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ROMBUS - AN INTEGRATED SYSTEMS CONCEPT FOR A REUSABLE ORBITAL MODULE (BOOSTER & UTILITY SHUTTLE)

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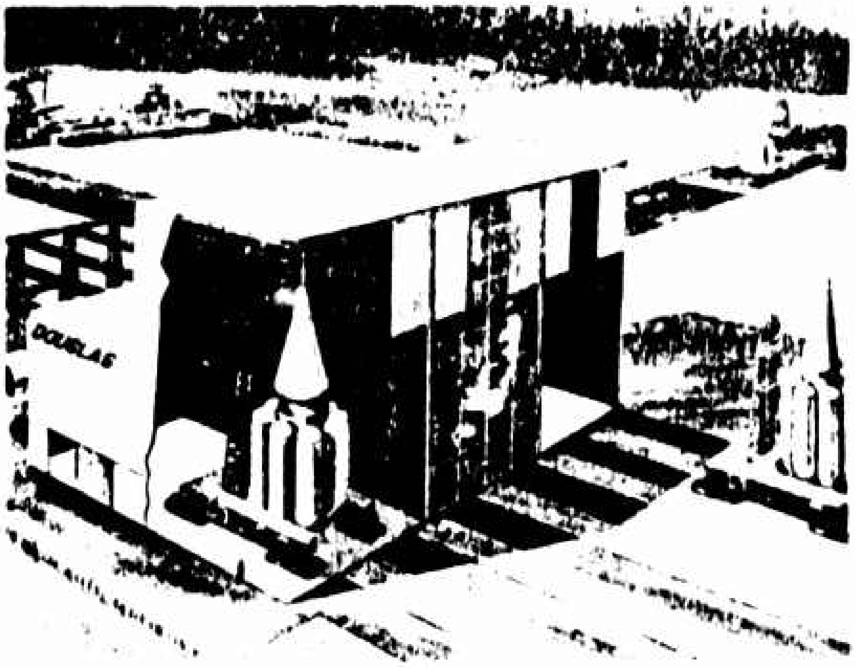
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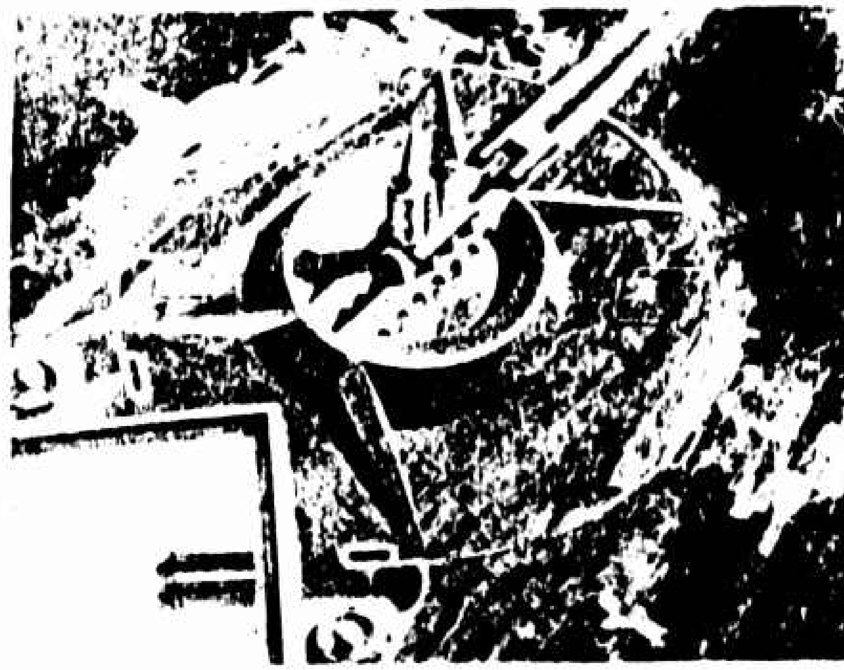

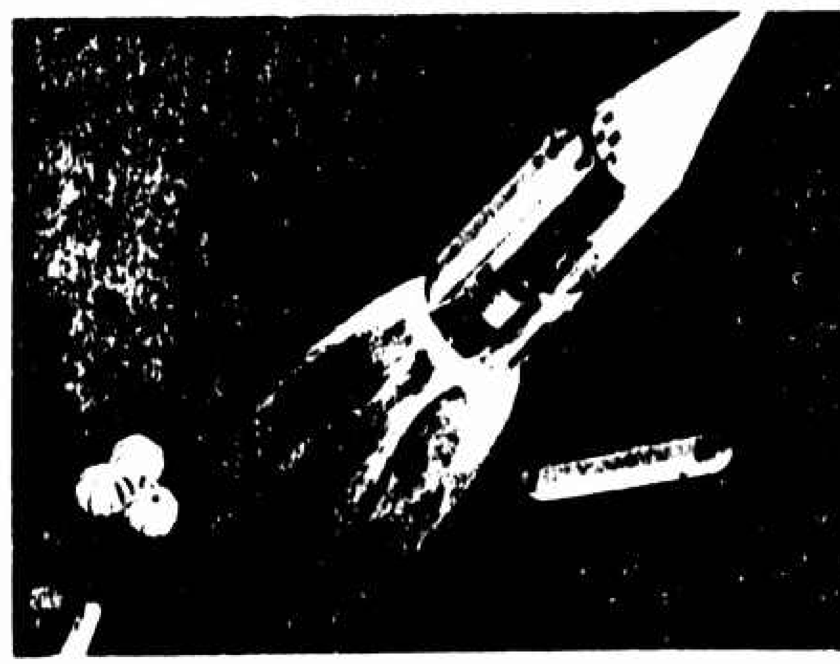

ROMBUS - AN INTEGRATED SYSTEMS CONCEPT FOR A REUSABLE ORBITAL MODULE (BOOSTER AND UTILITY SHUTTLE)

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ABSTRACT

The objective of this paper is to define a conceptual design for a reusable surface-to-orbit manned cargo carrier which can deliver a payload of many hundreds of tons to a 175 nautical mile orbital altitude. Recovery is accomplished by deployment of a subsonic drag device and terminal retro-thrust. Touch down on land is implemented by settling on four extensible legs, in a manner similar to the basic method which is being developed for landing on the moon.

The vehicle is propelled by a conjectural altitude-compensating plug-nozzle engine which can be throttled for the landing maneuver. The engine operates at a high chamber pressure with a mixture ratio (LO_2/LH_2) of 7/1. The 80-foot diameter base of the vehicle provides a vacuum expansion ratio of 200/1. The hypothetical main engine serves a multiple function. It is also used for orbit injection, rejection, landing retro-thrust, and as a regeneratively-cooled re-entry body. A hypergolic secondary propulsion system provides attitude control.

Through disposal, (but recovery by parachute), of the eight hydrogen tank segments immediately after depletion, the <u>effective</u> propellant mass fraction is increased. Hence, performance is improved, and the magnitude of the ground handling problems after recovery are significantly diminished due to the reduced size and weight. Disposal of the parallel-arranged LH_2 tanks is accomplished with greater plumbing simplicity and reliability than the complex cross-pumping required for many parallel-staged theoretical vehicles proposed in the past. An investigation into the direct operating costs, in terms of dollars per pound of payload, is presented.

The paper tentatively concludes that the assumed high-chamber pressure LO_2/LH_2 engines of altitude-compensating design (with high expansion ratios) would present a dramatic improvement over conventional engines, in achieving the desired objectives of minimum vehicle growth factors, and minimum program costs. It appears economically justified that the next generation of large launch vehicles should be: (1) reusable, (2) single-stage-to-orbit, and (3) developed to incorporate maximum mission flexibility.

INTRODUCTION

The hypothetical vehicle described in this paper was evolved to serve as a representative example of a generic type of utility "shuttle" which could best satisfy the objectives of a four-fold improvement over current vehicles in (a) payload capability, and (b) direct operating costs. Toward this end, the following assumptions were adopted during the course of the conceptual design: (1) Primary missions - lunar base support and planetary exploration, (2) Assumed funds available - 1 billion \$/yr for 10 years, (3) Desired objective - direct operating costs (DOC) of \$25/lb payload, or less, (4) Vehicle features - reusability highly desirable, (5) Vertical assembly and transport on "crawler," (6) Eastward launch from A.M.R.

Throughout the major portion of the vehicle conception phase, the payload was assumed to be of LH_2 density (4.43 lb/ft^3). A lower-density payload representative of an interplanetary spacecraft was also investigated to determine the relative complexity of the vehicle control problems.

In order to determine rough order of magnitude (ROM) cost estimates for the vehicle, a number of assumptions were made regarding some possible engine configurations. For the sake of illustration, it appears that either of two potential engine concepts can satisfy the combined (boost plus re-entry) requirements for this vehicle concept. One of these two basic approaches to combustion chamber configuration appears necessary if the assumed objectives of the hypothetical vehicle are to be met. As a preliminary estimate, the lift-off thrust was assumed at 18 million pounds, in order to evolve a potential mode of operation for the vehicle concept.

The segmented-annular combustion chamber (Figures 1 and 3) was adopted for postulating a representative sequence during boost and recovery, although the can-annular combustion chamber of Figure 2 appears as attractive for a conjectural engine. The combustion chamber was speculated to be comprised of 36 segments, each producing one-half million pounds of thrust at sea level. The conceptual vehicle can then successfully perform its intended mission with six (to eight) engine segments inoperative during boost. The orbital injection, ejection, and terminal retro-thrust maneuvers can be accomplished by ignition of any four of the 36 engine segments. Two turbo-pumps were assumed to provide the required propellant flow rate, with a third turbo-pump carried on board as a fully-redundant unit.

The plug-nozzle type of engine, shown in Figure 3, modified with a spherical nose radius, is a requirement for this concept. This shape alone is adaptable for both the altitude-compensating engine and re-entry body requirements. To assure a stable condition during recovery, the re-entry body should have its center of gravity located as far forward as possible (aft direction during ascent). This requirement would suggest that the booster should re-enter aft-end-first, so that the engines, which have the largest concentration of mass, would be located toward the forward end of the re-entry body. A conventional bell-nozzle engine could not survive the aerodynamic heating, as the edge of the nozzle would be heated to prohibitive temperatures.

The plug-nozzle engine is the most effective solution to this problem. Moreover, regenerative cooling during re-entry is much more practical for a truly reusable vehicle, since ablatant, which would have to be replaced after each flight, would also char and foul the annular engine throat. This fouling could result in a malfunction when the engine is later expected to restart for terminal retro-thrust prior to touch-down. The large base diameter serves a dual purpose, since it also increases the drag during re-entry. Because the isentropic plug-nozzle engine must be regeneratively cooled during operation, it appears feasible to use this same method for cooling during re-entry. Since the engine is inoperative during the maximum-heating regime of re-entry, a gas generator could be used to run the on-board turbo-pump, which pressure-feeds the LH_2 coolant. After cooling the centerbody, the hydrogen is fed through the injector and discharged overboard through the annular throat, helping to cool the combustion chamber in the process. It appears extremely likely that this discharged gaseous hydrogen will burn under aerodynamic heating within the atmosphere; however, the heat flux which would be added to the plug nozzle, should this burning occur, appears tolerable. The effect of a flame on the lower portion of the centerbody structure should be investigated, however.

OPERATIONAL SEQUENCE

A possible location for the four ROMBUS launch pads is shown in Figure 4. The nearest of these pads would be located approximately 11,400 feet (1.87 n.mi.) from the nearest existing facility. All ROMBUS launch complexes would be this same distance apart, so that a launch from any pad would produce an overpressure of not more than 0.5 psi at any adjacent pad. Also shown is a proposed location for the vertical assembly building. The proposed landing site location is shown in Figure 13. No exposed personnel would be permitted closer than 900 feet from the ROMBUS launch pad, even with ear protection. The figure indicates that under adverse weather conditions, when low clouds and winds amplify the sound transmission, ear muffs may be required as far away as 89,000 feet (14.6 n.mi.) from the ROMBUS launch pad. This tentative conclusion is based on very limited empirical data which was extrapolated from recent Saturn launches for the increased power output of the 18-million-pound thrust level of ROMBUS. It is presented only to indicate the strong influence of weather on sound transmission.

The individual major subassemblies of the ROMBUS vehicle would be fabricated at a remote site and transported by barge to the vertical assembly building (Figure 5). This vehicle has an inherent advantage over conventional boosters, since it does not require any welding operations within the vertical assembly building. That is, each major subassembly can be completely fabricated elsewhere, and merely bolted together during final assembly at the launch site. A ROOST-type vehicle (Reference 1 and 2) would necessitate that large segments of the cylindrical propellant tank be transported by barge, and later welded to each other in the vertical assembly building, thereby consuming much valuable time at the launch site for the assembly procedures. Figure 6 illustrates how the vehicle would be transported from the assembly building to the arming tower and launch pad.

Final checkout and fueling of the vehicle would be performed after it has been transferred from the crawler to the launch pad supports (Figure 7). The crawler is detached and moved to a safe distance in order to protect it from the destructive blast effects. The unconventional design of the exhaust blast deflector is necessary to minimize the acoustic energy reflected to the structure during lift-off. A more detailed description of this deflector is covered in Figure 17.

The first set of four tanks are emptied in 130 seconds and separated after maximum dynamic pressures are encountered. When a tank is stripped of the engine, engine thrust structure, and bracket-mounted "black boxes," it is less likely to be damaged by water impact. No bulky masses are on board to tear loose when the tank strikes the ocean. The external dynamic pressure ($1/2\ \rho v^2$) casued by the water impact will not exceed the tank internal pressure; consequently, the tank will not collapse. A recovery system weight estimated at 1460 pounds would include the weight of the parachutes, stowage canisters, deployment mortars, radio beacon, markers, and all devices usually required for a successful sea recovery.

As indicated by Figure 8, the next group of liquid hydrogen tanks are depleted of fuel in diametrically opposed pairs so that the vehicle stability will not be affected. They are then detached and recovered by parachute. After separation, the tanks are allowed to tumble during descent. Because of their extremely low hypersonic ballistic parameter, they are subjected to only moderate heat flux. Hence, these tanks effectively survive re-entry heating conditions. The first four tanks require no thermal protection whatsoever, because of the superior insulating properties of the 1-inch-thick (titanium) sandwich construction. The trapped and residual hydrogen remaining within each tank will boil under the influence of re-entry heating. Each tank vent valve is set to relieve the built-up pressure at 37 psia. In this manner, internal pressure is maintained within the hydrogen tanks by passive means.

Deployment of recovery parachutes was assumed to occur at a 30,000-foot altitude. The chutes are sized to provide a subsonic ballistic parameter (W/C_DA) of 4 pounds per square foot, which results in an impact velocity of 57 fps (Figure 9). Providing that the tank recovery maneuver remains within the orbital plane and that the vehicle is launched due east from Canaveral, the first set of 4 tanks would impact 30 n.mi. down the Atlantic Missile Range (AMR), where a surface vessel would be located for recovery. The second drop (2 tanks) would impact 310 n.mi. from the "Cape," or approximately 180 n.mi. from Grand Bahamas Island, a radar tracking station within the AMR which is fully equipped with a "splash net" containing hydrophones for detection and location of sea impact.

The low orbit of the last pair of tanks, separated at an altitude of 50 n.mi., would rapidly decay (19 minutes from separation to "splash") and they would impact 2400 n.mi. downrange, or approximately 1300 n.mi. from Antigua. The 2400-n.mi. impact point is well within the expanse of the AMR. Since Antigua has tracking capabilities equivalent to those described for Grand Bahamas, it appears that the cost-reduction advantages of recovering this last pair of tanks would more than offset the weight penalty of a parachute system and a moderate amount of additional ablatant.

A slightly modified Landing Ship Dock (LSD) would be dispatched at a cruising speed of 15 knots (24 knots max.), to the predicted area of impact for the liquid hydrogen tanks. This surplus surface vessel is already equipped with two 50-ton cranes which can be used to retrieve the tanks at sea and assist in unloading at dockside. The dimensions of the ship's well (44 ft. x 252 ft.) are such that it can accommodate a pair of (25-foot diameter x 118 ft. long) hydrogen tanks (in tandem) within its hull. It may prove necessary to provide an airmat cushion between the recovered tanks and the hull of the ship, to prevent damage to the tanks from bumping. This surface vessel is ideally suited for this operation, since its high sidewalls will reduce the surface action of the sea to a moderately calm state, minimizing the relative motion of the recovered tanks within the hull. The aft gate of the ship is closed after both tanks are on board, and the water is pumped out of the hull. The tanks then are guided to settle on restraining cradles within the ship (Figure 10).

The centerbody of the ROMBUS vehicle, which contains all the guidance and electronic equipment, is parked in orbit for a period of 24 hours, or multiples thereof. When this procedure is followed, the necessity of providing cross-range velocity capability in the vehicle is avoided. In this manner, after 16 vehicle orbits, and one Earth revolution, the centerbody is in synchronous position to be recovered near the launch site. At the appropriate moment, a ground signal will command the main engines, which provide the 500 fps required for orbit ejection, to ignite. The recovered main body tentatively appears to be subsonically stable. A drogue chute will assure its transonic stability, until the vehicle has descended to an altitude of 30,000 feet. At this time, the five main recovery chutes are deployed (Figure 11). When the centerbody has descended to 2500 feet, four segments of the main engine will ignite, and the parachutes will be separated. Upon ignition, the four engine segments will provide pitch and yaw control. The attitude control system will orient the vehicle and the retro-thrust vector to cancel any horizontal velocity component. The attitude control system also provides roll control.

The main engines are cut off when the four landing legs are compressed at touch-down. The crawler is designed so it can move beneath the recovered centerbody (Figure 12), and support the vehicle near the engine attachments. The recovered weight is then supported along the same structural path which carries the engine thrust load, during boost. It appears extremely feasible to include a man in the control loop, for recovery of the centerbody. Radar tracking and guidance could supplement the recovery operation, just as Instrument Landing and Ground Control Approach systems currently assist the landing of commercial airliners.

The map shown in Figure 13 suggests a possible landing site. It is located approximately 28 n.mi. from a ROMBUS launch pad, near the intercoastal waterway and coastline. The crawler, therefore, after retrieving the vehicle, need only travel a short distance to the transportation barge at a nearby dock. The internal ellipse shown (discussed later) contains the 3-sigma dispersion. The outer ellipse represents the periphery of an unpopulated buffer zone, which may not require surface improvement.

This illustration (Figure 14) shows how the centerbody would be returned to the inspection and checkout bay of the vertical assembly building, prior to refurbishment. The crawler, supporting the recovered centerbody, would be transported by barge (from the landing site) through a canal to the rear of the inspection and checkout building. Also illustrated is a ground-based hoist, which would assist in transferring the recovered hydrogen tanks from the surface vessel to a transportation dolly. These hydrogen tanks are then rolled into the building for pressure testing, damage inspection, and eventual refurbishment prior to reuse.

ACOUSTIC CONSIDERATIONS

Acoustic problems generated by high-thrust rocket engines require special consideration when launch sites for large boost vehicles are being selected. In areas occupied by unprotected humans, minimum separation distances will be established so that the threshold of pain is not approached during launch. Parameters that have been considered in determining the separation distances required for high-thrust boost vehicles include the duration of exposure, the sound-pressure level of noise, the frequency content of noise, and the individual threshold of pain. Based upon such parameters, an overall sound-pressure level of 130 db has been selected as an acoustical criterion for unprotected humans. Estimates of the acoustic noise from the ROMBUS engines are based on such engine parameters as thrust, specific impulse, nozzle exit diameter, exhaust gas velocity, and nozzle configuration. To demonstrate the separation distance that may be required for high-thrust engines, parameters other than thrust have been fixed at values similar to those of the recent Saturn-I stage, and the acoustic power output has been increased in proportion to the ratio of ROMBUS thrust to Saturn-I thrust. Acoustic studies should be conducted, to determine the effect of noise on the forward portion of the ROMBUS vehicle, as well as on the aft end, near the engine.

Preliminary investigations indicate that acoustic fatigue may be one of the principal structural problems attendant with vehicles of this size and thrust level. The plot in Figure 16 compares the predicted ROMBUS sound pressure levels with those measured for Saturn-I, within various octave bands. Between the first octave (50 cps) and the second octave (100 cps) a predicted maximum sound pressure level is estimated at 175 db. Vibrations within various octave bands are major criteria for analyzing vibrational amplitude of components under resonant frequency. Although a maximum level of 174 db (and an overall sound pressure level of 181 db) is predicted, it should be noted that the structure will be designed for an acoustic load corresponding to a spectrum value of 158 db. At this level, an over-pressure of 13 psf, or 0.1 psi, would be experienced. A preliminary estimate indicates that 60,000 psi of unused structural strength may need to be retained in reserve, to compensate for acoustic fatigue.

Figure 17 presents a suggested solution to the problem of reflected acoustic energy from the launch complex. The parabolic dish illustrated has a focus approximately 520 feet above the bottom of the blast reflector. With the vehicle on the launch pad, the noise source (engine) is well below the focus of the parabola, causing the acoustic energy to be dispersed away from the longitudinal centerline of the vehicle. After the vehicle has ascended above the focus, the acoustic energy is concentrated at a point aft of the vehicle. It should be noted that the 80-foot nozzle diameter is only valid for vacuum conditions, when the nozzle is "flowing full." The equivalent nozzle diameter is 25 feet at sea level, due to the atmospheric pressure on the exhaust stream. Preliminary estimates indicate that the parabola should be approximately 20 nozzle diameters (500 feet) across its edges and should extend approximately 60 feet below ground level. Preliminary studies indicate that 9 db of acoustic attenuation may be realized through the use of such a parabolic reflector.

Moreover, by filling the bottom of the reflector with a water level which is 250 feet in diameter, an additional 5 db of attenuation may be acquired, through the energy-dissipation capability of the water. The water will not retain the level shown in the illustration after engine ignition, but will take on an irregular quasi-parabolic shape under the influence of the reflector. The irregular surface of the water generated by the exhaust gases will dissipate acoustic energy in the process. Preliminary investigations suggest that similar noise-attenuation devices such as acoustic baffles and fine water-spray can be incorporated into the launch pad blast deflector, further damping the reflected noise to within tolerable limits.

COST ANALYSIS

Figure 18 is a plot of the direct operating cost (DOC) sensitivity to errors in the estimated mission reliability. It indicates that if the actual mission reliability of ROMBUS were closer to 0.750 (rather than the nominal estimated value of 0.850), the DOC would increase from $12/lb to $19/lb. At the other end of the spectrum, if mission reliability were improved to 0.950 through repeated reuse and correction of subsystem malfunctions, the DOC could be reduced to $5/lb.

Figure 19 plots the previous cost data in terms of the number of uses per vehicle, and shows that for a nominal DOC of \$12/lb, each centerbody would be used approximately 5.6 times. (Each LH_2 tank might be used as many as 10 times.) When sufficient missions exist for this type of vehicle, and when the mission reliability can be improved to allow 20 or more reuses of the vehicle, the DOC would approach an asymptote of approximately \$5/lb of payload. With 20 or more reuses, the DOC would be reduced to a level comparable to that of commercial airline operations, in which costs are comprised of only the expenditures for flight operations and fuel consumption.

The cost discussion of the two preceding figures was concerned primarily with DOC, which excludes the non-recurring costs of facilities, tooling, ground support equipment (GSE), maintenance, and R&D. Figure 20 plots the total program costs, in addition to DOC. When a program with a guideline of constant expenditure rate (\$1 billion/yr) was assessed, the specific DOC was calculated to be \$12/lb of payload (to orbit). A specific total program cost of \$24/lb of payload was calculated under this same guideline. The total expenditure is \$8.4 billion, without R&D, and \$17 billion, with R&D.

When a different program with a guideline of constant launch rate (15 vehicles/yr) was assessed, the specific DOC was calculated to be \$24/lb of payload. A specific total program cost of \$69/lb of payload was calculated under this same guideline. The total expenditure, over a 10-year period, is \$3 billion, without R&D and \$9 billion, with R&D.

When available program funds are reduced to \$5 billion or less, the entire monies would be expended for developing this type of vehicle, with no allocations remaining for operational launches. This is confirmed by the estimate that it would cost approximately \$5.1 billion to develop a ROMBUS-type vehicle.

Figure 21 is cost-breakdown of the two assumed programs previously discussed. The first program is predicated on a constant expenditure rate of \$1 billion/yr, and the second is commensurate with a constant launch rate of 15 vehicles/yr. Both programs result in a vehicle development cost of \$5.1 billion, including the non-recurring cost of facilities, tooling, and GSE required for development.

The propellants required for chill-down and for topping of the tanks during an 8-hour ground hold was estimated. It was calculated that an additional 9% of LH_2 (based on tank capacity) and an additional 1.6% of LO_2 would compensate for the boil-off losses resulting from these conditions. The cost of LH_2 was estimated at \$0.27/lb for large-quantity production (full-plant capacity). The propellant entry tabulated in Figure 21 reflects these quantity and cost estimates.

The evolution of the data tabulated in Figure 21 was based on the following time estimates: (1) one week required for launch pad refurbishment, (2) one week stay-time of vehicle on launch pad, (3) 16 days required for vehicle refurbishment, and (4) 76 days of vehicle turn-around time (from launch to re-launch). Based on the preceding time estimates, the four launch pads shown in Figure 4 are capable of supporting the one billion $/yr (DOC) program.

ADVANTAGES OF SINGLE-STAGE REUSABLE VEHICLES

To support lunar operations and manned interplanetary flights of the future with a minimum of rendezvous maneuvers in Earth orbit, it appears that the next large launch vehicle should have at least four times the orbital payload capability of its predecessor; it should provide at least a four-fold reduction in DOC, or both. Reusability appears to be, by an overwhelming margin, the most desirable feature of this hypothetical launch vehicle. Only with a reusable system can the DOC for present expendable vehicles be significantly reduced. According to recent articles in industry weeklies, current cost per pound of payload to orbit have been estimated to start at $1000/lb for Saturn and be eventually reduced to $400/lb. For Saturn 5 costs will start at $250/lb and eventually be reduced to $150/lb. A reusable vehicle could reduce these costs to a level of $12/lb (for constant expenditure rate program), or $23/lb (for a constant launch rate program of 15/yr), even with a moderately conservative estimate of reuses (5 to 6).

Use of a completely recoverable vehicle would increase eventual reliability. For example, commercial aircraft probably would be much less reliable if they were expendable. Post-flight examination of aircraft, which can be returned with large segments of equipment inoperative, has been extremely important in the attainment of performance and reliability improvements. A comparable situation would exist for reusable space vehicles: inspection of components which have failed in flight would (1) facilitate troubleshooting, and (2) enable solution of technical problems after substantially fewer flights than would be required if non-reusable vehicles were used.

Results of the cost analysis discussed herein clearly demonstrate the cost reductions which can be realized from a reusable vehicle, even for a moderately conservative number (5 to 6) of reuses. Preliminary trends ($19/lb of payload) from a cost investigation of a typical two-stage, reusable configuration also establish an economic superiority for the reusable vehicles as a one-stage-to-orbit configuration. The economic advantages of the single-stage vehicle offer cost reductions derived from the operational simplicity and increased reliability of such a system. Maximum program economy is acquired when the entire launch vehicle can be recovered. Moreover, the turn-around time for a reusable orbital booster (76 days for ROMBUS) is considerably less than the time required for recovery and refurbishment of a comparable two-stage reusable vehicle. This factor tends to diminish the number of vehicles (and launch facilities) required in the squadron inventory, in order to maintain a specified launch rate.

When only one stage is required to attain orbital velocity, vehicle development costs are lowered, since only a single set of tanks and engines need be developed. Segmented, but identical, combustion chambers offer further cost reductions during development, where only one engine module need be extensively qualification-tested. Furthermore, the record corroborates the contention that when separate stages are required, the total program costs are amplified out of proportion, merely because of the manufacturing participation of more than one major contractor. In addition, use of only one set of tanks and engines minimizes costs of ground support and checkout equipment for the single-stage vehicle.

Reliability, which has a profound effect on DOC, is improved with a single-stage vehicle, although an acknowledged performance and weight degradation is incurred. The absence of stage separation eliminates the need for ordnance devices and retro-rockets for interstage detachment. Altitude start of engines is avoided. All engines can be effectively checked out during the brief hold-down period on the launch pad. The absence of the stage-separation maneuver, with its attendant tipover problems under dynamic pressures, results in large reliability gains. Without stage separation, less flight instrumentation and fewer malfunction detection systems are required.

A historical sample of all U.S. space launchings will show that the nation has already succeeded in achieving fairly impressive booster reliabilities. However, when upper stage malfunctions are considered, the mission reliability has been severely compromised. The undeniable conclusion inherent in these recorded facts should be readily apparent.

Indeed, the alternative choice, non-recovery of boosters, deserves serious consideration. How long will it take for our spaceways to become littered with spent booster cases, if we don't begin returning them? There appears to be no assurance that the entire booster, including engines, would be incinerated during re-entry. According to a recent trade journal report, (as of April 30, 1963) 318 man-made objects were in earth orbit, of which 59 are payloads (52 U.S., 7 Russian). This indicates that the orbits of 259 spent rocket motors and other space debris have not yet decayed to the point where they will present a hazard to the populated areas on Earth.

In response to the question, "Does it really pay to recover boosters?", the economic rationale discussed herein provides the best reply. For the sake of expediency and urgency, the vehicle need not be developed for immediate reusability, provided that the initial design is committed to an eventual mode of recovery. However, it appears imperative to design the next launch vehicle for reusability, and to incorporate the necessary provisions for recovery features into the vehicle design, from its inception.

It is an extremely difficult task to postulate the payload sizes and weights which will be required during the next decade. Therefore, the next large launch vehicle, which may be the only one our national economy can afford for years to come, should be designed for maximum mission flexibility. The vehicle must be designed to perform the limited number of million-pound payload missions. However, if the vehicle is, in fact, reusable, its propellants can be off-loaded for other missions. It seems certain that there will be a requirement quite soon for a reusable "trucking system" which can provide the Earth-surface-to-Earth-orbit transportation necessary for supporting the many relatively small payload missions involved with extended orbital operations.

As an additional application for off-loading of propellants, the ROMBUS vehicle could serve as a compatible launch platform for a nuclear earth-escape stage. When a constant lift-off weight is maintained, and four of the LH_2 tanks are removed (the LO_2 tank would be filled to approximately 50% capacity), the ROMBUS vehicle can function as a half-stage for boosting a nuclear upper stage (plus payload) to an altitude of 106,000 feet, where the reactor could safely be ignited.

In conclusion, it appears imperative that the next generation of orbital transport carriers to supercede the Saturn class of boosters should be designed as reusable "trucking" systems. Very often, a dependable 10-ton truck can be effectively and economically used for a 2-ton delivery job, when that truck is truly reusable. No medium of transportation can long survive the extravagance of using the carrier vehicle only once. Over the years to come, long-range economy, and not immediate space-spectaculars, may well establish superiority of space exploitation.

It is an impossible task to postulate the mission reliability improvements which can be realized through reusability. In due time, the technical skepticism currently associated with 20 or more booster reuses ($5/lb) may be completely dispelled. The one most paramount conclusion, which became a prevailing manifestation during the subject investigation, can be clearly stated with fundamental logic--the current avenues of using (expendable) progressively larger "brute force" techniques of weight-lifting can only lead to a dead-end of perhaps $150/lb of payload, whereas an explicit direction can be pursued which may lead to an eventual cost of only $5/lb. Selection of the most advantageous "fork in the road" is basically a straightforward, indisputable decision......................Finis

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9. Goldbaum, G. C. and White, J.F.: "Effects of Vehicle Cost on Design and Sizing of Multi-Stage Rockets" presented to the 4th Symposium on Ballistic Missile and Space Technology at UCLA in August 1959. Also Douglas Aircraft Co. engineering paper No. 801.

10. Stone, John W., Adv. Launch Veh. Studies Mgr. NASA: "Future of Large Launch Vehicles" presented to AIAA Second Manned Space Flight Meeting, Co-sponsored by National Aeronautics & Space Administration on April 22-24, 1963, in Dallas, Texas.

11. Kalitinsky, Andrew: "NOVA Launch Vehicle Design Studies" presented at the Syposium on the Exploration of Mars sponsored by the American Astronautical Society on June 6-7, 1963 at Denver, Colorado, Preprint (5).

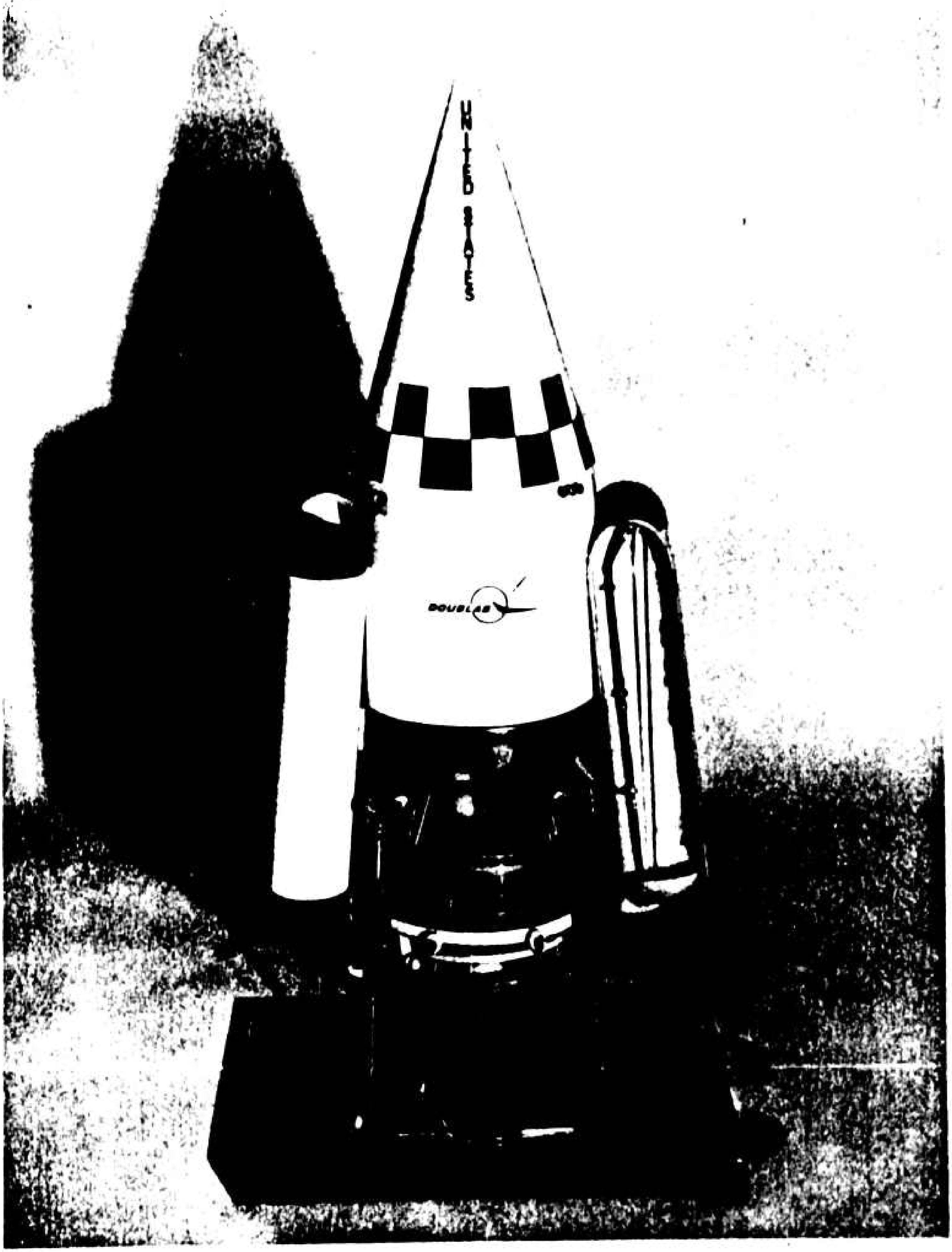

FIGURE 1

ROMBUS VEHICLE CONFIGURATION

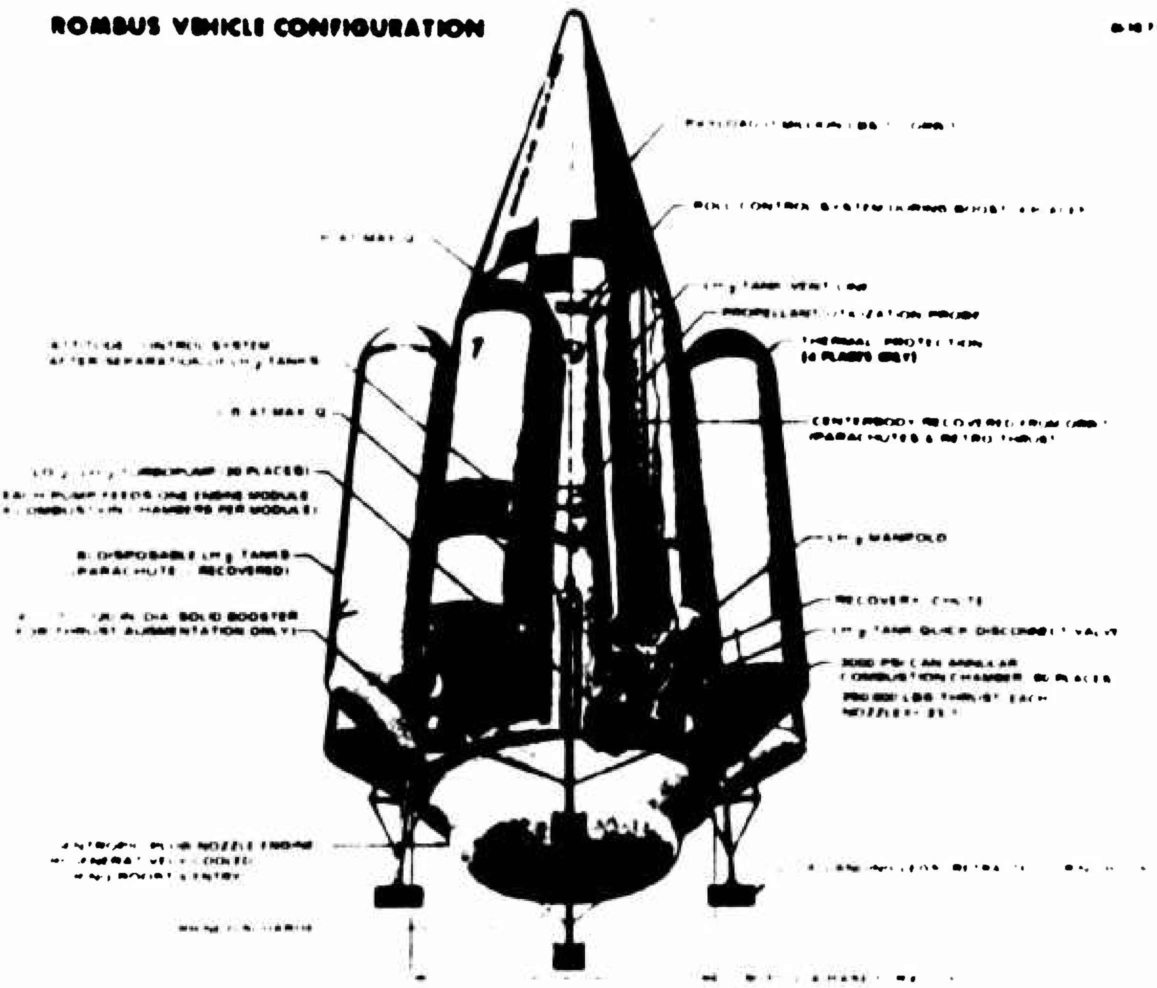

FIGURE 2

ROMBUS ENTRY CONFIGURATION

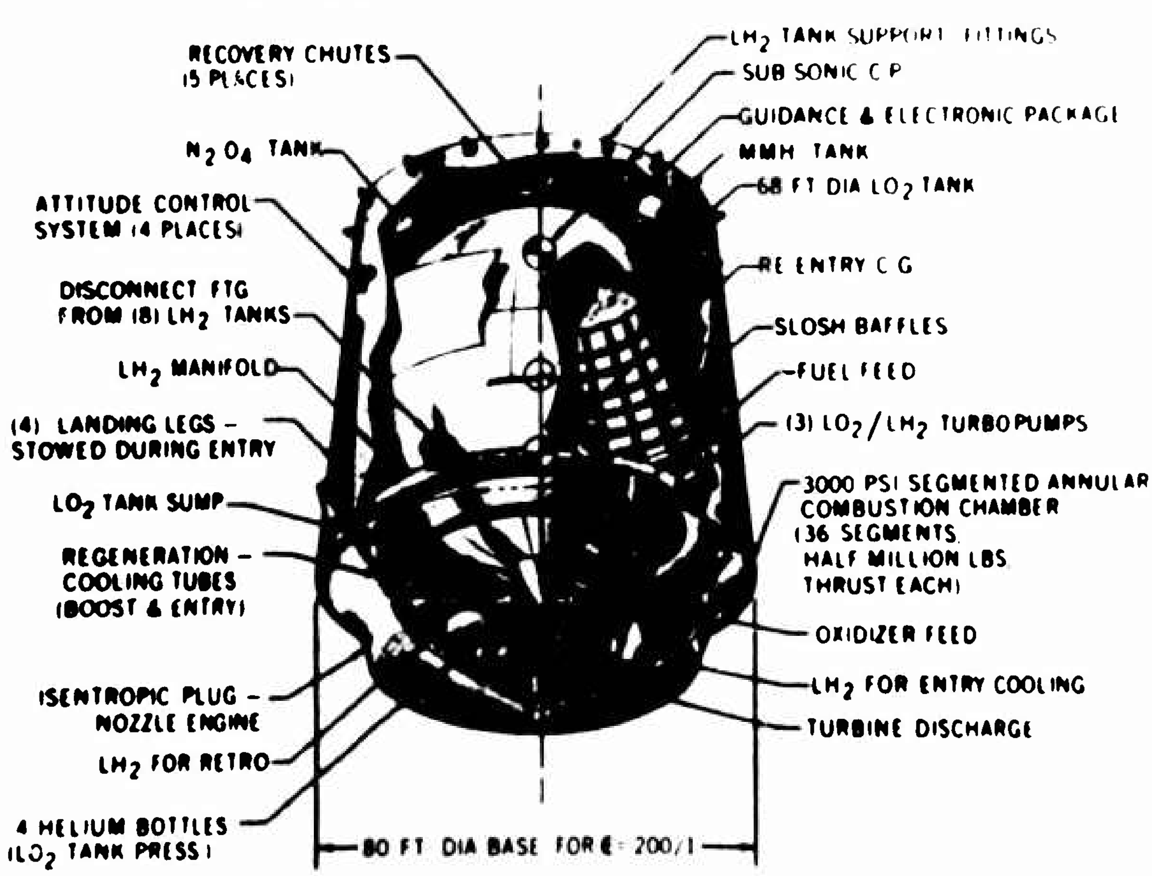

FIGURE 3

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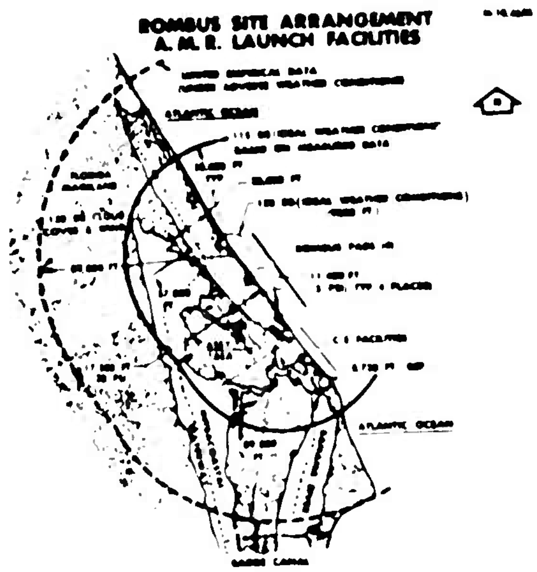


FIGURE 4

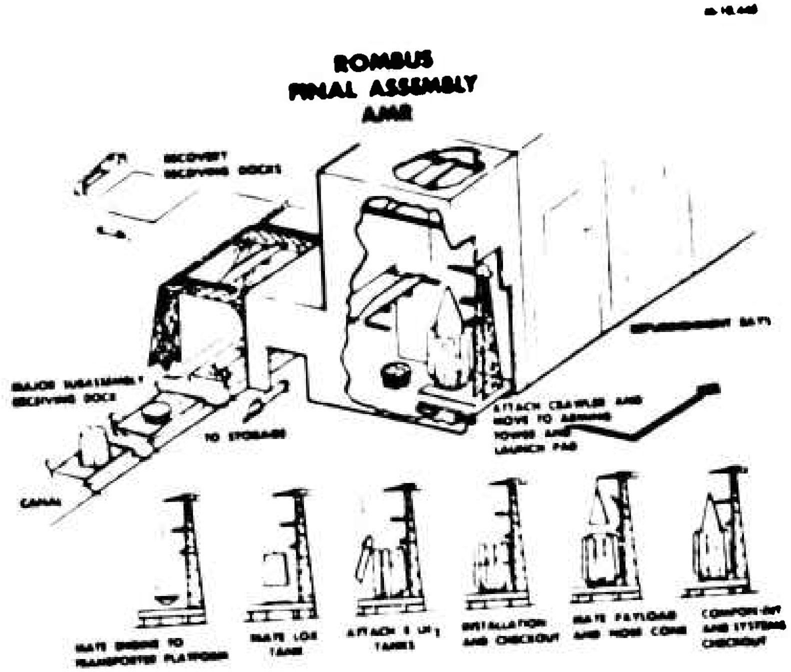


FIGURE 5

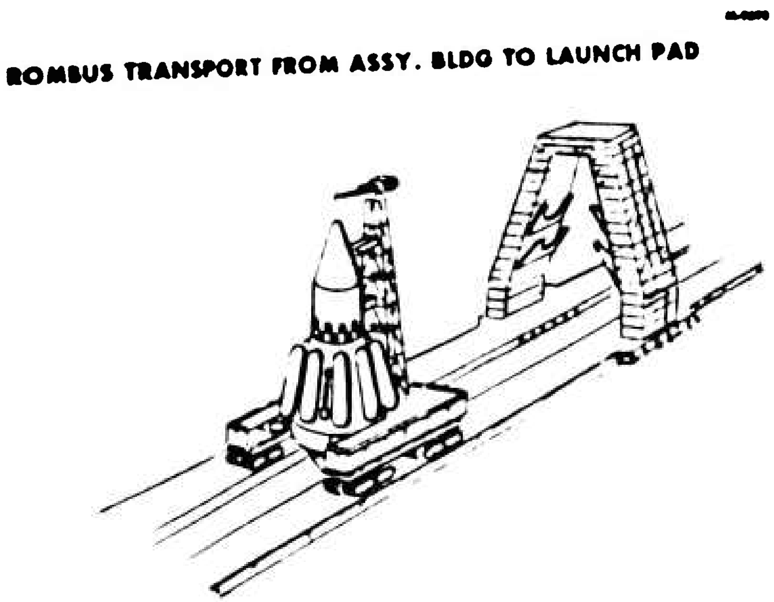


FIGURE 6

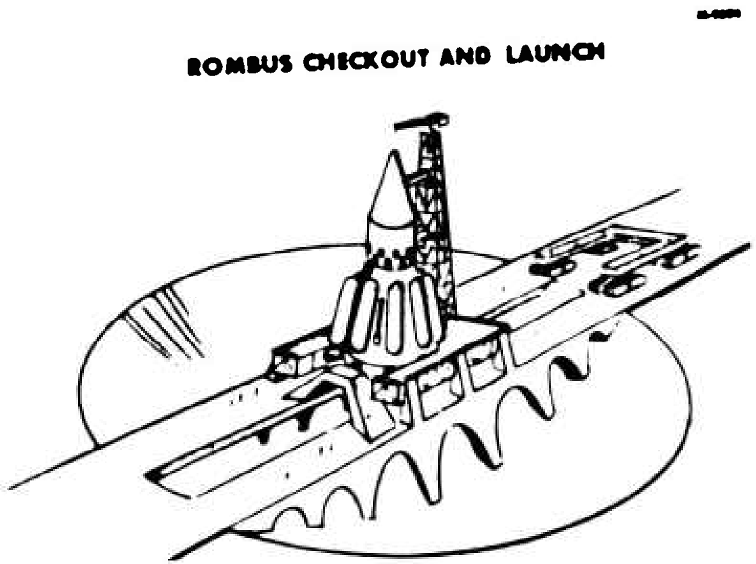


FIGURE 7

ROMBUS BOOST AND SEPARATION OF LH_2 TANKS NO. 5 & 6

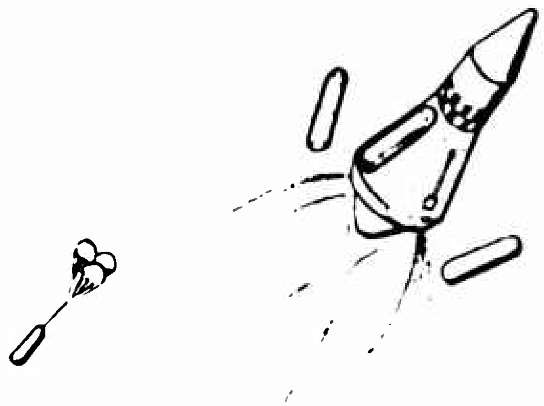


FIGURE 8

RECOVERY OF LH_2 TANKS

RECOVERED WEIGHT = [illegible] LBS

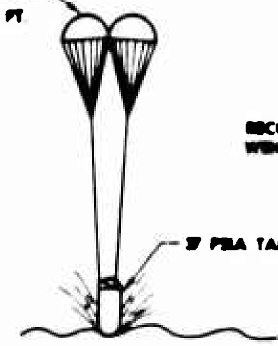


FIGURE 9

ROMBUS LH_2 TANK RECOVERY

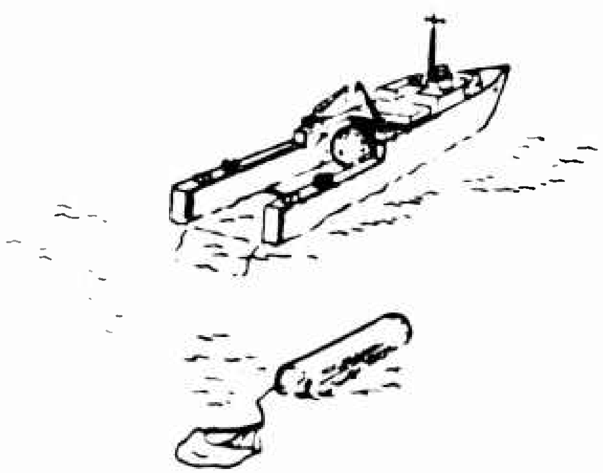
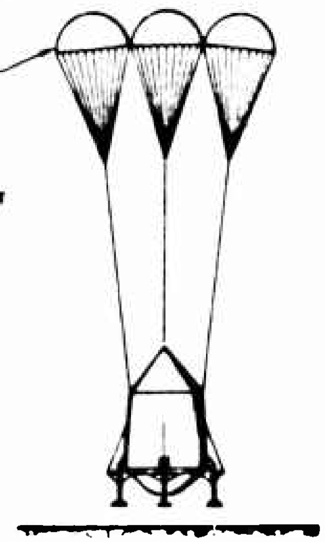


FIGURE 10

ROMBUS MAIN STAGE RECOVERY



- RECOVERED WEIGHT = 555,640 LBS
- PARACHUTE DEPLOYMENT ALTITUDE = [illegible] FT
- RETRO-THRUST BEGINS AT 2500 FT
- VELOCITY AT 2500 FT WITH CHUTES = 170 FPS
- RETRO-THRUST DURATION = 12 SECONDS
- LANDING GEAR STROKE = 2 FT
- MAX. TOUCH DOWN DECELERATION = 2 g's

FIGURE 11

ROMBUS MAIN BODY RECOVERY WITH "CRAWLER"

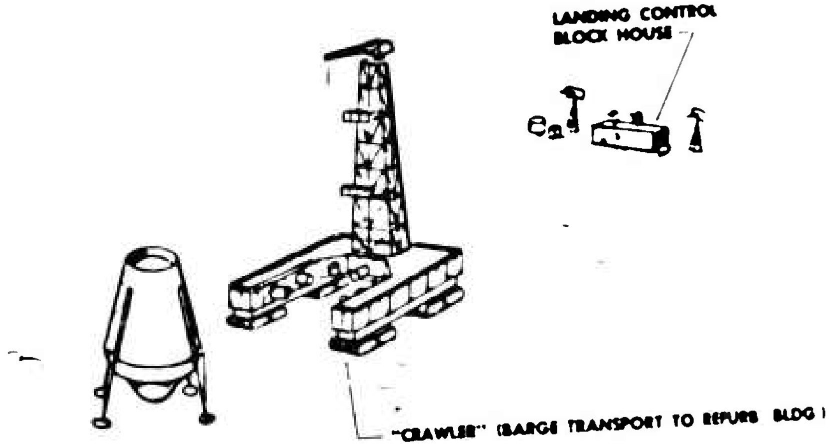
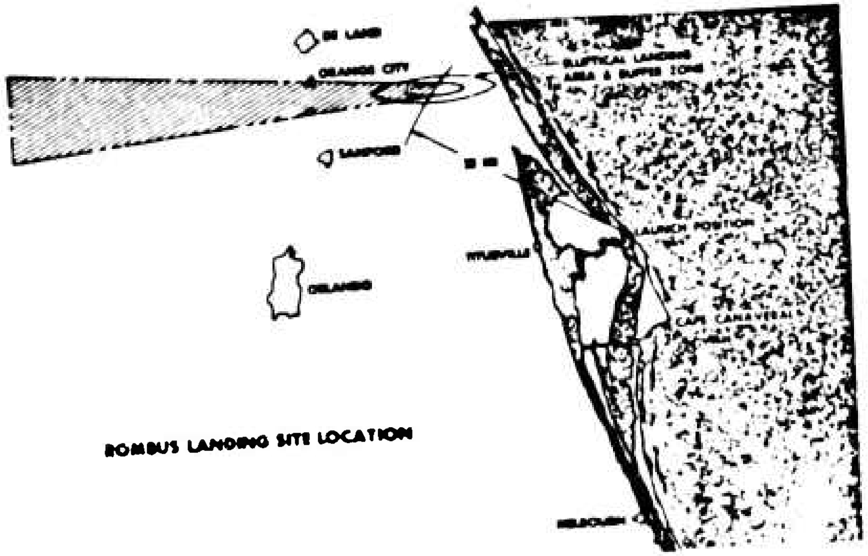


FIGURE 12



ROMBUS LANDING SITE LOCATION

FIGURE 13

ROMBUS VERTICAL ASSY. AND CHECK OUT

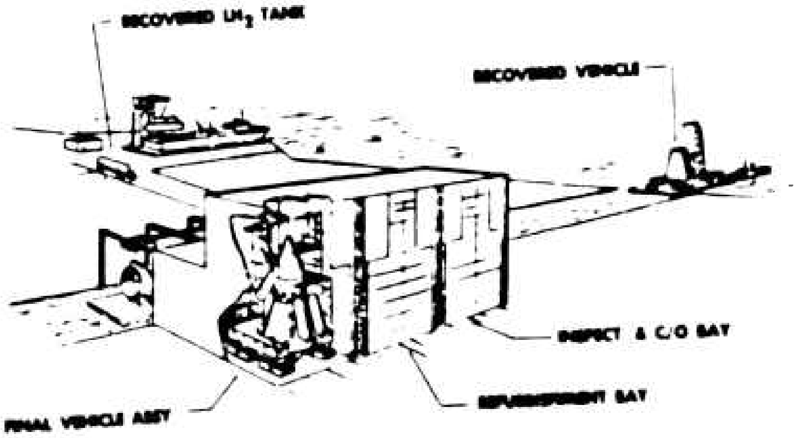


FIGURE 14

FIGURE 15

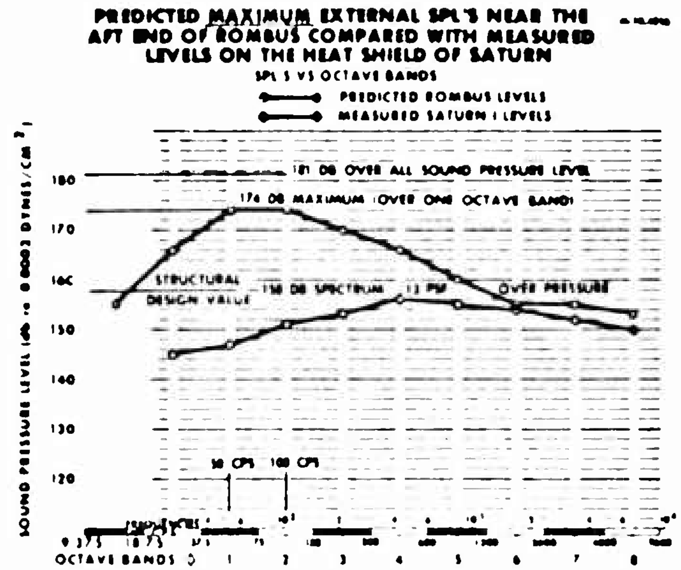


FIGURE 16

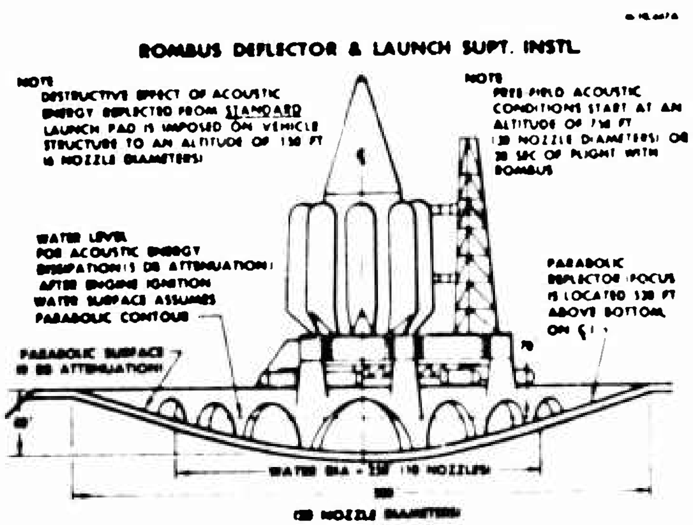
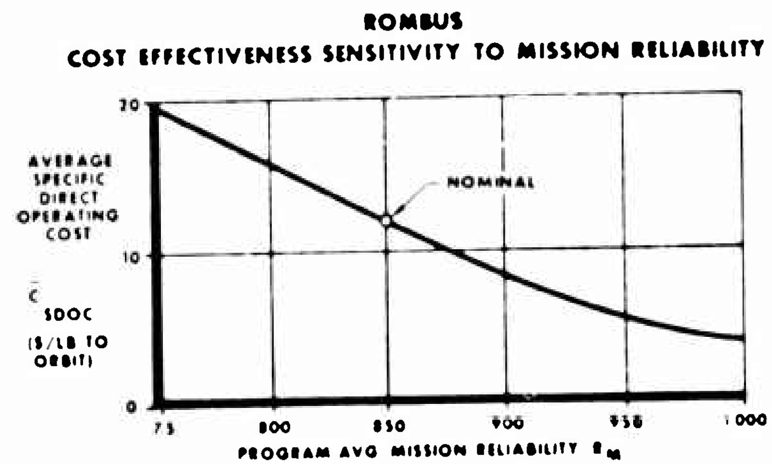


FIGURE 17



FIGURE

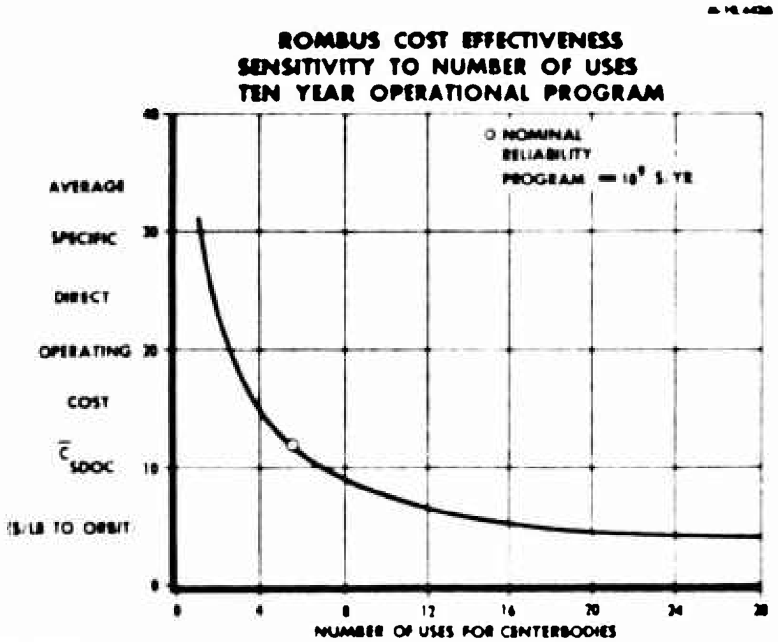


FIGURE 19

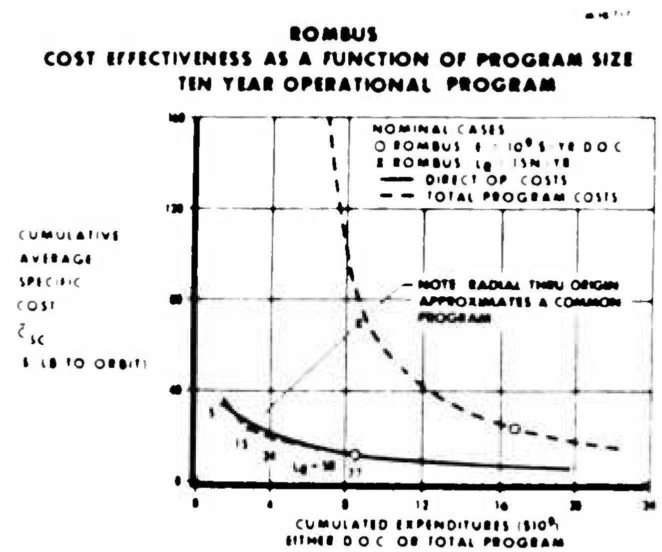


FIGURE 20

ROMBUS PROGRAM COSTS

MILLIONS OF DOLLARS

TOTAL PROGRAM - NINETEEN YEARS
OPERATIONAL PROGRAM - TEN YEARS

ITEMS	PROGRAM CRITERIA: D O C EXPENDITURE RATE - 10^9 $/YR		CONSTANT LAUNCH RATE - 15/YR	
DEVELOPMENT PROGRAM				
ENGINEERING DEVELOPMENT & LAB TEST	1 714		1 714	
STATIC & FLT TEST VEHICLES	1 167		1 167	
STATIC & FLT TEST OPERATIONS	752		752	
OTHER	[illegible]		[illegible]	
SUB-TOTAL	[illegible] (1)		[illegible] (1)	
FACILITIES	ADD OPS	DEV	ADD OPS	DEV
MANUFACTURING & PROD TEST	14	107	6	107
STATIC TEST	00	94	00	94
FLIGHT TEST	00	354	00	354
ADDITIONAL LAUNCH OPERATIONS	700	000	33	000
SUB-TOTAL	717 (A)	555 (2)	39 (A)	555 (2)
TOOLING ENGINEERING & FABRICATION	19 (B)	102 (3)	18 (B)	102 (3)
GSE ENGINEERING & FABRICATION	750 (C)	361 (4)	94 (C)	361 (4)
TOTAL DEVELOPMENT (OR ADD INVEST FOR OPL)	1 514 (D)*	5 106 (5)*	151 (D)*	5 106 (5)*
OPERATIONAL PROGRAM				
VEHICLE PROCUREMENT	7 036		2 613	
PROPELLANTS	640		135	
OPERATIONS	715		219	
TOTAL DIRECT OPERATING COST	[illegible]*		[illegible]*	
TOTAL PROGRAM MANAGEMENT	300*		160*	
TOTAL MAINTENANCE (OPERATIONAL)	1,300*		530*	
GRAND TOTAL PROGRAM COST	16,701**		[illegible]**	

NOTE (A) + (B) + (C) = (D)
(1) + (2) + (3) + (4) = (5)   SUM OF FIVE ITEMS MARKED * = **

FIGURE 21