabrication	Determine feasibility and problems of fabricating full-scale SEV; determine shape conformance of large volumes of foam	Several full-scale shapes fabricated with various shaped payload cavities	Shape conformance—good, foam exotherm—acceptable, properties uniformity—fair, density uniformity—good, foam distribution—fair	Evaluation of several processes exercise of best process
Neutral buoyancy simulation of zero g ingress operations	Determine feasibility of zero g ingress to a SEV which is not attached to a large mass vehicle; evaluate handholds and restraints placement to facilitate ingress; establish degree of effort required for ingress	Preliminary phase complete; one set of handholds and restraints evaluated; report prepared	Selected handhold and restraint system is poor; ingress capability established even with poor handholds and restraints; effort for ingress appears not to be excessive	Additional handholds and restraints to be evaluated; simulation procedures and equipments to be modified and tested; ingress procedure to be modified and tested; multiple operators to be tested
Eyeball orientation system simulation	Determine capability of operator to stabilize and orient simulated vehicle for deorbit	Test plan complete; 50% equipment available	None yet	To be run when equipment is available

Heat Shield Requirements and Design

The heat shield must survive both the space storage environment (vacuum, low and high temperatures, UV and ionizing radiation, and micrometeoroids) and the re-entry conditions shown in Fig. 6. The shallow path angle of 1.5° results in a boundary-layer transition from laminar to turbulent flow near the end of the heating period. Hot gas radiation from the shock layer is negligible.

Several approaches were examined for the heat shield material. Whereas, reradiative materials with the required insulation impose severe weight penalties, and high-density charring ablators or low-temperature sublimers require higher heating rates to operate efficiently, the chosen low-density charring ablators possess low thermal conductivity and good ablation performance. A number of low-density charring ablators were considered; these include GE's ESM 1001P, 1004BP, 1004X, 1030-1B, and NASA602 G-H/C-S. The density of the ESM 1030-1B and 1004X are considerably lower than the other materials considered. These low-density materials possess the low thermal conductivity desired.

These two low-density charring ablators were then investigated (see Table 3). ESM 1004X is a methyl phenyl silicone reinforced with aluminum silicate fibers chemically foamed under reduced pressure. ESM 1030-1B is an epoxy-silicone reinforced with aluminum silicate fibers and foamed at atmospheric pressure. The maximum predicted surface temperature of the design shield is 3100°R, which is below the surface recession temperature (3400°R) of the two ESM's. ESM 1030-1B was chosen for its performance in both space storage and re-entry environments and ease of manufacture.

The heat shield thickness and char depth with axial location along the vehicle, for a maximum backface temperature of 300°F, is shown in Fig. 10. The thickness varies from 1.29 in. at the stagnation point to 1.03 in. at the end of the skirt. The total weight is 74.2 lb. Backface temperature histories for the heat shield are shown in Fig. 11a. The maximum backface temperature does not reach 300°F until 650 sec after re-entry and at an altitude of 12,000 ft. The maximum temperature in the polyurethane foam 2 in. inside the backface of the shield does not reach 100°F until impact.

Afterbody heating has been analyzed for both the normal attitude (shield forward) and backward re-entry mode. As shown in Fig. 11b, the afterbody heating for normal re-entry is small. The afterbody heating in the backward re-entry mode is significant with a peak $\dot{q}$ of 31 Btu/ft^2-sec at 210K ft altitude, and the vehicle is provided with a passive destabilizing flap to prevent backward re-entry.

Design Concept

Based on the foregoing design concept requirements, the SEV design concept (Fig. 1) is considered to be the minimum vehicle for such a mission. The vehicle weight statement is presented in Table 4. Significant interface and integration features of several key subsystems are discussed in the following paragraphs.

The personnel parachutes and survival kits are located beneath each crewman and are part of the couch profile. The units are secured to the foam structure by pins, in a fashion similar to inflatable evacuation slide attachments in our present-day commercial aircraft, and are released by activation of the cable release mechanism lever. The crew are secured to the vehicle by leg and parachute restraints. Although a standard parachute and leg restraint are shown, present plans to ease ingress and egress call for a modified NASA Apollo Five Point Release Restraint System and boot retention as in the Gemini or Apollo vehicles.

The vehicle subsystems are packaged within the foam structure so as to have all controls accessible to the crew and still provide favorable mass properties for attitude control, deorbit, and re-entry sequences. Access is provided to all subsystems or components of subsystems that will require servicing or periodic inspection (such as pressure indicators, fill ports, etc.). The panel display when rotated to the upright position will have provisions for traversing to the right so that vehicle flight control can be accomplished by the crewman in either the center or right-hand couch. This provides the capability for one, two, or three-man space escape. The Eyeball Orientation System is located within the panel display which is stored above the head of the center crewman. When the display is rotated to the upright, operating position (reference section A-A) proper eyeball/reticle orientation is established.

To achieve maximum attitude control system effectiveness, favorable mass properties, and minimum weight, the solenoid valves and nozzles are located and secured to the aft face of the aluminum ring, 90° apart. Through radial plumbing, they are attached to the remaining components located forward of the center crewman. The entire system is capsulated in foam with removable service plugs at critical components. Activation of the solenoid valves is accomplished through the hand controller on the arm rest located between the center and right hand crewman.

To maintain the required mass properties, and access to replaceable or serviceable components, the electrical components are located in the aft face of the vehicle. The LSS battery is located to the right of the right-hand crewman, the battery for the vehicle control is located to the left of the left-hand crewman, and the beacon battery is located above the center crewman. The *C*-band beacon is located below the center crewman's parachute, and the two antenna windows are located within the aluminum ring 180° apart. The drogue chute is located above the head of the center crewman and is secured by four risers that are housed in foam channels with tearaway covers. The ends of the risers are secured to the aluminum ring.

Additional capabilities such as the following can be incorporated in this design concept at weight penalties shown in parentheses: 1) an aft cover and hatch (175 lb) to provide a shirt-sleeve environment with the additional capability of access to the vehicle through a spacecraft airlock or hatch without a pressure suit, 2) a larger parachute, 57 ft in diam (53 lb) to provide the crew with the option of remaining within the vehicle for water landings without incurring impact loads greater than 20 *g*, and 3) a vehicle righting system (30 lb) to right a capsized vehicle and maintain it in the righted position.

Test Programs

Table 5 presents a summary of key testing which has been accomplished in support of the subject SEV concept. Much of the testing has been of a preliminary nature, sufficient to establish feasibility but not a final design. In addition to the tabulated tests, several tests have been completed which will bear strongly on alternative approaches; e.g., should further study establish a requirement for the astronauts to rise the SEV down to the surface, two significant test programs are of interest: 1) small-scale and full-scale tests of impact decelerations for various terminal velocities and 2) small-scale tests of a simple system to right a capsized space escape vehicle.

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