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MANNED ORBITAL TRANSFER VEHICLE (MOTV)

volume 4

supporting analysis

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GRUMMAN AEROSPACE CORPORATION

MANNED ORBITAL TRANSFER VEHICLE (MOTV)

volume 4
supporting analysis

prepared for
National Aeronautics and Space Administration
Johnson Space Center
Houston, Texas

prepared by
Grumman Aerospace Corporation
Bethpage, New York 11714

November 7, 1979

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FOREWORD

This final report documents the results of a study performed under NASA Contract NAS 9-15779. The study was conducted under the technical direction of the Contracting Officer's Representative (COR), Herbert G. Patterson, Systems Design, Johnson Space Center. Mr. Lester K. Fero, NASA Headquarters, Office of Space Transportation Systems, Advanced Concepts, was the cognizant representative of that agency.

The Grumman Aerospace Corporation's study manager was Charles J. Goodwin. The major contributors and principal investigators were Ron E. Boyland, Stanley W. Sherman and Henry W. Morfin.

This final report consists of the following volumes:

- Executive Summary - Volume 1
- Mission Handbook - Volume 2
- Program Requirements Document - Volume 3
- Supporting Analysis - Volume 4
- Turnaround Analysis - Volume 5
- Five Year Program Plan - Volume 6

1 - INTRODUCTION

This document is issued in support of the Mission Handbook and Program Requirements documents. Its intent is to give background analysis and data that will impart some insight into the work performed to arrive at conclusions defined in those documents. The data are organized in subject, rather than task or chronological order.

Work performed up to the midterm review of phase 2 is discussed here. The last quarter is reported in the final presentation.

2 - GENERIC MISSIONS

A listing of 100 potential MOTV user programs was identified and categorized as follows:

- 35% Public service
- 27% Defense
- 21% earth observations and astronomy
- 12% Energy.

From this list of potential programs, a range of 20 generic missions was difined to a level that will enable a potential user to find a mission close to the one he has in mind, and thus determine such parameters as MOTV capabilities, number of STS flights to support its mission, mission-peculiar equipment requirements, parametrics on mission phasing and ΔV requirements, ground and flight support requirements, recovery considerations, and IVA/EVA trade analysis. This information is reported fully in the Mission Handbook, which is issued with this report.

This work took place mainly in phase 1 and, in summary, we learned the following:

• Potential MOTV Users	Accommodated by 5 mission classes comprising 20 generic missions	
• For These 20 Missions:		
Crew Size	2 to 3	
Duration	4 to 30 days	
Main Propulsion Avg	6785 to 8925 m/sec	(22,255 to 29,276 fps)
RCS Avg	63 to 290 m/sec	(205 to 685 fps)
Payload Out	2900 to 37,000 kg	(6400 to 82,000 lb)
Payload Back	2900 to 16,000 kg	(6400 to 35,000 lb)
Gen Purpose Mission Equip.	Manipulators	
	Stabilizer	
	EVA suits	
	EVA tools & C/O equip	

- For MOTV Study Purpose — 7 of the 20 generic mission cover the range of size/duration/orbit/function

In phase 2, the data were updated, as necessary, to reflect revised weights and contingencies.

Also in phase 2 of the study, two contractors were appointed to study the OTV propulsion system. Grumman defined a Design Reference Mission (DRM) for them to work to. This mission is S1 generic mission, which services four communications satellites, 90° apart in GEO. They use the MMS subsystems support modules which are replaced on a routine basis.

The Mission Handbook contains a full description of this mission and illustrates the performance and functions of the MOTV. Some supporting information is contained in Section 6 of this report.

3 - MISSION MODES

3.1 STUDY MATRIX

It was necessary to identify, early in the study, a representative mission mode or modes to provide operations, interfaces, performance, and cost data for study of the payload elements. Three concepts were considered:

- "Bare-bones" — Modest performance for minimum cost
- Full-capability — Performance to satisfy all or most of the generic missions
- Evolutionary — From "bare-bones" to full capability.

The study options for these concepts were:

- Launch
  - Single launch/mission or multi-launch/mission
  - Standard STS (28,300 Kg), augmented STS (40,000 Kg), or in-between
  - Cargo bay size unchanged
- Return Options
  - All-Propulsive Orbit Transfer Vehicle (APOTV)
  - Air-Maneuver Orbit Transfer Vehicle (AMOTV)
  - Air-Maneuver Return Vehicle (AMRV)
- Stages
  - 1, 1½ or 2 (slingshot)
- Main Engines Options
  - Min modification RL 10,444 sec I_{sp}, 15,000 lb thrust
  - Advanced space engine, 473 sec I_{sp}, rubberized thrust

3.2 SINGLE STS LAUNCH PER MISSION

Considering the "bare-bones" concept, the objective was to minimize cost (in particular DDT&E). Performance assumed was two crew, six-days mission, 3,900 Kg deployed and 2,400 Kg returned, maximum of three missions per year. Among the

manned capsule concepts considered were LM and MRWS derivatives, growth Gemini, Spacelab modules, and a dedicated capsule.

For an MOTV launched to LEO in a single Shuttle flight, the matrix shown in Fig. 3-1 gives the mission modes considered for candidate CTV recovery systems with propulsion stage options. Propulsion stages are categorized as single-stage, 1½ stages, and 2 stages. There are two systems alternates for recovery from LEO: an all propulsive (APOTV) vehicle; and an aero maneuvering (AMOTV) vehicle. A third system (AMRV) expends the propulsion stages and recovers only the crew capsule for a "land" Landing.

The APOTV uses propulsion for all orbit transfer and phasing operations. The single-stage vehicle is launched and recovered from LEO by the STS. With the 1½ stage, Fig. 3-1 shows drop tanks retained until GEO, where they are parked for subsequent disposal. Alternatively, each may be deorbited and burned-up in the atmosphere and this became the baseline later in the study. AMTOV uses propulsion for all orbit transfer and phasing, except the final circularization at LEO. For this maneuver, the vehicle is designed for aerobraking and circularization by "skipping" in the atmosphere. With this exception, the mission modes for the AMOTV follow those of the APOTV.

The AMRV mission modes are similar to the AMOTV, except that here the crew capsule deorbits directly to earth from GEO and expends the propulsion stages to either remain in GEO or to deorbit for burn-up in the atmosphere. In the single stage, propulsion separates from the crew module after deorbit firing and enters on its own. The 1½ stage is similar except for disposition of the drop tanks. A two-stage vehicle gives reasonable performance with the AMRV since the ascent stage can be separated after GEO circularization, and left in orbit. The deorbit propulsion is relatively small and separates, after burn, to enter the atmosphere on its own.

Five mission mode concepts, APOTV one and 1½ stages, AMRV one and 1½ stages, and AMRV single-stage, were candidates for development to determine feasibility, costs, and performance.

Referring to Fig. 3-2, variation of DDT&E costs with MOTV performance is given for each of the five concepts. Performance is given in terms of roundtrip payload to GEO, allowing for any TPS and recovery system penalties. A performance envelope is given for each concept. The envelope is shaped by the capabilities of two alternative engines, the RL10 (I_{sp} of 444 sec) and a new space engine (I_{sp} of 473 sec), and by

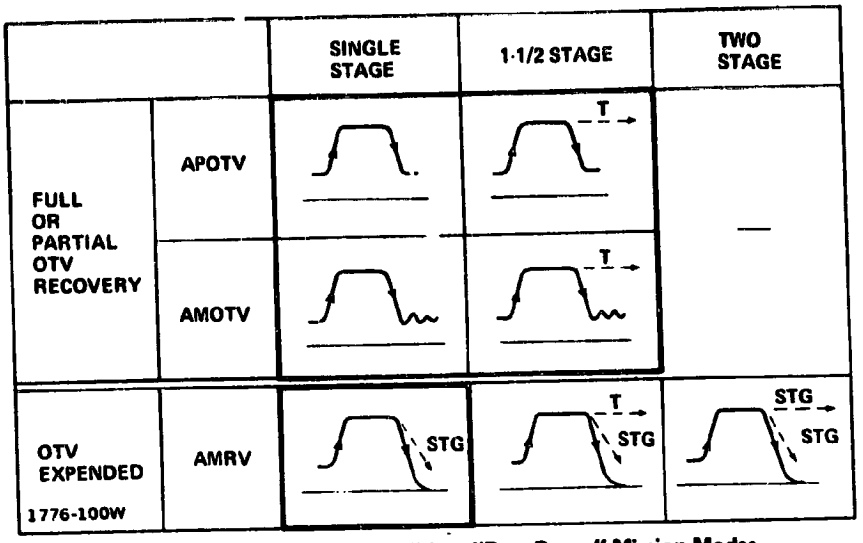

Fig. 3-1 Single Launch — Candidate "Bare-Bones" Mission Modes

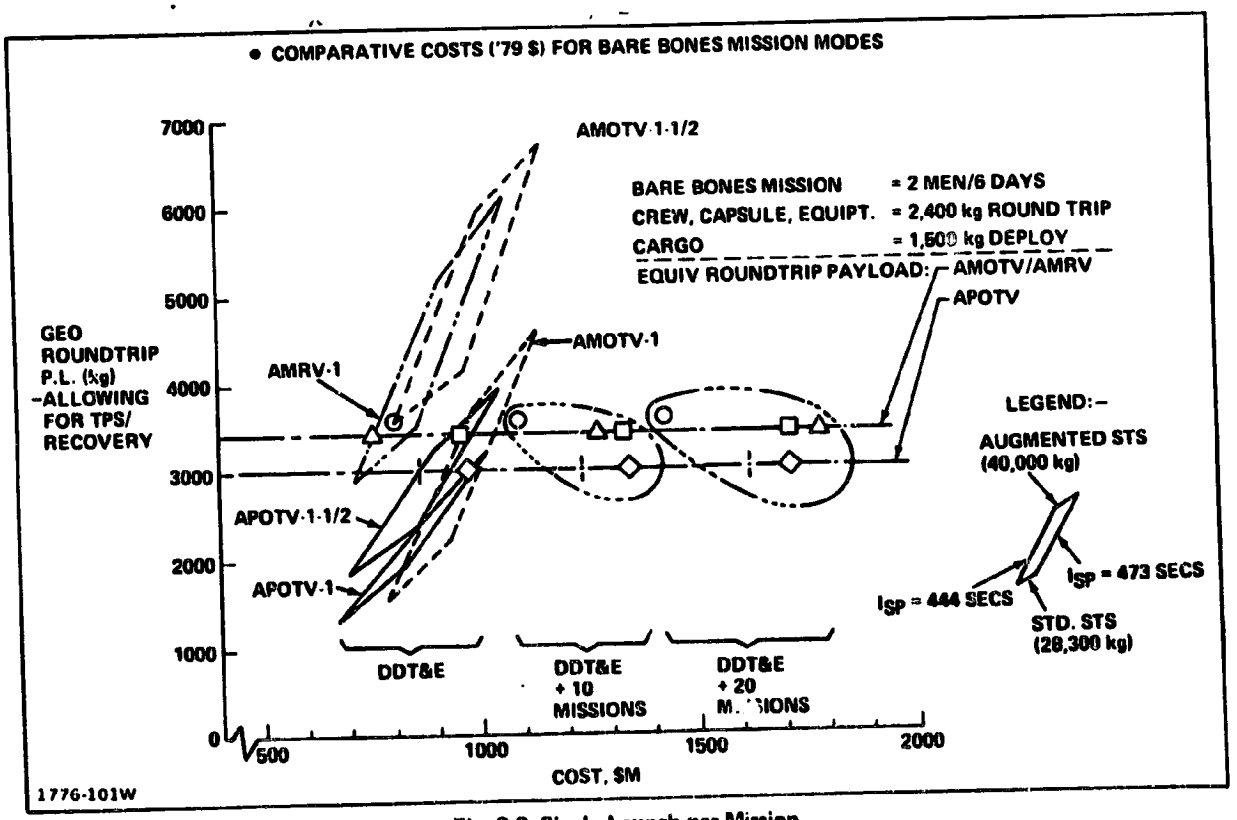

Fig. 3-2 Single Launch per Mission

the useful payload restrictions of the standard shuttle (28,300 Kg) and an augmented shuttle (40,000 Kg). With the APOTV and AMOTV, there is noticeable increase in performance in going from single-stage to 1½-stage, but little increase in DDT&E costs. Figure 3-2 also examines the impact of adding costs for 10 and 20 "bare-bones" missions. The payload, established as 2400 Kg roundtrip with an additional 1500 Kg deployed, is converted to an equivalent roundtrip payload of 3000 Kg for the APOTV and 3400 Kg for the AMOTV and AMRV.

Considering the APOTV, the single-stage does not have much in hand in meeting the required performance, whereas the 1½ stage meets it with the lower-performing engine but requires a launch to LEO capability approaching that of the augmented shuttle. As missions costs are added, the costs gap between the two modes gradually narrows as the number of missions increases. The AMOTV single-stage uses the lower-performing engine but requires an augmented shuttle for launch. Its increase in costs, with missions, approximates that of the APOTV's. The AMRV has lowest DDT&E costs but soon overtakes the others when missions costs are added. The most cost effective mode is the AMOTV 1½ stage, which can more than meet the 3400-Kg payload requirement, yet uses the lower-performing engine and the standard Shuttle for launch to LEO. DDT&E costs are a little higher than for the AMRV, but delta increase in costs with missions are much lower than for any of the other four mission nodes.

Results from the single STS launch per mission are summarized in Fig. 3-3. The recommendations were that, if evolution is a driver, the single-stage APOTV should be pursued; but if evolution is not a driver, then the 1½ stage should be continued.

3.3 MULTIPLE STS LAUNCHES PER MISSION

Turning to multiple STS launches to transfer an MOTV to LEO, Fig. 3-4 gives the mission mode matrix for candidate OTV recovery systems with propulsion stage options. It follows the single STS launch rationale with the following exceptions. With APOTV, a two-stage is viable, and here the mode deorbits the first stage after burn-out, for recovery from LEO by a Shuttle. The second stage provides propulsion for the rest of the mission, after which it is also recovered by the Shuttle. The two-stage AMOTV follows this pattern. With the AMRV, the two-stage recovers the first stage by deorbiting it to LEO for recovery by a Shuttle. From then on the two-stage mission mode follows the single-stage.

Concepts for eight of the nine options were developed to determine feasibility, performance, and costs. The option omitted was the single-stage AMOTV which had a

	APOTV-1	APOTV-1-1/2	AMOTV-1	AMOTV-1-1/2	AMRV-1
COMPETITIVE COST ① ($M)	1756	1674 ✓	1758	1606 ✓	1842
20% MARGIN (MIN)		✓	✓	✓	✓
CARGO DENSITY IN STS (kg/m^3)	38 ✓	70	96	100	55 ✓
RETRIEVE BULKY CARGO	✓	✓			
LOW DEV RISK	✓	✓	GUIDANCE TPS ($220m^2$)	GUIDANCE TPS ($80m^2$)	✓
OTV CORE SUITABLE FOR MULTIPLE LAUNCHES	✓		✓		✓
ADV SPACE ENG	✓	✓	✓	✓	
AUGMENTED STS	✓	✓	✓		
	RECOMMENDED WITH EVOLUTION TO MULTILAUNCH	RECOMMENDED NO EVOLUTION			

① • DDT & E + 20 MISSIONS
• TWIN MAIN ENGINES
• 2400-kg ROUND TRIP (CREW-CAPSULE-EQUIP)
• 1500-kg DEPLOY (MISSION HARDWARE)

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Fig. 3-3 Single Launch/Mission Results

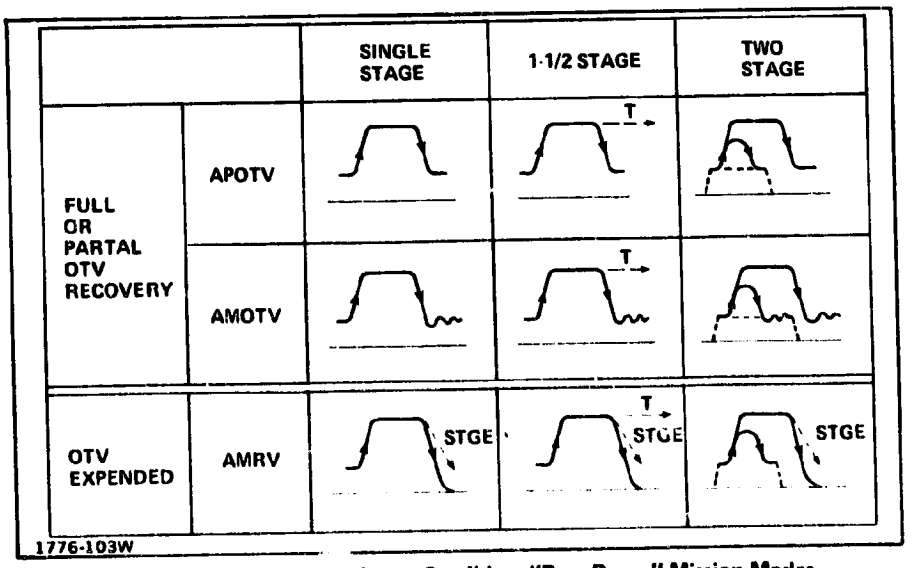

Fig. 3-4 Multiple Launches – Candidate "Bare Bones" Mission Modes

large monolithic OTV, propellant-loaded in orbit by fuel transfer from a tanker. It was severely volume-limited by the Orbiter cargo bay; in fact, it could not fully absorb the weight provided by two STS launches and, as such, would only just qualify as a Multi-Launch concept.

Figure 3-5 gives costs per mission as a function of roundtrip payload for three APOTV candidates. Each concept is illustrated; its DDT&E in 1979 constant $ is noted, and the number of launches per mission is indicated. The hatched line at the 3000 Kg roundtrip payload level shows the minimum required "bare-bones" performance. Engines are low cost (444 sec I_{sp}), and the lauchers are standard Shuttles. The 1½ stage concept has a significantly higher performance than the others. The two stage common concept, slingshot mode, is volume-limited to not more than four launches. When less than three launches are attempted, it deteriorates to a single-stage mode with low performance. For reference, the single launch per mission APOTV one stage candidates are shown in a small group at about $35M per mission.

AMOTV 1½ and two, two stage concepts have performance and cost comparisons illustrated in Fig. 3-6. Again the 1½ stage concept has the highest performance. Of the two, two stage candidates, one has a lower stage with clustered tanks that flies in an All Propulsive (AP) mode; the upper stage alone employs Air Maneuvering (AM) on return. This configuration depends on tank clustering for LEO build-up. The other two stage has two common stages; both use the AM mode and depend on fluid transfer in orbit to fill their tanks. As a multi-launch concept it is inflexible, being volume-limited to only three launches per mission. Both the two stage concepts have higher operational costs than the 1½ stage because of the need to recover two propulsion units.

Figure 3-7 shows costs vs performance for three AMRV candidates. Once more, the 1½ stage has the highest performance, but this time by only a small margin. The one stage "clustered" (at launch identical to the 1½ stage) follows as a close second. Loaded by fluid transfer in LEO, the two stage common arrangement is volume-limited with five launches and decays to a single-stage if the two launch level is assayed. The lower partner of this two stage stack is unique in that it returns to LEO and is recovered, whereas all other AMRV propulsion units are expended and burn-up on entry.

In view of the across-the-board superior performance provided by the 1½ stage mission mode, it is desirable to minimize the one disadvantage of that concept, namely, drop tanks left in space. Considering a cluster tank APOTV, Fig. 3-8 shows two bounds to the problem: the highest performance "classic" 1½ stage, which drops off the tanks

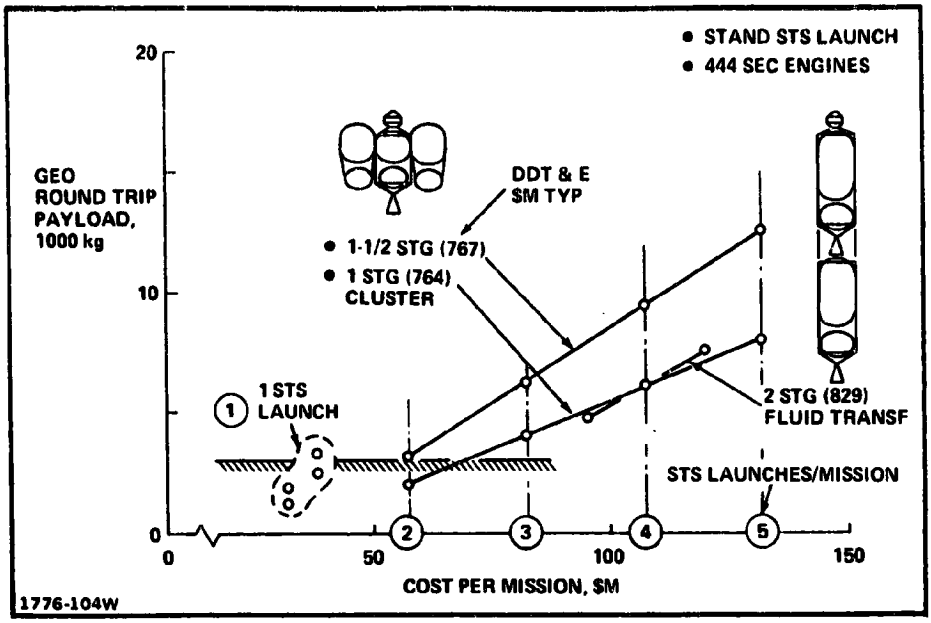

Fig. 3-5 APOTV Performance & Costs ('79 $M)

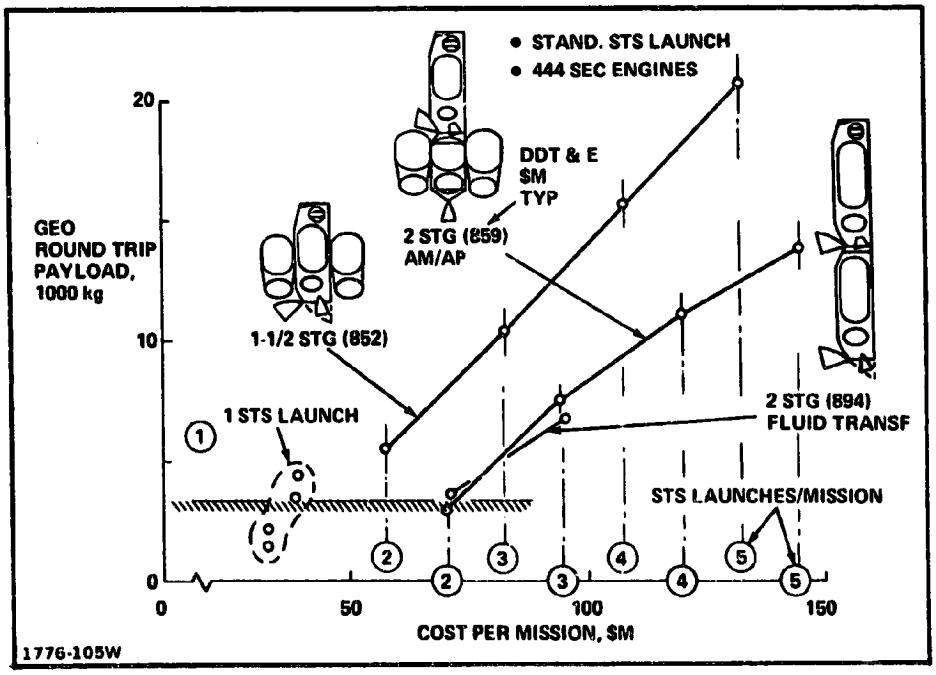

Fig. 3-6 AMOTV Performance & Costs ('79 $M)

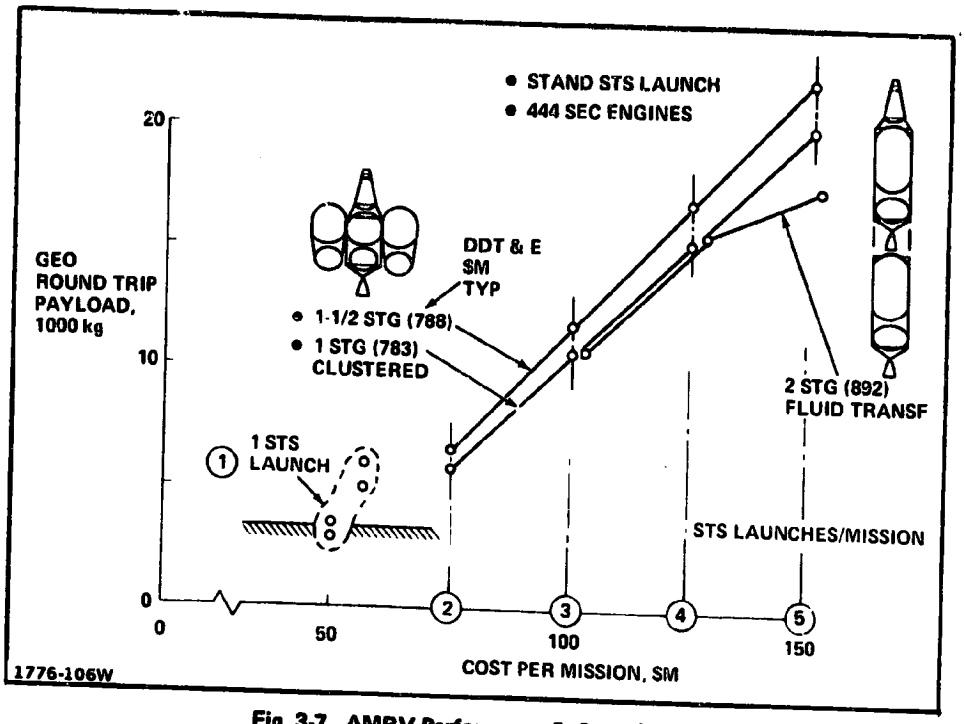

Fig. 3-7 AMRV Performance & Costs ('79 $M)

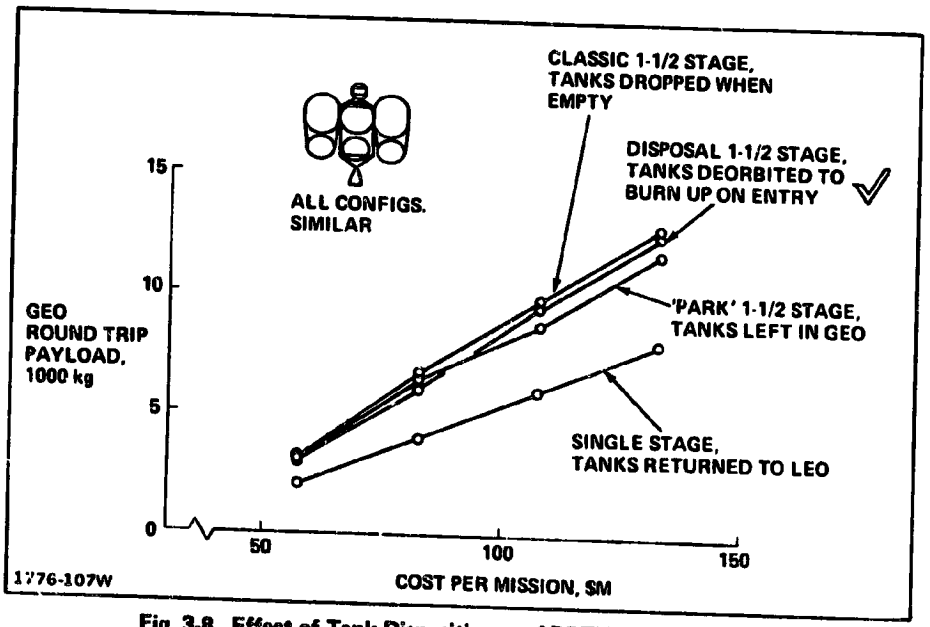

Fig. 3-8 Effect of Tank Disposition on APOTV Performance

as soon as they are empty; and the single-stage which brings all the tanks back to LEO. Two intermediate approaches are shown, both remarkably close to the "classic" 1½ stage in performance. In one, the tanks are retained on the core vehicle until geosynchronous orbit is reached. They are then parked there as a group in a known, relatively innocuous graveyard. In the other approach, each tank is provided with a small deorbit motor which is fired after separation to ensure re-entry and burn-up in the earth's atmosphere. This is the concept for which 1½ stage performance is given in other multi-launch charts.

As most of the generic missions to geosynchronous orbit have a return payload nearly equal to the deployed payload, Fig. 3-9 shows the performance levels required. (Mission P3, crew rotation with 30 men, is an indication of the approximate boundary of multi-launch performance when using engines with an I_{sp} of 444 sec and the standard STS.) The APOTV, AMOTV, and AMRV 1½ stage concepts are compared on the basis of their performance in this mode vs the costs per mission. The AMOTV and AMRV are superior but, by their very nature, they are not well suited to the return of bulky or wide-spreading cargoes.

At mid term, phase 1, our findings on the mission modes for multiple STS launches per MOTV missions were:

- With minimum cost engines and standard STS
  - multi-launch per mission provides high performance at high cost per mission (CPM)
  - DDT&E is about $100M less than single-launch candidates ($764M to $894M)
  - 1½ stage configurations, with tanks deorbited, outperform their one stage and two stage ("slingshot") rivals by between 12% and 72%
  - for roundtrip payloads, AMOTV and AMRV outperform APOTV, but the return of bulky/extended payloads is a problem
  - for "mainly deploy" payloads, AMOTV has a performance advantage of about 17%
  - LEO turnaround, working out of the Shuttle, is not cost-beneficial.
- With advanced space engines and augmented STS
  - compared with the minimum cost engine and standard STS approach:
    - o higher payloads, 85%
    - o higher CPM, 20%
  - direct DDT&E delta, $370M
  - evolutionary DDT&E delta, 460M.

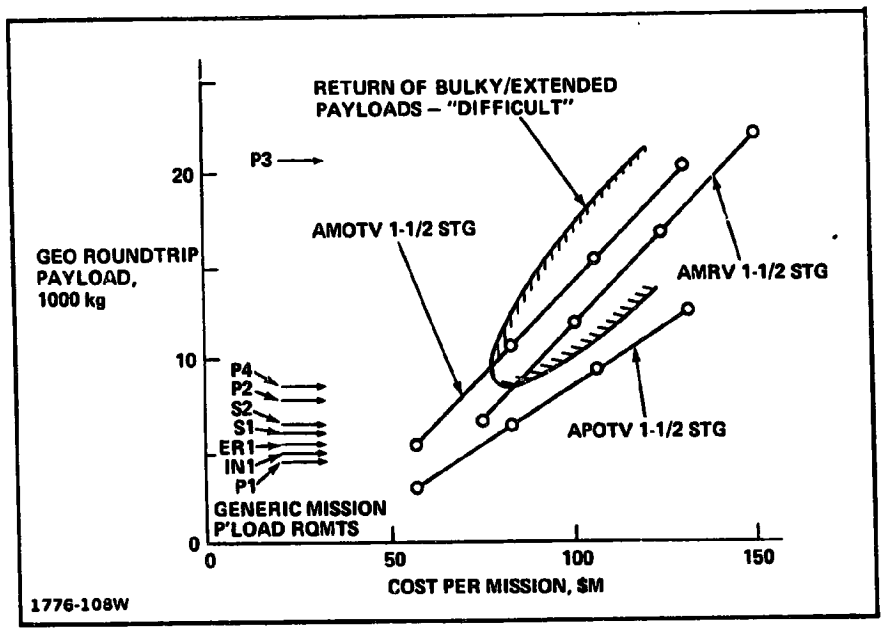

Fig. 3-9 Comparative Performance – "Roundtrip" Missions

- Recommendations
  - consider a "bare-bones" candidate APOTV 1½ stage:
    - o standard STS
    - o RL 10 (2)
    - o DDT&E $767M
  - evaluate evolution to:
    - o advanced space engine/augmented STS, 12,000 Kg roundtrip PL ≈ $99M CPM, or
    - o AMOTV (kit concept), 10,600 Kg roundtrip PL ≈ $83M CPM, or
    - o both, 20,000 Kg roundtrip PL ≈ $100M CPM.

To summarize the mission mode work conducted in the first quarter of the study

• 1½ Stage Concepts (tanks deorbited).	Highest Performance
• AMOTV and AMRV	Highest performance for "roundtrip" payloads, but return of bulky payloads is questionable
• APOTV, AMOTV, and AMRV	Equal performance for "heavy up - light back" payloads
• Single Launch/Mission	Modest performance using augmented STS & high performance engines - low margins
• Multi-Launch/Mission	Ample performance - higher cost per mission
• LEO Turnaround (using STS only)	Not cost beneficial
• LEO Turnaround (using manned depot)	May be worthwhile with eventual heavy traffic.

3.4 APOTV vs AMRV

At the start of the second quarter of phase 1, it was agreed with NASA that AMOTV should be considered an evolutionary concept to be studied as a separate issue. This was mainly because of the necessary development of real-time sensing of upper atmosphere conditions, which are both heterogeneous and constantly changing. Also at this time it was decided, again with NASA concurrence, to drop the single STS

launch per mission mode since it could not perform any of the generic missions, even with augmented STS and high performance engines. The second quarter, therefore, was devoted to studying APOTV and AMRV using multiple STS launches per mission.

Figure 3-10 illustrates the various normal and emergency modes of operation of each type of vehicle. Under normal operating conditions, the APOTV core stage and crew capsule returns to a waiting Shuttle in LEO and then to earth. Alternately, the AMRV capsule returns directly to earth while the core stage is returned by the Shuttle. Under emergency conditions where immediate return is imperative, the APOTV returns to LEO and is retrieved by a backup STS launched especially for this emergency. In the case of the AMRV, immediate return of the capsule to a predesignated landing site on earth is accomplished, but recovery of the core is not recommended, as will be discussed later.

These various modes of operation for the two vehicles, APOTV and AMRV, were evaluated based on their effect on crew and crew capsule safety consideration, overall MOTV performance and complexity of its operation under normal and emergency modes, and finally costs associated with each operating mode.

To determine costs and stack weights, layouts were made of capsules for a two man, three day mission, and a two man, 20 day mission, for both APOTV and AMRV. Figure 3-11 shows the weights breakdown for these capsules after they have been normalized to two crew, five days; it also indicates their shapes. Packaging of the AMRV capsules showed that, due to their shapes and clean lines necessary for entry, the flight deck and work stations were restrictive.

Comparing performance for APOTV and AMRV mission modes when carrying out the S1 mission, the data is shown in Figs. 3-12 and 3-13. These data were current at the time and, although APOTV has since been updated, they suffice for comparison. APOTV total Δ V for the mission is 30,446 fps, including 2% flight performance reserves, whereas AMRV is 33,662 fps. AMRV stack weight is approximately 6% higher than APOTV.

Considering emergency return procedures for the two modes, Fig. 3-10 illustrates that APOTV deorbits to LEO to rendezvous with a loitering STS, whereas AMRV deorbits directly to earth. Figure 3-14 illustrates the procedures associated with these modes for two emergencies situations: retreat from a solar event; and return with ailing crew and subsystems.

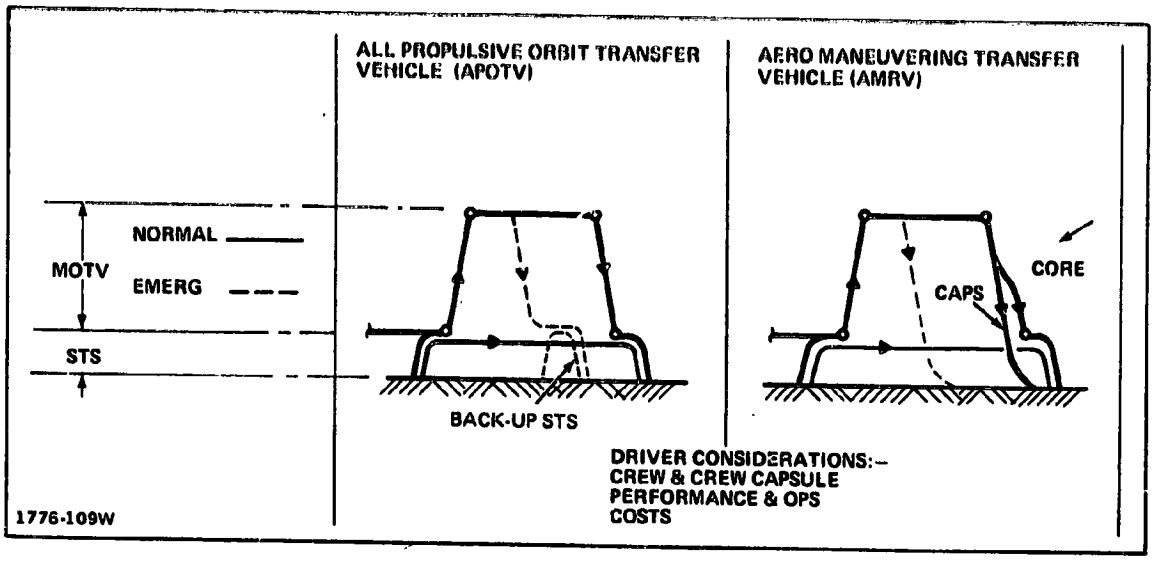

Fig. 3-10 APOTV vs AMRV – Modes & Issues

ENTRY MODE	APOTV		AMRV		
CAPSULE VOL.	10.5 m^3	25 m^3	11.5 m^3	25 m^3	AMRV WEIGHTS DIFFER FROM APOTV WEIGHTS BECAUSE:
STRUCT	856	1628	565	1055	LESS RAD. PROTECT.NEEDED
TPS	–	–	720	1202	ENTRY HEATING
EPS	56	60	76	80	SOLO PWR SUPPLY (4 Kw Hr)
AVIONICS	140	140	255	255	FULL GN&C AND COMM
ECLS	258	274	297	315	ENTRY HEAT SINK
CREW ACCOM	418	704	502	798	HIGH 'G' COUCHES
RCS			90	108	FULL SUBSYST.
RECOVERY			478	638	CHUTES/RETRO SRM/LND GR.
CONTINGENCY	173	281	298	445	
TOTAL DRY	1901	3087	3281	4896	
CREW	163	163	163	163	
CONSUMABLES	95	176	113	196	RCS PROPELLANT
CAPSULE	2159	3426	3557	5255	

1776-110W

Fig. 3-11 Typical Crew Capsule Weights (Kg) (Missions Normalized to 2 Crew x5 Days Duration)

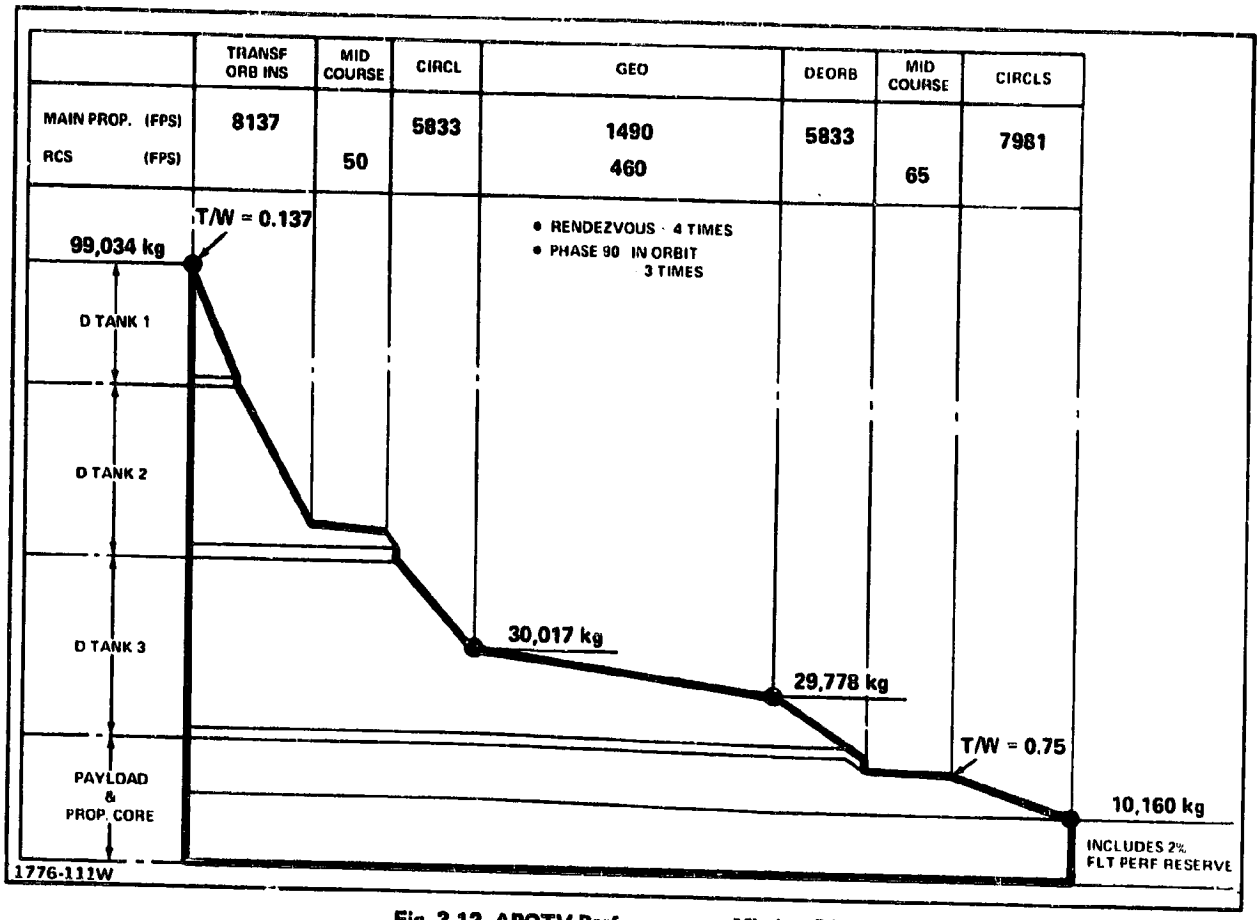

Fig. 3-12 APOTV Performance – Mission S1

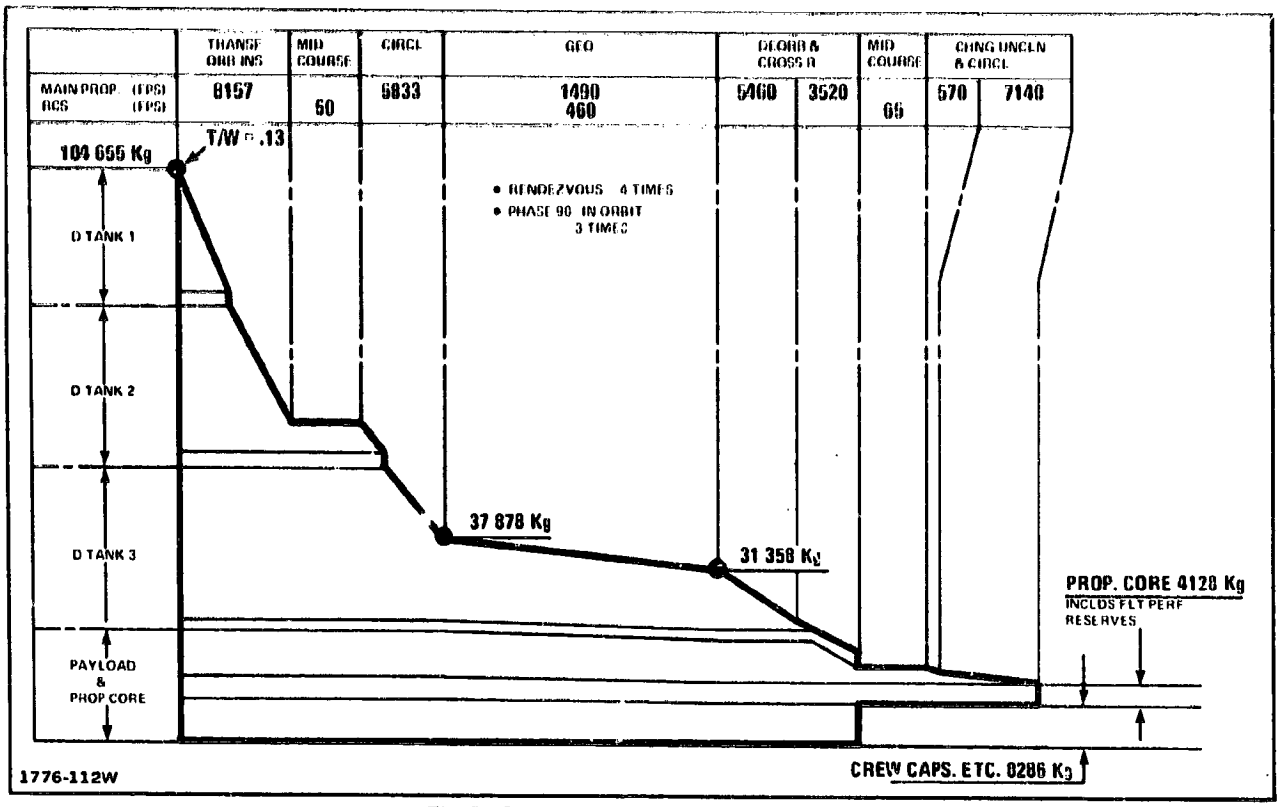

Fig. 3-13 AMRV Performance — Mission S1

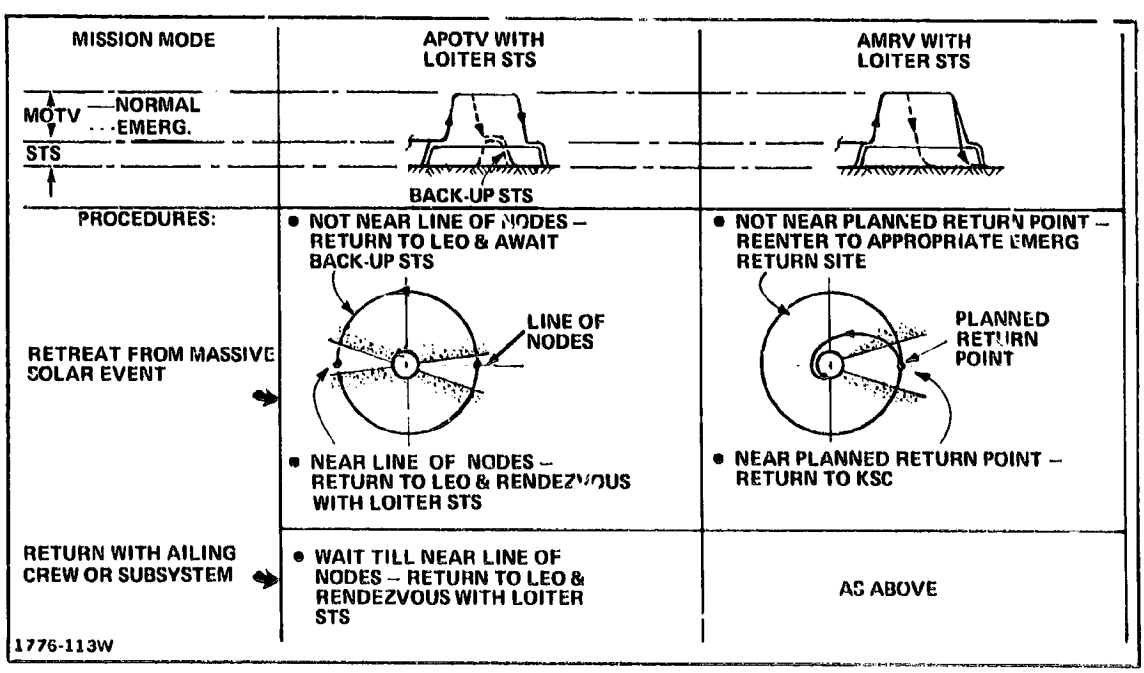

Fig. 3-14 Emergency Return Procedures

Orbits and penalties for operating the APOTV and AMRV are illustrated in Fig. 3-15. One APOTV and two AMRV mission modes are shown at the top; full lines show normal operations, dotted lines indicate emergency, and small circles symbolize the main impulse points. In both AMRV modes, the crew capsule returns to earth Apollo-style. In the center column the propulsion core is recovered by rendezvous with the loiter STS, in the right-hand column the core is expended, burning-up on re-entry.

For normal re-entry, the APOTV uses a conventional Hohmann transfer, the "classic" AMRV uses two burns, deorbit and inclination (with a moderately high perigee), and midcourse slowing to lower the perigee. The full recovery AMRV uses these same burns for the joint capsule/core body, then introduces two more impulses to adjust the core for the STS rendezvous. Under emergency conditions the APOTV follows its normal pattern. The AMRV's have orbits similar to the joint orbits previously described, but with ΔV's changed and total raised to 11,450 fps. There is a performance and cost impact arising from these ΔV requirements, together with the appropriate capsule and core weights. Further costs accrue from the operational penalties noted in the lowest tier of the figure; the loiter Shuttle for two concepts, the stand-by Shuttle for the APOTV, the stand-by launch sites for the AMRV's, and the OTV expenditure cost. These items are allowed for in the comparative costs analysis which follows. AMRV crossrange and downrange ΔV requirements for emergency return are discussed in the next section of this report (Section 4).

Comparing costs for APOTV and AMRV, Fig. 3-16 shows the total cost of development and production for AMRV and APOTV concepts in flying various generic missions. The AMRV costs were, on the average, 15% higher for all cases. Figure 3-17 compares the cost per mission. AMRV costs were, on the average, 10% higher for all cases.

3.5 CONCLUSIONS

To summarize the APOTV vs AMRV trade, the APOTV has a crew capsule with more room in the flight and work stations; it has approximately 6% lower stack weight, less complex return with lower g levels at entry, 13% less development cost, and 9% lower cost per mission. The AMRV gains only on time to return to ground for emergency return: seven hours compared to perhaps 24 hours for APOTV.

The whole mission modes analyses result in the following conclusions:

- Single Launch Per Mission: Can handle smaller missions with modest performance margins

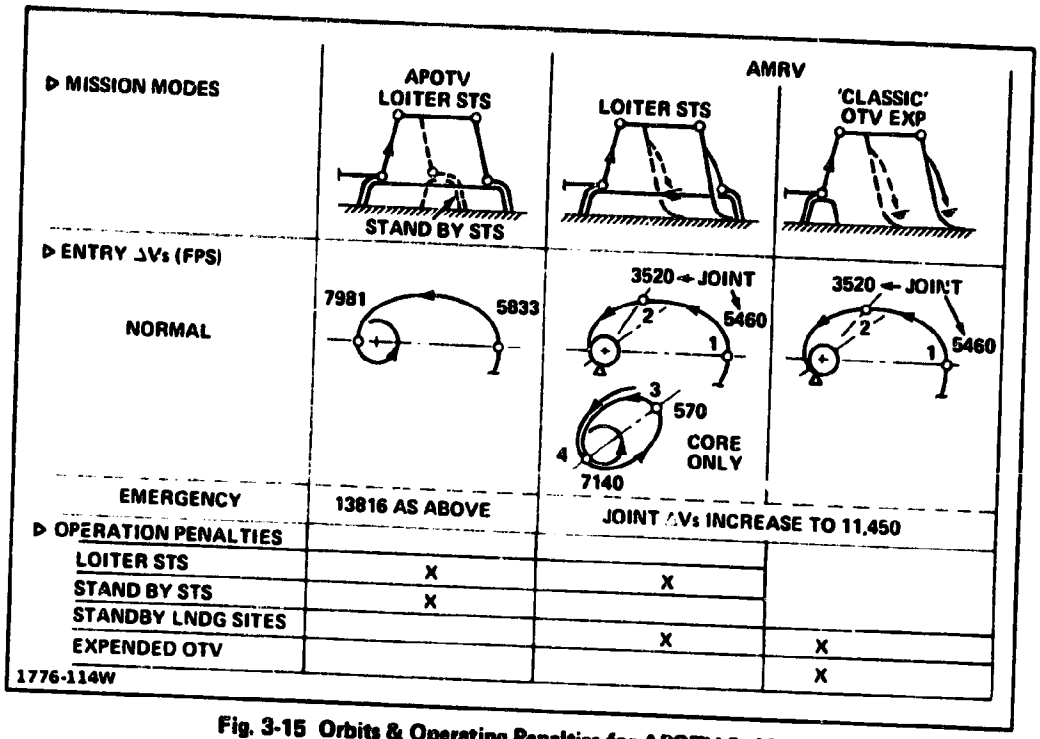

Fig. 3-15 Orbits & Operating Penalties for APOTV & AMRV

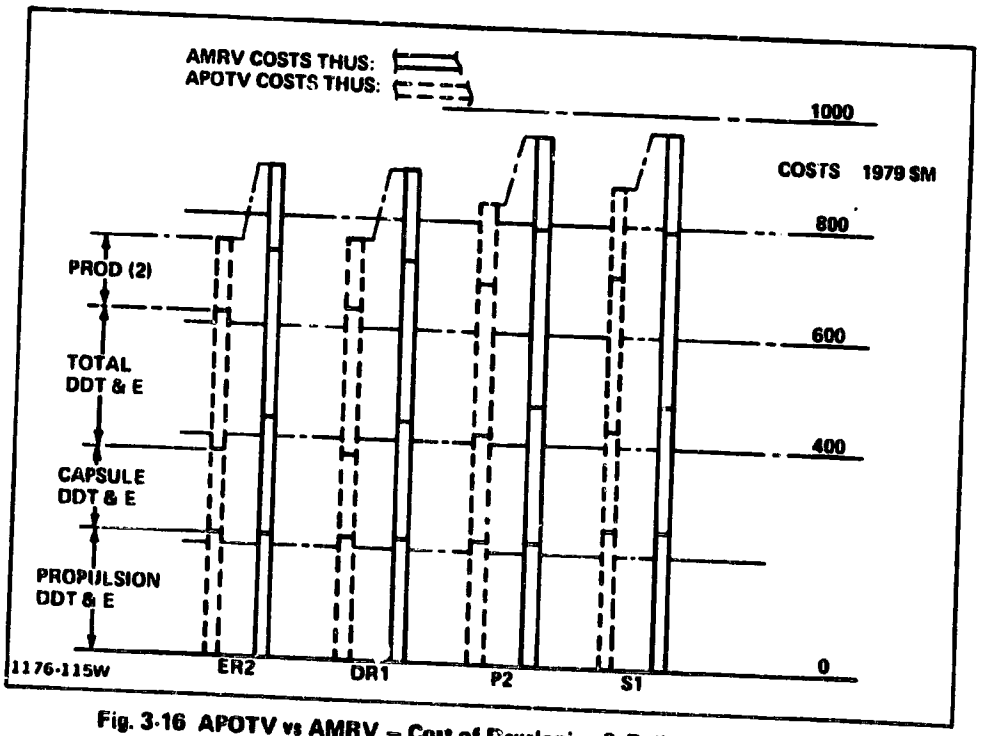

Fig. 3-16 APOTV vs AMRV – Cost of Developing & Building a Small Fleet

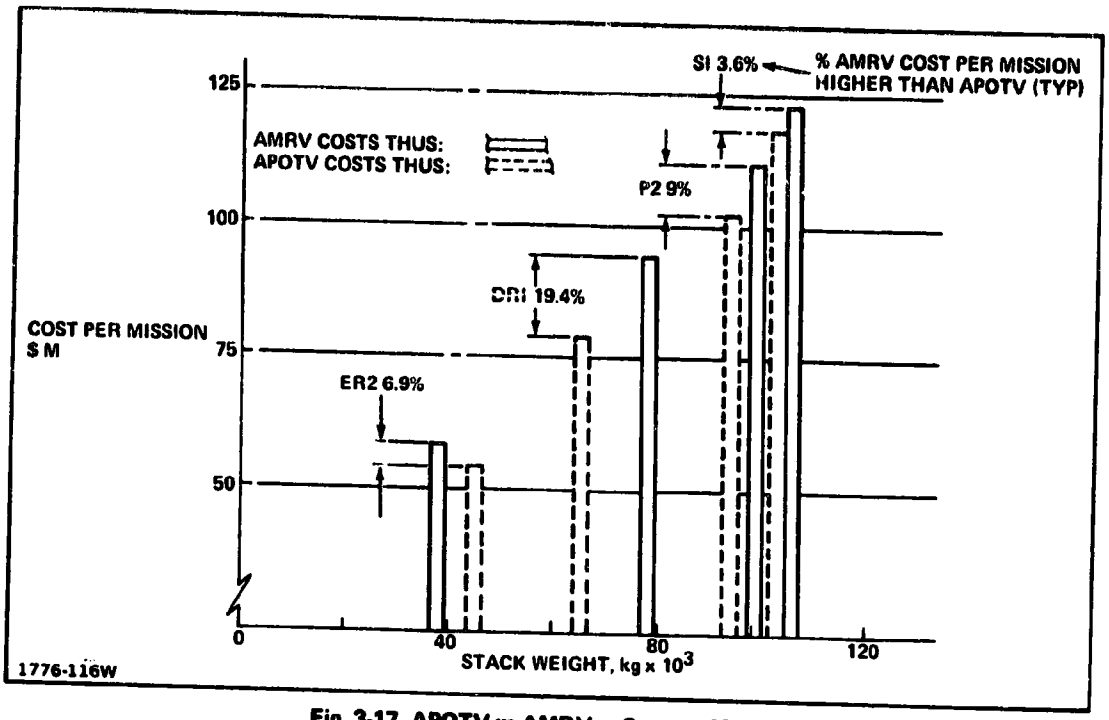

Fig. 3-17 APOTV vs AMRV – Cost per Mission

- Multiple Launch Per Mission: Can handle all identified generic missions
  - with standard shuttle
  - with modular propulsion vehicle with two RL10 type engines
- 1½ Stage Concepts : Have better performance and cost than single or two stages
  - at modest traffic rates
  - using ground turnaround with multi-launch/mission
- 1½ Stage Drop Tanks : Can be deorbited (no space debris) for little or no penalty
- APOTV vs AMRV : APOTV is superior to AMRV on most counts.

We recommended that in phase 2 of the study we concentrate on APOTV 1½ stage, deorbited drop tanks, and multiple STS launches per mission to handle all identified generic missions.

4 - MOTV SAFETY

4.1 REQUIREMENTS

Crew module safety requirements are defined in Fig. 4-1. A good, dependable warning system is necessary to provide timely warning and sufficient data so that proper decisions can be made relative to medical, system, or solar flare emergencies. Several contingency action options are available and are shown in Fig. 4-2. A rescue vehicle (lifeboat) can be stationed at LEO and accessed from GEO in an emergency. Alternatively, the lifeboat can be stationed at GEO so that return would be made directly to earth. A permanent refuge, located at GEO, would act as an independent shelter in case of system failure or medical problems; it could also include protection against solar storms. Another possibility is to provide a short-term solar flare shelter which would not have an independent life support system.

As indicated, none of the individual options provide for all emergencies (solar storms, medical, or system). It would be necessary to combine some of the features to provide additional safeguards for all contingencies; for example, to provide an independent lifeboat habitat plus a storm shelter. In every contingency, the warning time and emergency-related information available is very important and will in the final analysis dictate whether the mission will be aborted or other contingency action taken.

4.2 MOTV EMERGENCY RETURN

Consider the emergency lifeboat return, directly to earth. If the MOTV is near a planned return point, it will return directly to KSC. If not located near a planned return point, the vehicle will return to an appropriate emergency site. This probably requires crossrange and downrange capability. Figure 4-3 plots ΔV's required against crossrange and downrange. Starting from a deorbit station in equitorial geosynchronous orbit, a direct entry vehicle can return to an equitorial landing site about 100° due East with the minimum feasible ΔV of 1434 m/sec (4866 fps).

If, from the same nominal deorbit start point, it is required to reach a landing site further East or West (or to the North or South), more ΔV will be needed. Figure 4-3 shows the results of a preliminary return orbit search, aimed at minimizing the

EMERGENCY	WARNING	CONTINGENCY CAPABILITY
ATMOSPHERE	• CABIN DEPRESS	LEAKAGE MAKE-UP RATE (TBD)
	• FIRE & SMOKE ALARM	FIRE EXTING (TBD)
MEDICAL	• BIOMEDICAL SENSOR WITH GND DATA LINK	HEADACHE/ NAUSEA PILLS PLUS RETURN TO GND
	• RADIATION DOSIMETER	SAME AS ABOVE
SOLAR FLARES	• GND ALERT TIME (TBD)	STORM SHELTER (TBD)
SUBSYSTEM	• MASTER ALARM FOR PRIME SYS	F-O/F-O/F-S
1776-122W		

Fig. 4-1 MOTV Safety Requirements

EMERGENCY	• RETURN TO LEO • TO EARTH IN LEO LIFEBOAT	• DIRECT TO EARTH IN GEO LIFEBOAT	• PERMANENT REFUGE IN GEO • AWAIT RESCUE	• SOLAR FLARE SHELTER IN GEO
• SEVERE SOLAR STORM				1HR WARNING NEEDED
• SEVERE CREW ILLNESS/ ACCIDENT	≤ 20 HR TO EARTH	~ 6 HR TO EARTH		
• SECOND VITAL SYSTEM FAILURE	≤ 20 HR TO EARTH	~ 6 HR TO EARTH	• IMMEDIATE ESCAPE • LONG WAIT FOR RESCUE	1776-123W

Fig. 4-2 MOTV Contingency Options

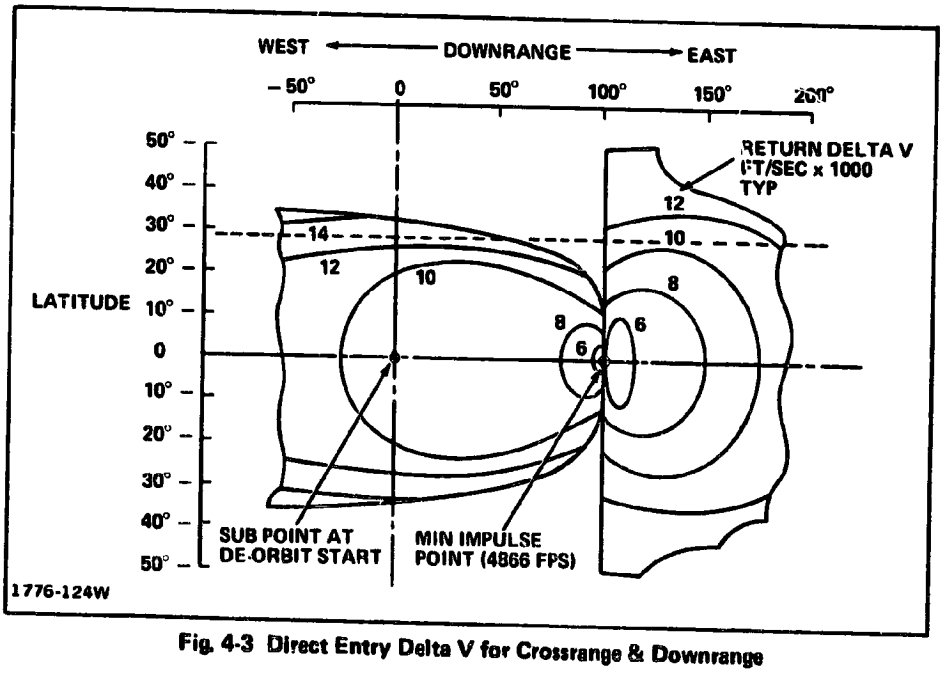

Fig. 4-3 Direct Entry Delta V for Crossrange & Downrange

extra impulse required to achieve combinations of crossrange and up/downrange. Later refinements, used in the AMRV performance and costs calculations, reduced the levels shown; i.e., the minimum ΔV to reach a latitude of 28.5° North is now thought to be 8980 fps rather than the 10,600 fps shown.

As discussed in Section 3, the baseline philosophy for emergency return of the APOTV is to return to LEO in the vehicle and rendezvous with a loitering Shuttle.

4.3 MEASURES AGAINST RADIATION

For this study we used the crew radiation dosage limits permitted for the Shuttle crew rather than OSHA levels, which are over two-orders-of-magnitude more stringent. Figure 4-4 summarizes the current shuttle crew limits. Staying within the permitted 30 day dose level shown, the average rems that a crew member will experience each day in geosynchronous orbit is plotted in Fig. 4-5 as a function of cabin and EVA suit thickness (these thicknesses are of aluminum or its equivalent). The outside environment is taken to be the "normal" for that orbit, i.e., mostly electrons with no heavy solar proton flux present. The cabin/suit internal environment consists of electrons and bremsstrahlung.

As a starting point, Option A shows a cabin and suit of the same thickness, 6 mm of aluminum. Option B shows the effects of thickening the cabin to 8 mm, thereby permitting a thinner suit. By inspection, it is not effective to continue this process further by continuing to increase the aluminum cabin wall thickness, but by adding a thin inner layer of tantalum (Option C), the in-cabin radiation level is driven lower and the suit can be thinner to about the equivalent of 4 mm aluminum.

These options are based on each crew member performing one six-hour EVA per day, clearly a severe assumption. If, for the cabin wall of Option C, only one EVA per mission is performed by each crew member, the suit used can be reduced to 1½ mm equivalent aluminum shielding power. In practice, we expect the eventual suit thickness to lie somewhere towards the lower limit of the 1½ to 4 mm range.

Typically, protection against the electron flux, encountered under normal conditions in GEO, requires a cabin wall of 1.5 cm thickness of aluminum or its equivalent. We propose a layered protection. Outer epoxy tiles are the first barrier to inhibit photon production. The pressure shell is aluminum and, finally, an inner shield of tantalum which is efficient at blocking bremsstrahlung.

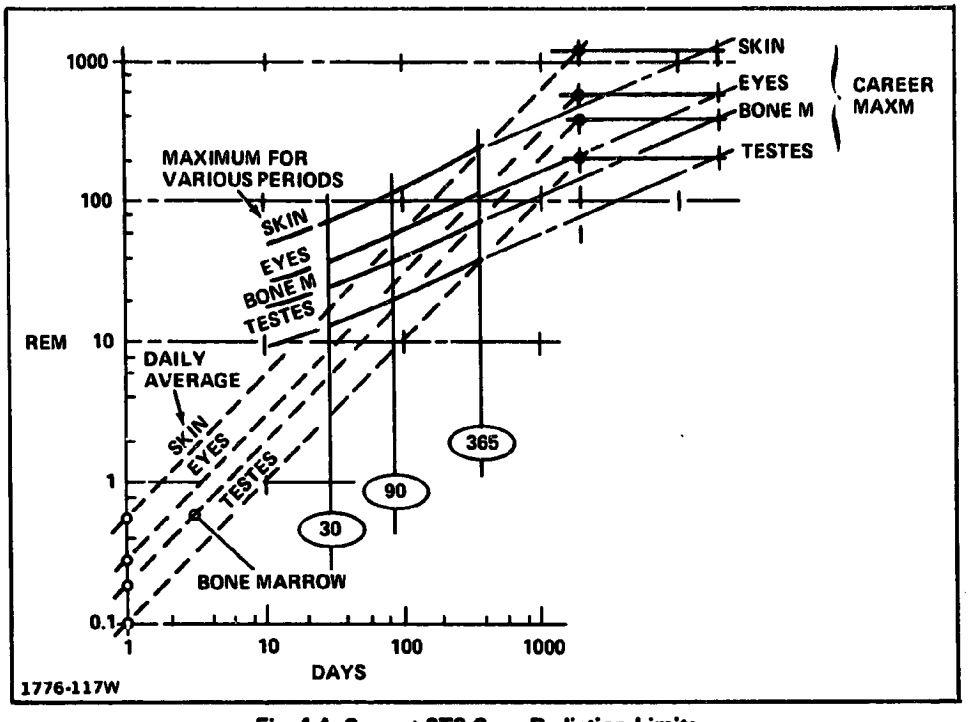

Fig. 4-4 Current STS Crew Radiation Limits

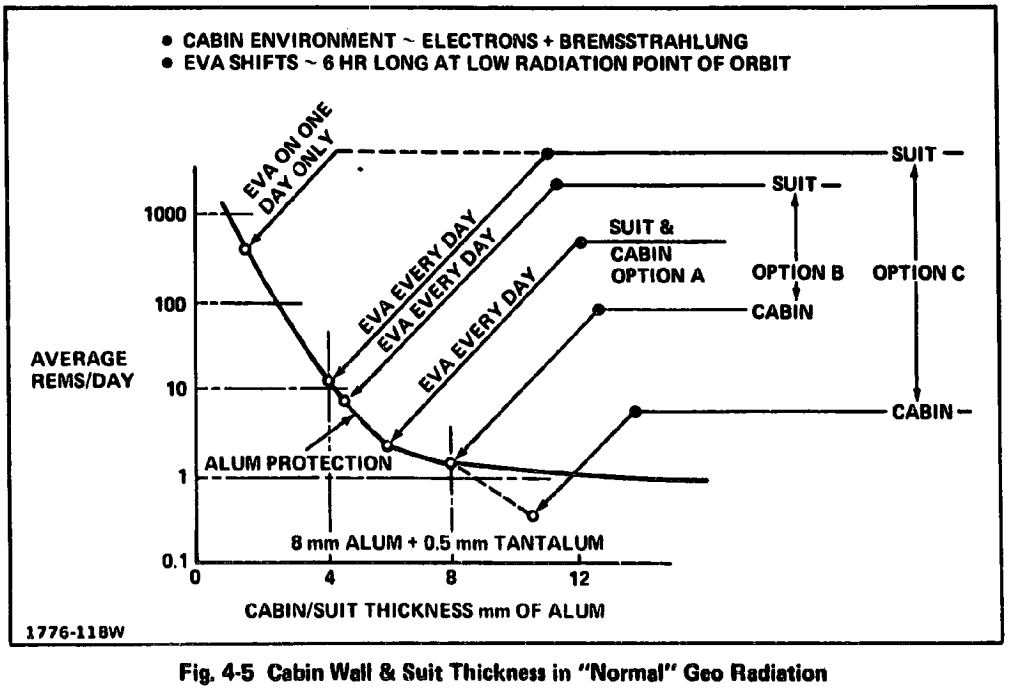

Fig. 4-5 Cabin Wall & Suit Thickness in "Normal" Geo Radiation

Solar storms produce massive proton flux which requires either deorbiting to below three earth radii or shelter from the storm. Figure 4-6 uses cycle 20 to plot solar event duration, in days, against the size of the event. The "size" is taken to be the total flux of protons, above 30 MEV, that falls on each square centimeter during the course of that event. The preliminary conclusion for major events (10^8 p/cm^2 and above) is that the minimum duration is about four days while the maximum is TBD days. If a "stay in orbit sheltering inside the cabin" strategy were to be adopted for missions averaging 10 days in length, this duration could take a significant bite out of the useful mission time.

The frequency of solar events and the necessary protection is plotted in Fig. 4-7. In the upper part of this figure, the frequency of occurrence of solar events is plotted as a function of the size of the event (protons/cm^2). Smoothed historical data are shown for the two most recent solar cycles. Cycle 21 is now underway and resembles cycle 19 rather than cycle 20.

In the lower part of Fig. 4-7 the cabin wall or shelter thickness to protect against this range of event sizes is shown. A typical cabin wall thickness providing protection against "normal" (Van Allen belts only) non-solar event conditions in geosynchronous orbit is shown at about 4 g/cm^2 (i.e., 1.5 cm of aluminum). It can be seen that this gives protection for any event up to about 1 x 10^9 p/cm^2 event flux. Our present strategy is to require the development, by IOC, of a solar event real-time alert system. The system will, at onset of the event, forecast the expected size of that event. We have selected for preliminary evaluation a flux of 1 x 10^8 p/cm^2 as the forecast event size at which the MOTV would retreat from geosynchronous orbit. Taken in conjunction with the typical wall thickness protection, this would provide an order-of-magnitude margin for error in forecasting accuracy.

Considering the time to reach exposure limit as a function of shield thickness, Fig. 4-8 uses the August 1972 Solar Event as a design case. Based on an analysis by Wilson and Denn, this shows the time from the onset of the major flux for various areas of the crew's bodies to reach the exposure limit, as a function of shield weight/thickness. Aside from the eye lens and testes, which can be given extra local protection, the skin is the most vulnerable general area.

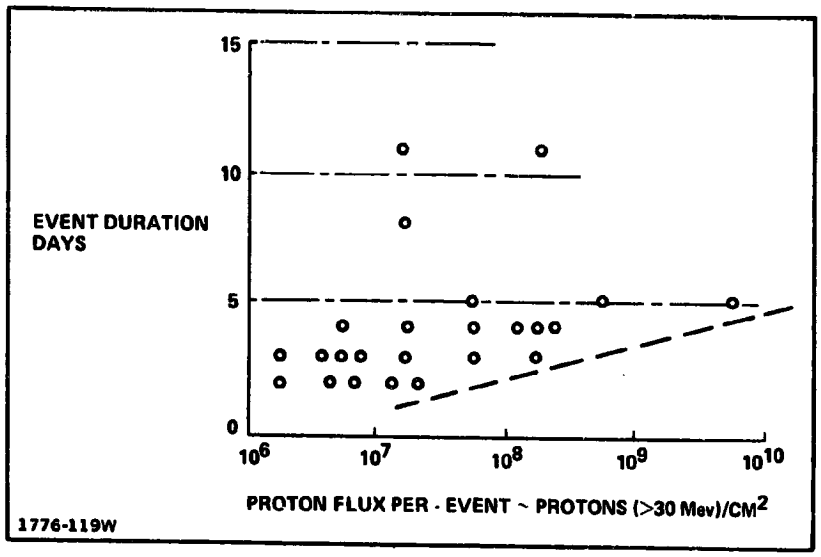

Fig. 4-6 Solar Event Duration vs Event Size (Data From Cycle 20)

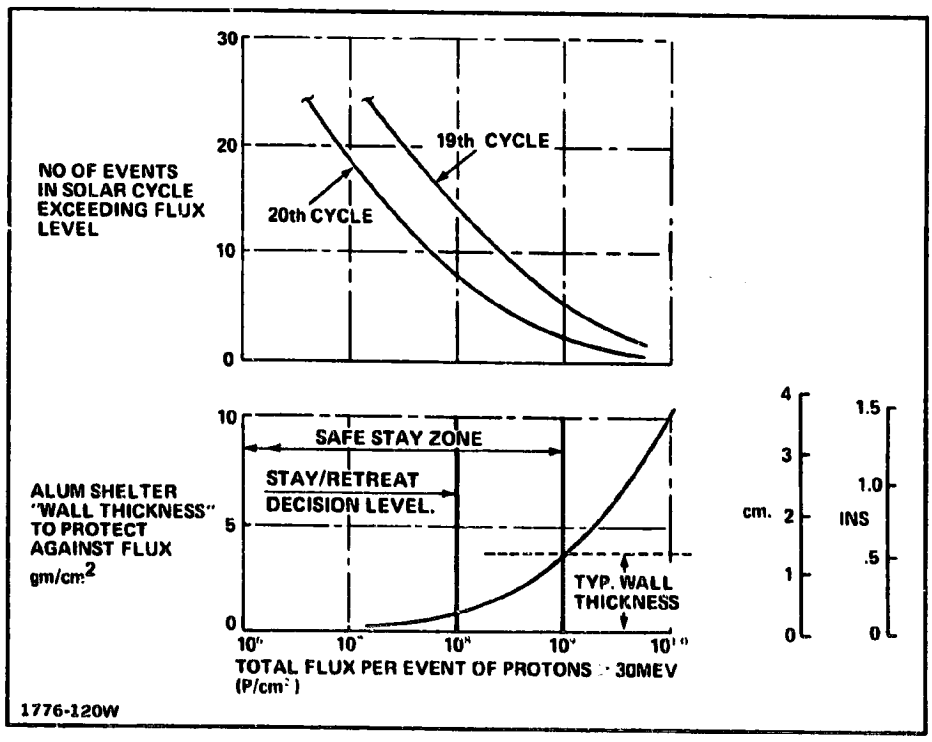

Fig. 4-7 Radiation ~ Solar Event Frequency & Protection

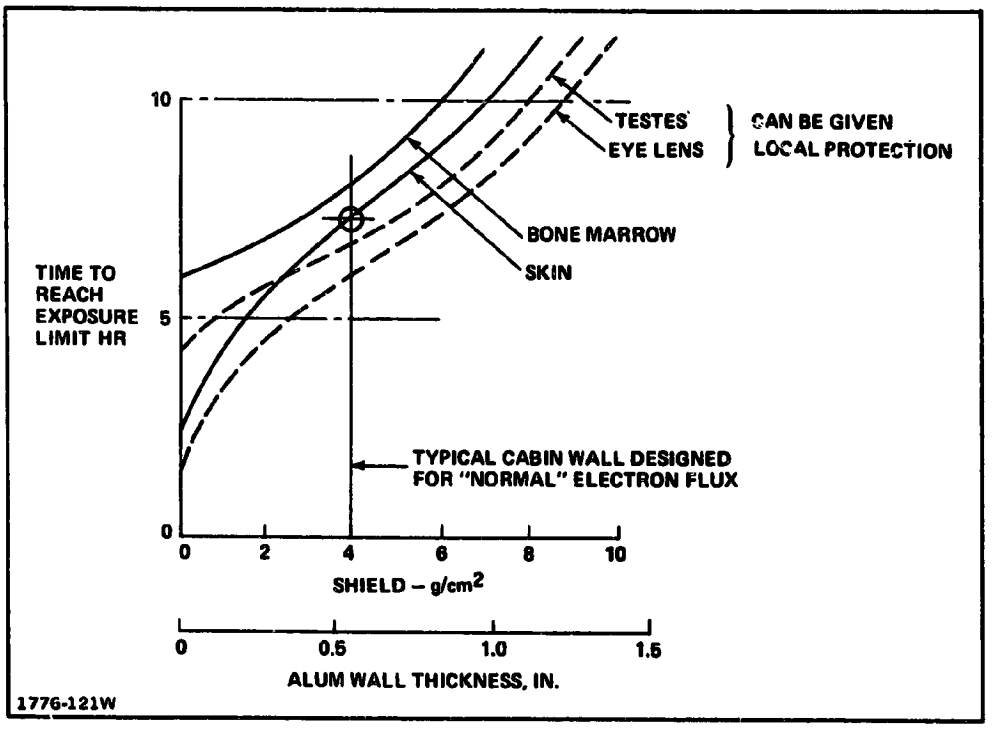

Fig. 4-8 Time to Reach Exposure Limit vs Shield Thickness

If, as previously discussed, the available cabin wall is about 4 g/cm^2, there will be little more than seven hours to get below an altitude of three earth radii, into the zone protected by the earth's magnetosphere. To reach this point in a Hohmann transfer takes about 4.4 hours, leaving just over two hours for the retreat decision to be made, the navigation plans updated, and the deorbit preparations completed.

For less massive events the available time will be greater and 4 g/cm^2 provides adequate protection for an ordinary solar flare (i.e., no more than 1×10^8 protons/cm^2).

In summary, our philosophy is that once the alert of a solar event has been sounded, it is necessary to know the level of flux anticipated and its duration. It would then be determined whether the cabin protection is sufficient to weather the storm or whether it is necessary to deorbit. To this end, we expect a flux forecasting system to be developed before this MOTV becomes operational.

There are two classes of missions where this philosophy cannot apply. Firstly, a deep space crew rotation mission of about 14 days duration each way, where the vehicle would be too far out to retreat to a safe orbit. Secondly, there are those missions with highly elliptic, 12-hour orbits which entail passing through the Van Allen belt four times a day. The deep space mission will, and the 12-hour orbit may, require a heavier cabin wall or a storm shelter.

Solar storm shelters for geospace operations can be fairly massive structures. As indicated in Fig. 4-9, an aluminum structure for four crewmen, providing 1.4 m^3/occupant, would weigh approximately 9000 kg.

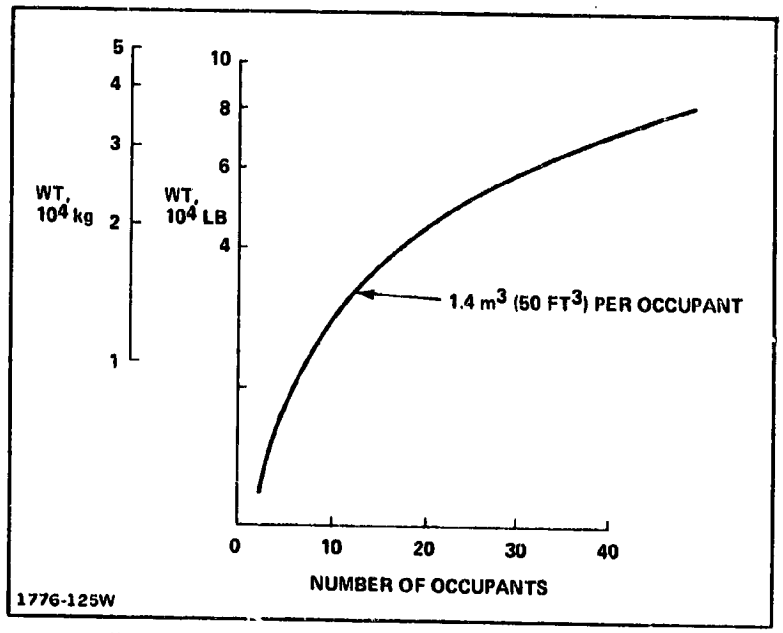

Fig. 4-9 Solar Flare Shelter Aluminum Shield Requirements

5 - IVA vs EVA TO PERFORM MISSIONS

5.1 OPTIONS

The three basic options for performing any mission requiring work external to the crew capsules are, firstly, by all IVA (where the crew remains in the capsule and uses manipulator devices controlled from within the capsule). Secondly, the work is performed mainly by IVA, with EVA used to perform only those tasks which are much more cost effective if done that way. The third option is using EVA to perform all the tasks.

5.2 TRADE

Figure 5-1 shows how weight penalties increase with manhours worked to perform a typical three-man construction mission. The first option, "IVA only," includes the penalties for basic equipment together with requirements for one emergency EVA, assuming atmosphere dump and repressurization of the cabin. The second option assumes that 90% of the mission is performed IVA, leaving 10% EVA. No airlock was considered for the EVA, which means that the cabin atmosphere can be either dumped and repressurized or pumped down and stored. Both of these methods are considered and include the basic and emergency equipments together with the expendables used on each work shift. The third alternative considers EVA as the only method of performing the mission. For this method, we considered not only the penalties for depressurizing the cabin by dump and by pump down but, as an alternate, we included an airlock with atmosphere dump or pump down.

"IVA only" carries the lowest penalty, which is constant irrespective of manhours. This is closely followed by combined IVA/EVA where the penalty for expendables rises very little with manhours worked. It is about the same for cabin atmosphere dump as it is for pump down. The airlock carried a high penalty with "EVA only" and, therefore, was not considered for IVA/EVA. With "EVA only," the cabin pump down system expendables penalty does not increase much with manhours and is little more than IVA or IVA/EVA. The trade was repeated for a two-man inspection/repair mission with similar results.

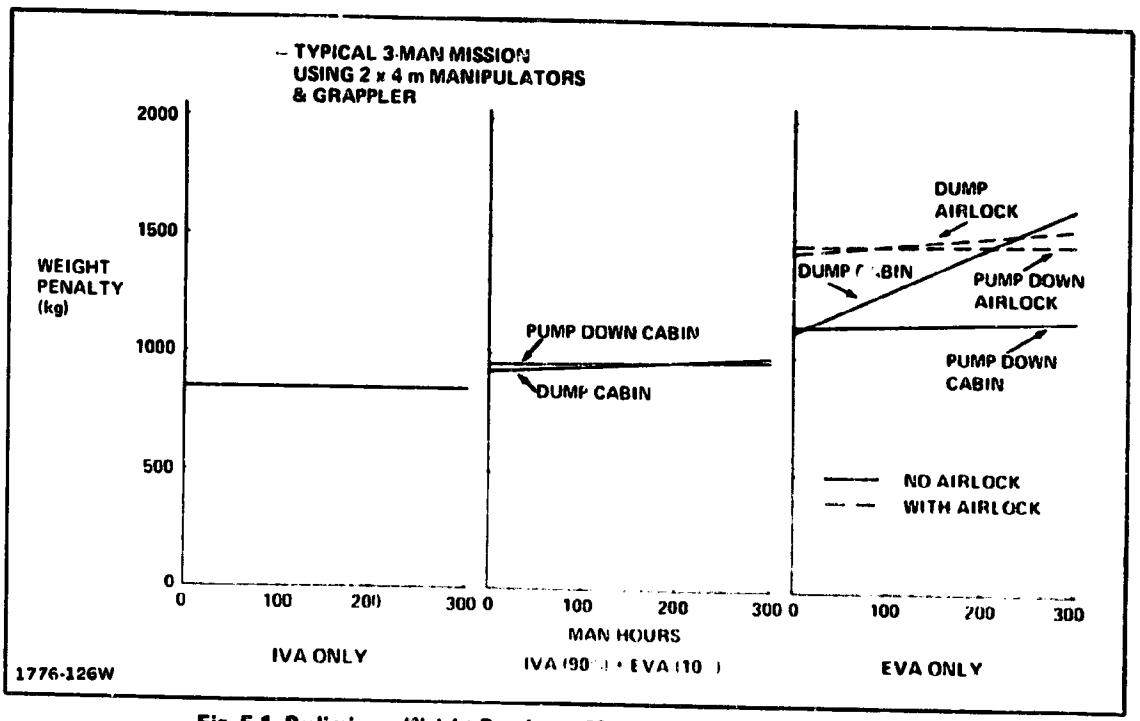

Fig. 5-1 Preliminary Weight Penalty vs Man-Hours for 3-Man Construction Mission

Since the best candidates in all three options use very few expendables with hours worked, this is not a discriminator and productivity must be considered to resolve the trade.

Two types of tasks were considered in the productivity analysis. A service task involving replacement of MMS-type modules on a satellite in GEO, and a construction/repair task. In each case, event times to complete the tasks were compared when performing them via EVA and IVA. EVA operations assume the need for an Open Cherry Picker (OCP) to provide a stable work platform for MMS module replacement. The EVA event times were derived from NASA water tank tests, Skylab data, and MRWS study data. The IVA times are based on Princeton "Tokamak" and Los Alamos experience using the Bilateral Force Reflection (BFR) type manipulator.

The approximate relative efficiencies among several IVA control modes and performing the same tasks EVA, are compared in Fig. 5-2. Among the IVA options, BFR manipulator system is the most efficient. Nonforce Reflecting (NFR) relies only on visual cues for operator feedback. The Resolved Motion Rate Control (RMRC) uses a six-DOF hand controller to establish the direction of motion and tip speed of a slave arm.

All known test results indicate an advantage in task efficiency when using the BFR manipulator system. Performing similar tasks EVA in an Apollo Suit is slightly more efficient, but this suit does not have the radiation protection necessary for GEO EVA operations. It is estimated that this time advantage would largely disappear for a GEO EVA suit. Since the Apollo Suit seems to represent the most efficient EVA system to date, and the BFR the most efficient IVA system, they were selected for performing subsequent trades.

Figure 5-3 shows the results of IVA vs EVA productivity trades. A typical service mission is shown in the left hand bar chart. Notice that the time to don the EVA suit, prepare the cabin for depressurization, egress, unstow and checkout the OCP, and reverse the process (i.e., nonproductive time) is greater than the time it takes to perform the task IVA. Moreover, even during "productive time," the task takes 33% longer EVA and IVA because of the additional time in maneuvering the OCP to and from the MMS module stowage area. The actual task time in module removal and replacement is about equal for both modes of operation. The service task can be accomplished well within one work shift, so the overall saving in mission time is not particularly significant. In the case of a construction/repair task, as shown in the right hand bar chart, the task is on-going through many shifts. Here the efficiency of

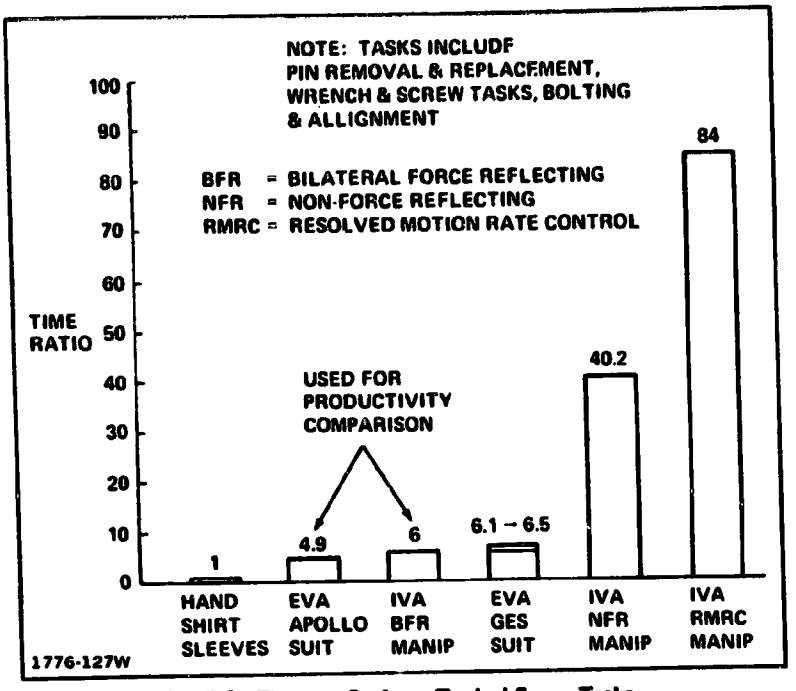

Fig. 5-2 Times to Perform Typical Space Tasks

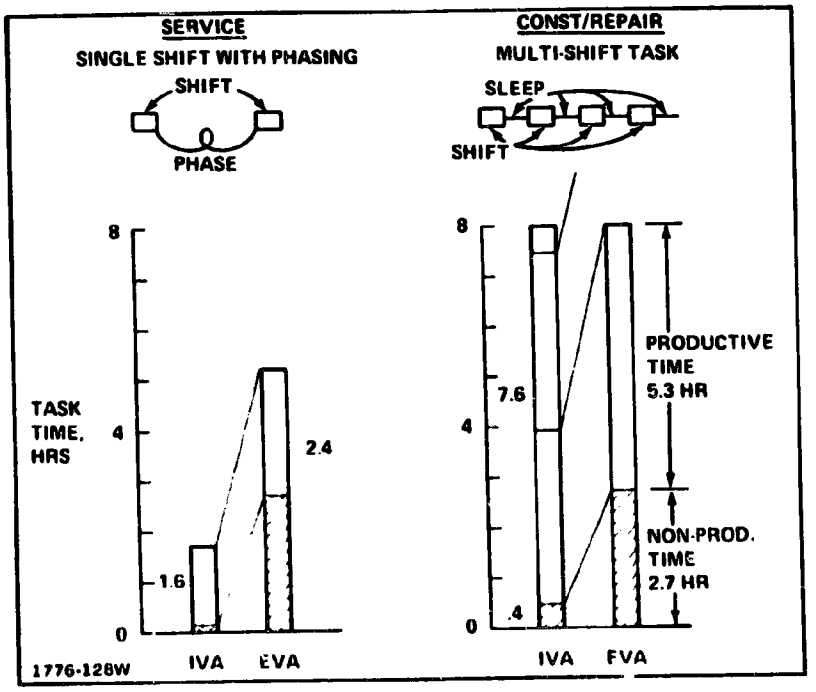

Fig. 5-3 IVA/EVA Productivity Comparison for 2 Types of Tasks

performing the tasks IVA pays substantial dividends since twice as much per shift can be accomplished IVA.

IVA appears to be more productive than EVA, provided there is no impediment to performing these tasks using BFR manipulators and the task is well understood. Under certain contingency operations where the task had not been planned, such as emergency repair of a satellite, EVA may be the only way of performing the task, particularly if the work area is tight. The MOTV may not be able to get close enough to perform the task IVA using manipulators.

It was recommended that, for phase 2 of the study, the preferred method for performing mission tasks would be IVA using BFR manipulators and a stabilizer for berthing. EVA would be used for contingency and emergency operations, with two crewmen outside using GEO Suits tethered to the MOTV.

6 - CONCEPT DEFINITION

6.1 MOTV

During the first quarter of phase 1 of the study, we looked at concepts to cover the mission modes discussed in that section of this report. Manned capsules for these concepts included LM and MRWS derivatives, growth Gemini, Spacelab modules, and a dedicated capsule. These concepts were reported fully at that time in our mid-term, phase 1 report. They are not repeated here since, although of historical interest, the recommendation following that report was that we should concentrate on APOTV with multiple STS launches per mission. Consequently, in the second quarter of phase 1 we looked at APOTV concepts and performance requirements for each of the generic missions. This information has been updated in phase 2 and is given in the Mission Handbook. The overall configuration for each mission did not change, except in detail, and is typified by Fig. 6-1 which is the S1 mission. The propulsion core is common for all missions and has a propellant capacity of 17,500 Kg, which is off loaded as required for a particular mission. Propellant is contained in an aft liquid oxygen tank and in a forward liquid hydrogen tank. Thrust is provided by two RL10 CAT IIB engines. The vehicle is controlled by RCS thrusters mounted in four modules located about the c.g. to provide translation along all three orthogonal axes, together with pitch, yaw, and roll. The electrical power source for the whole vehicle is mounted on the core. It has fuel cells, with their reactants, located between the propellant tanks together with the propulsion subsystems. Radiators to thermally control the fuel cells are mounted on the inter tank skirt. This mission uses solar cells to recharge the fuel cells during non-peak-load periods. The solar cell array is mounted on the propulsion core thrust structure. Other missions with less demanding electrical power do not have a solar array.

The crew is housed in a capsule which is mounted forward of the core. Considering mission equipment, a grappler to berth the MOTV to the satellite is mounted at the forward end of the capsule. This 4 degrees of freedom grappler also moves the MOTV relative to the satellite to enhance operator viewing and to locate a pair of manipulators for handling mission hardware. These manipulators have 7 degrees of freedom and are a bilateral force reflecting type, presently being investigated in our

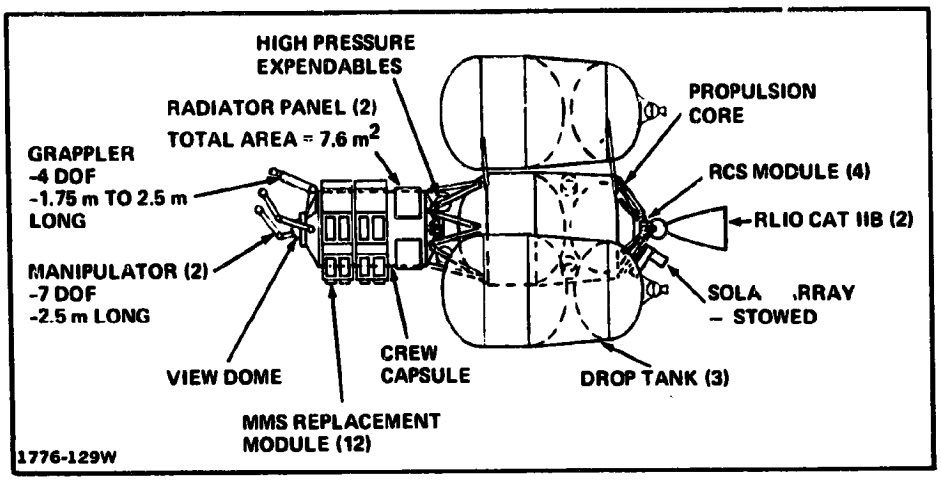

Fig. 6-1 Configuration for Mission S1 at LEO Ignition

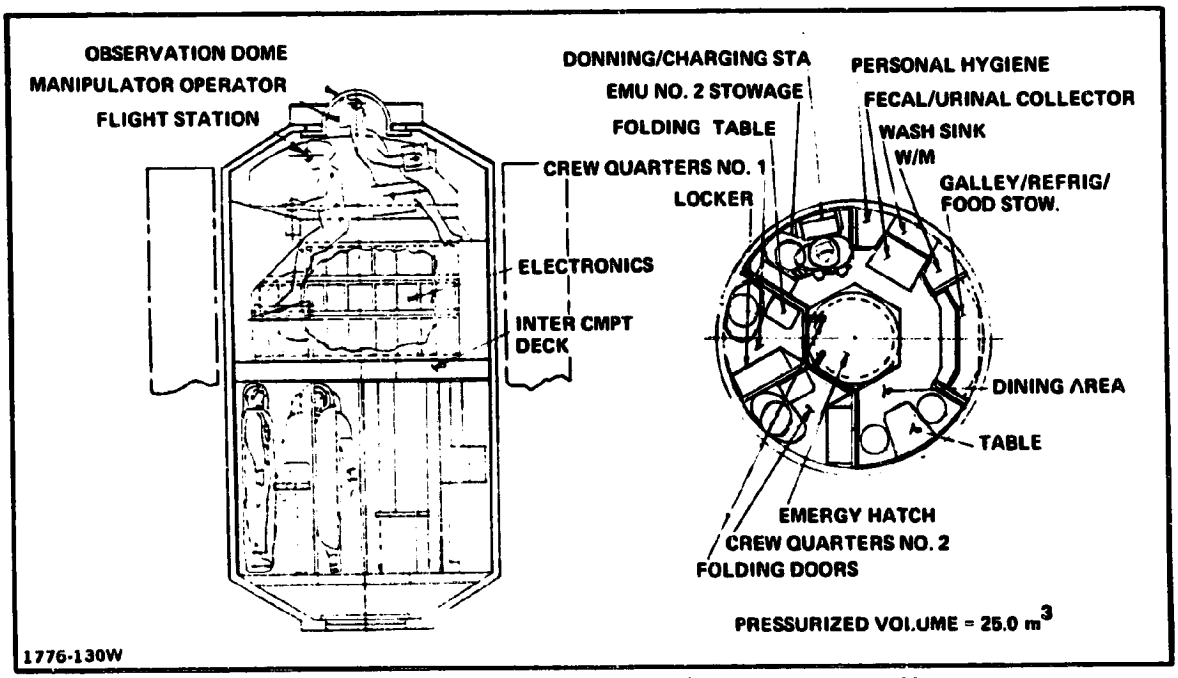

Fig. 6-2 APOTV Crew Capsule (Phase 1) -2 Men for Mission S1

MRWS studies. The manipulators are operated from within the crew capsule by a master/slave system. The S1 mission shown requires three drop tanks; other missions require from one to four tanks.

The crew capsule, derived in phase 1, is shown in Fig. 6-2 as typical for that stage of the study. The capsule is divided into two basic compartment areas; the upper half contains those functions associated with mission performance while the lower half provides living accommodations. Removal of the upper hatch (with observation dome) provides a clear 1 m diameter opening for access into the module. The forward flight station contains controls and instrumentation necessary to maneuver the MOTV. The aft station contains a set of master control arms which operates the slave manipulator arms and the grappler. The lower part of this compartment houses the electronics and ECS hardware. A hole in the inter-compartment deck provides passage to the lower area of the module which contains crew quarters for two people, waste management and personal hygiene, galley/refrigerator/food stowage, dining area, and EMU donning/charging station. Capsule dry weight was 3192 Kg, including 10% contingency. At burn-out, the capsule weighed 3880 Kg.

The drop tank is typical for all missions and it is still currently viable. Figure 6-3 shows the tank in the Shuttle cargo bay and gives some characteristics. Propellant loading of each tank is mission dependent but the tank must be located within the Shuttle cargo bay so that its c.g. is within the allowable envelope. In general, tanks loaded to less than maximum capacity will be positioned as far aft as possible to leave volume for the carrying of another payload. The combined c.g. of the total orbiter payloads must remain within the allowable envelope. Since the tank is about the maximum diameter to fit in the bay, local fittings will be provided to mount it directly to the shuttle pick-up points. It will not, therefore, penalize the Shuttle useful payload capability by carrying a cradle to mount it. This means that the tank must be deployed directly by the Shuttle RMS. The drop tank has a propellant capacity of 27,270 Kg contained in a forward LH_2 tank and an aft LO_2 tank. This capacity diminishes with boil-off as the tank remains in LEO, waiting for MOTV assembly of drop tanks to propulsion core. The rate at which usable propellant diminishes with time can be seen from the plot of "days parked in LEO" vs "max usable propellant." The tank is disposed of by being deorbited by a motor mounted to the LO_2 tank. It is spin-stabilized by jets in the inter-tank skirt. Nutation dampers are also mounted there.

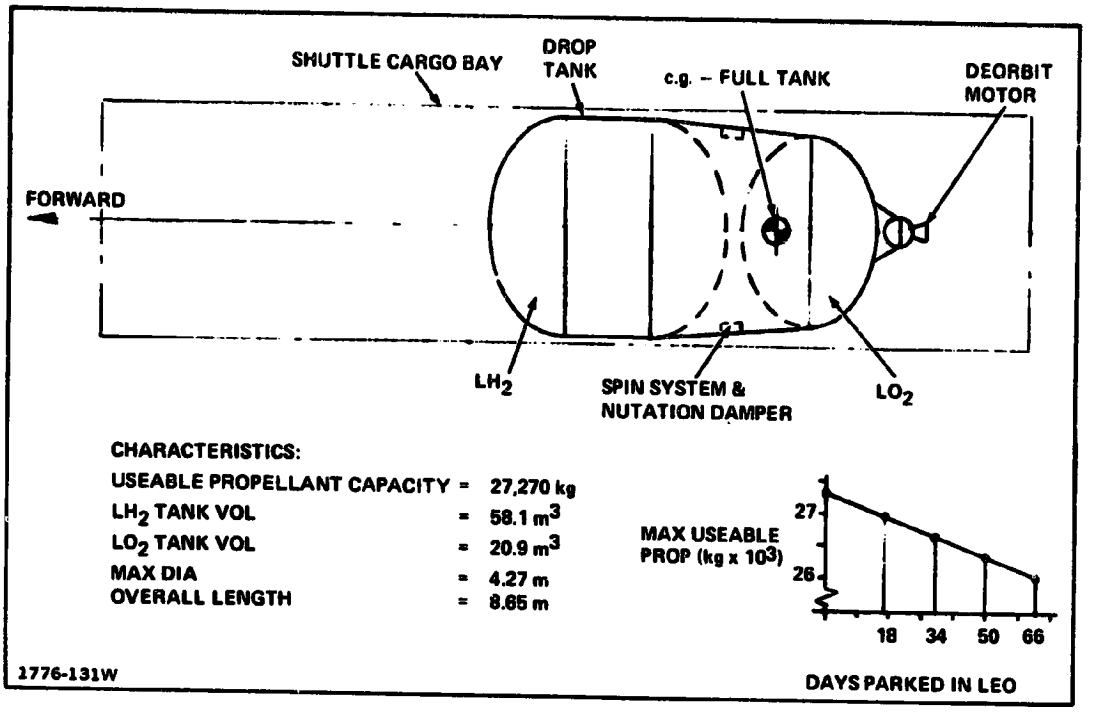

Fig. 6-3 Drop Tank: STS Launch Configuration

• MAIN PROPELLANT CAPACITY	= 17,500 kg
– LH_2 TANK VOLUME	= 36.36 m^3
– LO_2 TANK VOLUME	= 13.62 m^3
• MAXIMUM DIAMETER	= 4.27 m
• OVERALL LENGTH	= 9.8 m
• MAX BOILOFF RATE/TANK	= 19.0 kg/DAY
• MAIN ENGINE (2)	= RLIO CAT IIB
– THRUST LEVEL EACH	= 66,700 N
– I_{SP} (EFFECTIVE)	= 458 SEC AT 6.0 MR
– GIMBAL ANGLE	= ± 20°
• RCS PROPELLANT CAPACITY	= 2600 kg
– THRUSTERS (IN 4 MODULES)	= HYDRAZINE
– THRUST LEVEL EACH	= 700 N
– I_{SP} (EFFECTIVE)	= 230 SEC
• MOUNTED INSIDE OR OUTSIDE INTERTANK SKIRT	
– FUEL CELL (3)	
– REACTANTS	
– RADIATOR	= 4.53 m^2
– SOLAR ARRAY (IF REQD)	= 12 kW_e
– OTHER SUBSYSTEMS	
– INTERFACE ELECTRONICS	

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Fig. 6-4 Propulsion Core Characteristics

At the start of phase 2, study of the propulsion system was awarded to other contractors while Grumman concentrated on the crew capsule. For these propulsion studies, we provided a Design Reference Mission (DRM) as a baseline. This was the S1 mission which is discussed, as appropriate, in other sections of this report and, in more detail, in the Missions Handbook. That handbook shows the MOTV configuration for that mission which changed very little from the S1 configuration discussed earlier in this section. Figure 6-4 defines the main characteristics of the core which has been standardized for all missions. It carries all equipments listed, except for the solar array which is carried only on the longer duration missions to rechange fuel cells during non-peak-load periods.

6.2 CREW CAPSULE

To develop the crew capsule, we started with the phase 1 capsule (Fig. 6-2) and looked at reworking the flight and mission stations. Candidate layouts were drawn using the following assumptions:

- manipulators will be the bilateral force reflecting type with either "low gear" or "high gear" master
- there will be a grappler which will not have a separate control station
- a minimum crew of two will be on duty when the work station is in operation
- the crew capsule is 3 m diameter with the work station at its forward end; it must fit into the STS cargo bay.

Five alternates were drawn as shown in Figs. 6-5 through 6-9. In Fig. 6-5, the two crew on duty are stationed side-by-side to operate a pilot station, which doubles for grappler operation, and a station for the mission operator. The "low gear" manipulator necessitates a bubble on the capsule to allow for the operator's arm movement. Figure 6-6 has a one-man operating station, which accommodates pilot, manipulator, and grappler functions. The second man is in tandem with the first and monitors his activities via TV screens, displays, etc. He also acts as emergency back-up for the prime operator. In this configuration, the front of the capsule can be tapered to allow less restricted entry into the work area. Figure 6-7 reverted to the side-by-side crew, but the manipulator operator has the less volume-demanding "high gear" master. This also allows for a tapered front end to the capsule. Continuing the desire to taper the capsule for better entry to the work area, Fig. 6-8 combines the pilot "high gear" master manipulator and the grappler operations into one station. Again, the second man is monitoring and back-up for the first. To taper the capsule even further, the

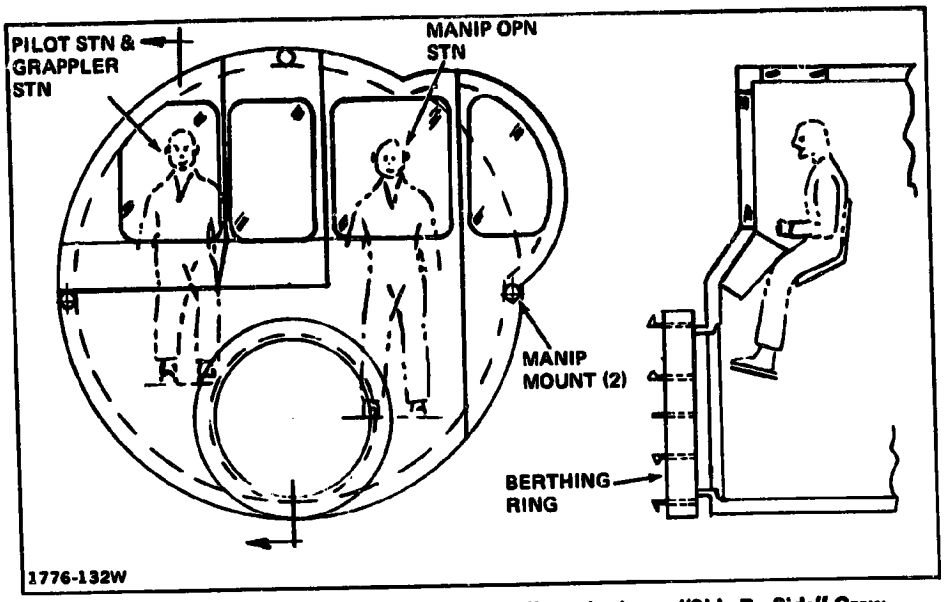

Fig. 6-5 Work Station – "Low Gear Master" Manipulator, "Side-By-Side" Crew

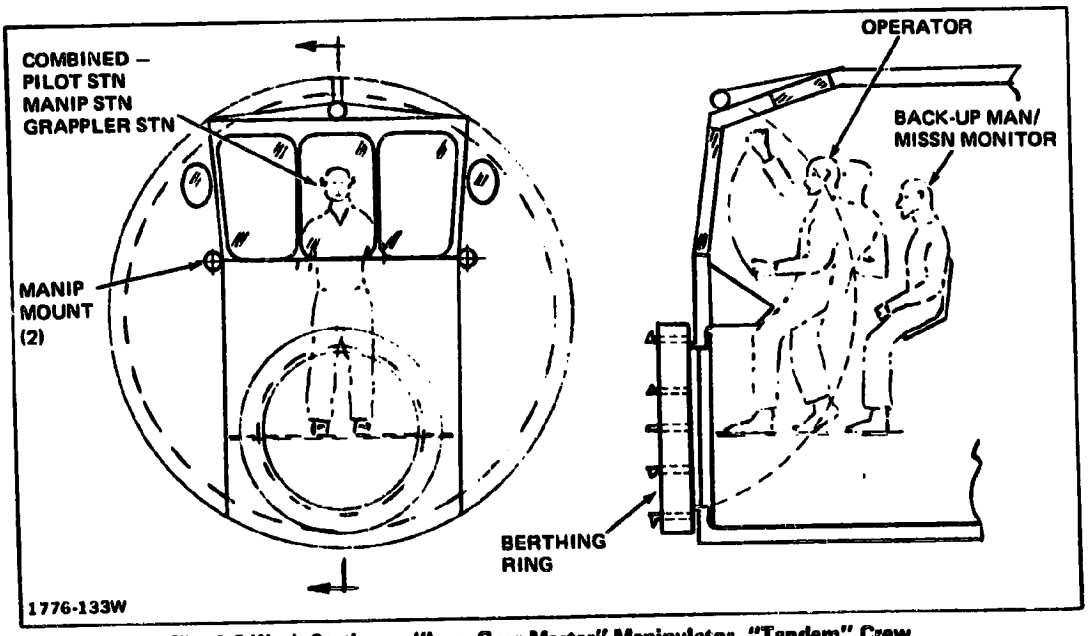

Fig. 6-6 Work Station – "Low Gear Master" Manipulator, "Tandem" Crew

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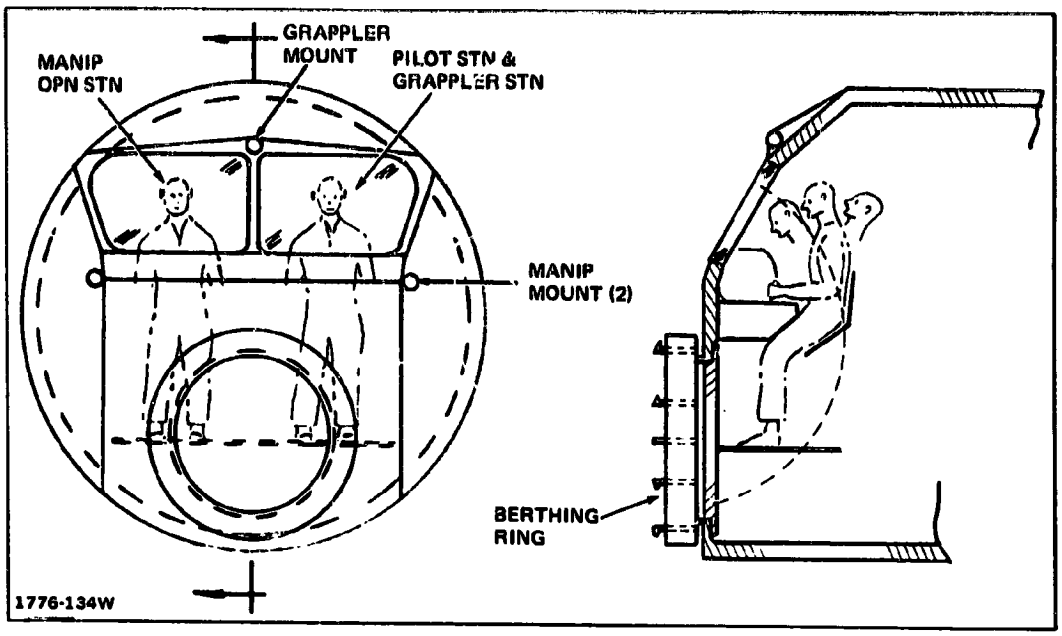

Fig. 6-7 Work Station -- "High Gear Master" Manipulator, "Side-By-Side" Crew

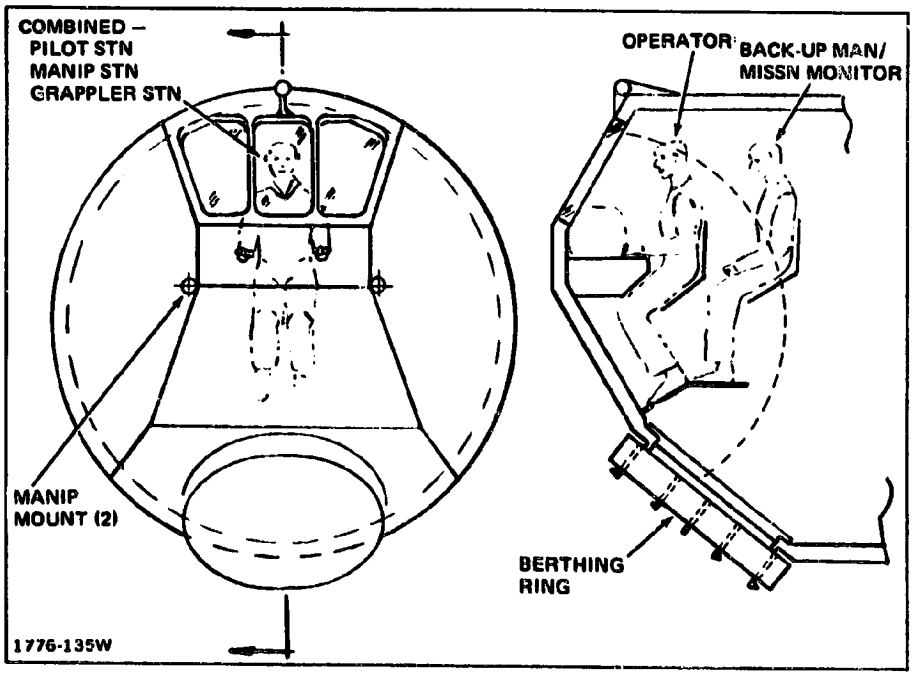

Fig. 6-8 Work Station – "High Gear Master" Manip, "Tandem" Crew

berthing ring is located on a sloping surface, as shown. In an all out attempt to make the capsule nose as pointed as possible, Fig. 6-9 went to a "bomber aeroplane" pilot and nose bomb aimer relationship for the two crew. The berthing ring is on the side of the cabin.

These five alternates were traded and ranked by the awarding of points for various criteria, as shown in the table in Fig. 6-10. Each criterion was considered and awarded points from 1 to 5, with the lowest being the most preferred. Figure 6-7 had the best score and was selected as the baseline work station for further studies of the crew capsule.

As described in the Commonality and Sensitivities section (Section 8), we traded off two alternative internal layouts of the crew capsule. One was based on our phase 1 capsule (Fig. 6-2) with a transverse deck, while the other had a longitudinal deck. The preferred configuration is that shown in Fig. 6-11 and 6-12. This is a single-deck arrangement with a common floor throughout the capsule. Each crew member has a compartment with storage for personal belongings. The crew members are depicted "off duty," sitting on seats which can be stowed at the compartment ceiling. Curtains can be drawn to provide privacy from other crew members. To sleep, the crewman lays "fore and aft" with his legs and feet in a box which demands space in the next aft compartment. A galley and food storage are provided. There is a combined personal hygiene and waste management compartment which occupies the aft end of the capsule. Folded doors may be closed for privacy. Volume between the hygiene and waste management facilities provides the free volume called for in the 1/30/76 final report of the Orbiter Habitability Study, NAS9-14686. Stowage for two EVA suits is provided in this compartment, and the volume necessary for donning a suit is available in the compartment free volume. An emergency hatch is provided in the "roof." Subsystems are stowed over the length of the capsule, under the floor, and above the ceiling. This layout gives a reasonable feeling of spaciousness at eye level from most areas of the capsule.

6.3 ON-ORBIT ASSEMBLY

Each mission studied to date requires a number of propellant drop tanks to be added to the propulsion core in LEO. The question arises as to the sequence in which these components of the MOTV propulsion should be launched and mated. There are three options available for this sequence: tanks first, core last; core first, tanks last; some tanks, core, rest of the tanks. The first two options were traded to determine the preferred sequence. The third option was considered but not pursued

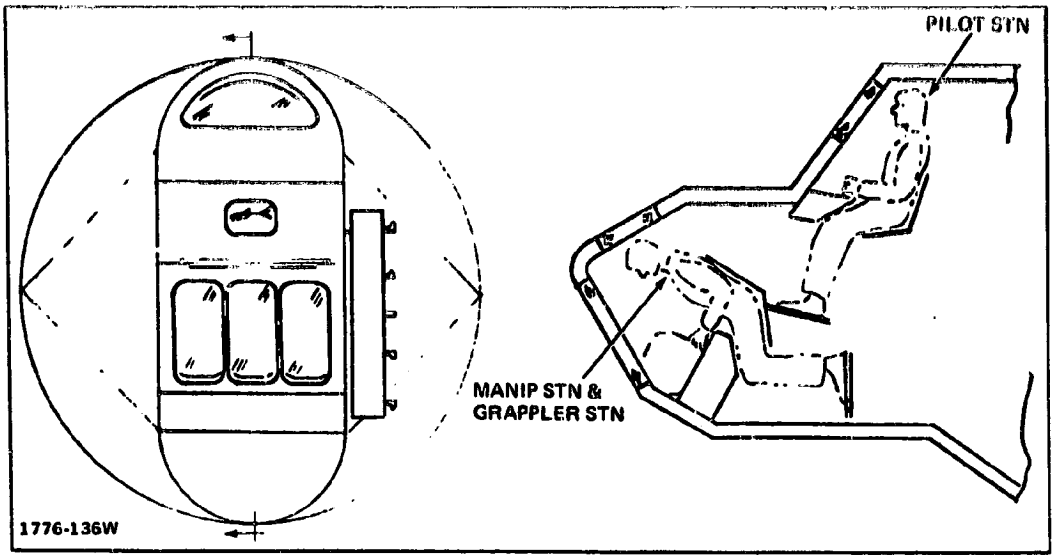

Fig. 6-9 Work Station – "High Gear Master" Manip, "Offset Tandem" Crew

CRITERIA	CONFIGURATION ALTERNATE				
	FIG. 6-5	FIG. 6-6	FIG. 6-7	FIG. 6-8	FIG. 6-9
WEIGHT/WETTED AREA/MAJOR INDENTS.	4	3	1.5	1.5	5
PACKAGING EFFICIENCY/LOST VOL	1.5	4.5	1.5	4.5	3
OVERALL LENGTH	2	4	1	3	5
NOSE PENETRATION INTO WORK PIECE	5	3.5	3.5	2	1
CREW WORKING RELATIONSHIP	1.5	3.5	1.5	3.5	5
VISIBILITY SECTORS	1.5	3.5	1.5	3.5	5
DISPLAYS AREA/SIMPLICITY	1	3.5	2	3.5	5
ACCESS CABIN TO FLT STN & STN SWAPPING	2.5	2.5	2.5	2.5	5
CARGO BAY TUNNEL I/F & GRND C/O	2.5	2.5	2.5	2.5	5
BERTHING MANEUVER	2.0	2.0	2.0	4.0	5
1776-138W	23.5	29.5	19.5	30.5	44.0

Fig. 6-10 Work Station Configuration Options Ranking (By Points Allocation)

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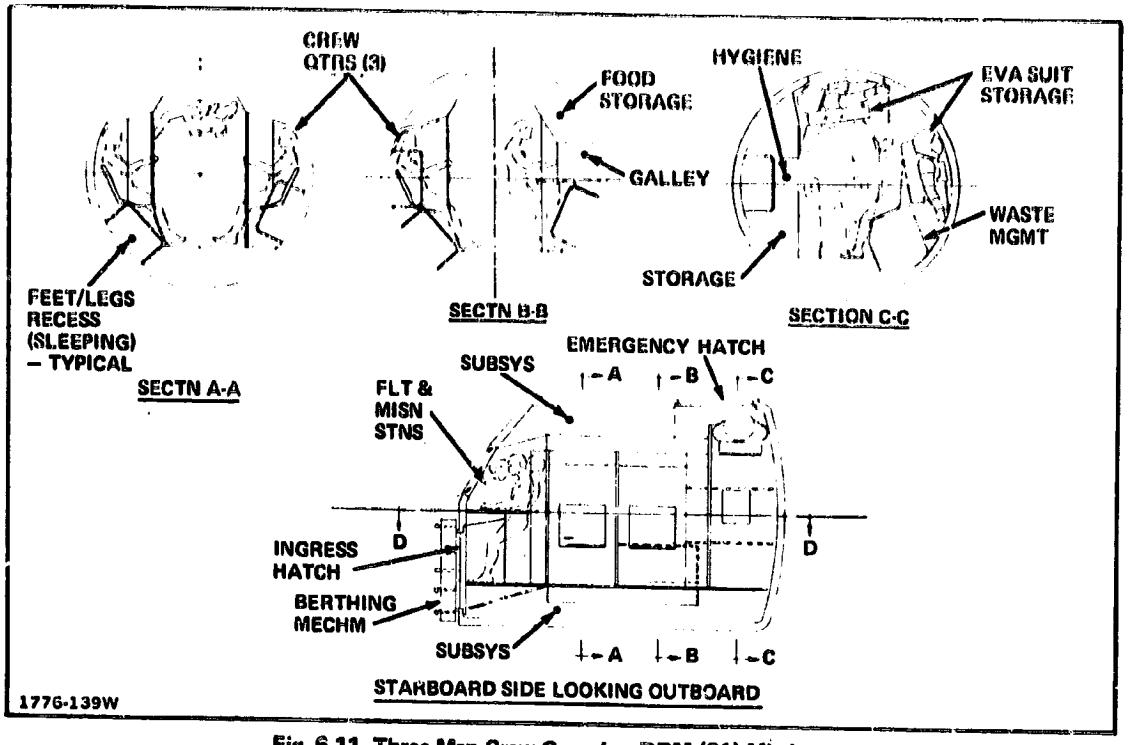

Fig. 6-11 Three-Man Crew Capsule: DRM (S1) Mission

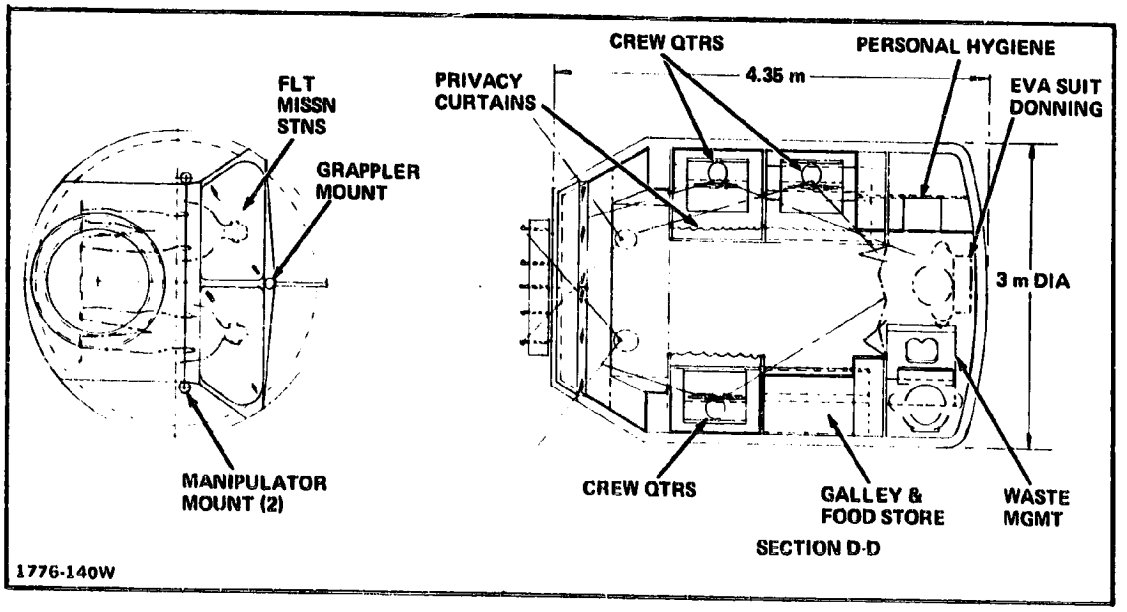

Fig. 6-12 Three-Man Crew Capsule: DRM (S1) Mission (Contd)

since it seemed to offer no benefit. Figure 6-13 shows the core first, tanks last sequence. First, the capsule/core assembly is launched in the shuttle. The orbiter RMS deploys the assembly and leaves it in orbit. The orbiter cargo c.g. allowable envelope is reproduced with the capsule/core combined c.g. at launch and deorbit, spotted on it. They fall within the envelope. For each drop tank except the last, an STS launch transfers the tank to LEO, rendezvous with the orbiting capsule/core, captures it using the RMS, then berths it to a jury rig pad in the orbiter cargo bay. The new tank is then transferred to the capsule/core by the Shuttle RMS, and located for mating. An EVA crew from the orbiter performs the final attachment routine. The orbiter then separates, leaving the capsule/core to stabilize the partially assembled MOTV in orbit. At the final STS launch, the last drop tank is orbited and the MOTV crew rides in the orbiter cabin. The same routine occurs as just described, except that the capsule/core is berthed to a pad mounted on a tunnel from the orbiter cabin. After attachment and check out of the drop tank, the MOTV crew transfers in shirt-sleeves through the tunnel to the MOTV capsule. Final check-out is performed and the Shuttle separates.

The tanks first, core last sequence is similar to that just described but carries some penalties as is shown in the comparison in Fig. 6-14, where the two options are illustrated with mission mode sketches and the scenario for each launch and mating. Considering the weight penalties for each, there is a direct reduction in orbiter payload capability due to the deploy and rendezvous launches to 200 n mi orbit, and the dedicated hardware necessary for a particular event. The selected assembly sequence is core/capsule first, tanks last. There is a weight benefit and, in addition, each tank is mounted directly to the core at each launch which greatly simplifies on-orbit assembly of the MOTV.

A typical timeline for this assembly sequence and the S1 mission is shown in Fig. 6-15. Assembly of the MOTV is accomplished over the first 52 days.

One Shuttle vehicle is used for the entire operation. On the last flight, the MOTV crew is orbited and transferred to the crew capsule and the vehicle is finally assembled, checked out, and readied for the service mission to GEO. All drop tanks are deorbited safely to earth after their propellants are expended, leaving no space debris from their mission. The Shuttle is kept in a loiter mode for the 19-day duration of the MOTV mission, and returns the crew capsule and core stage back to earth.

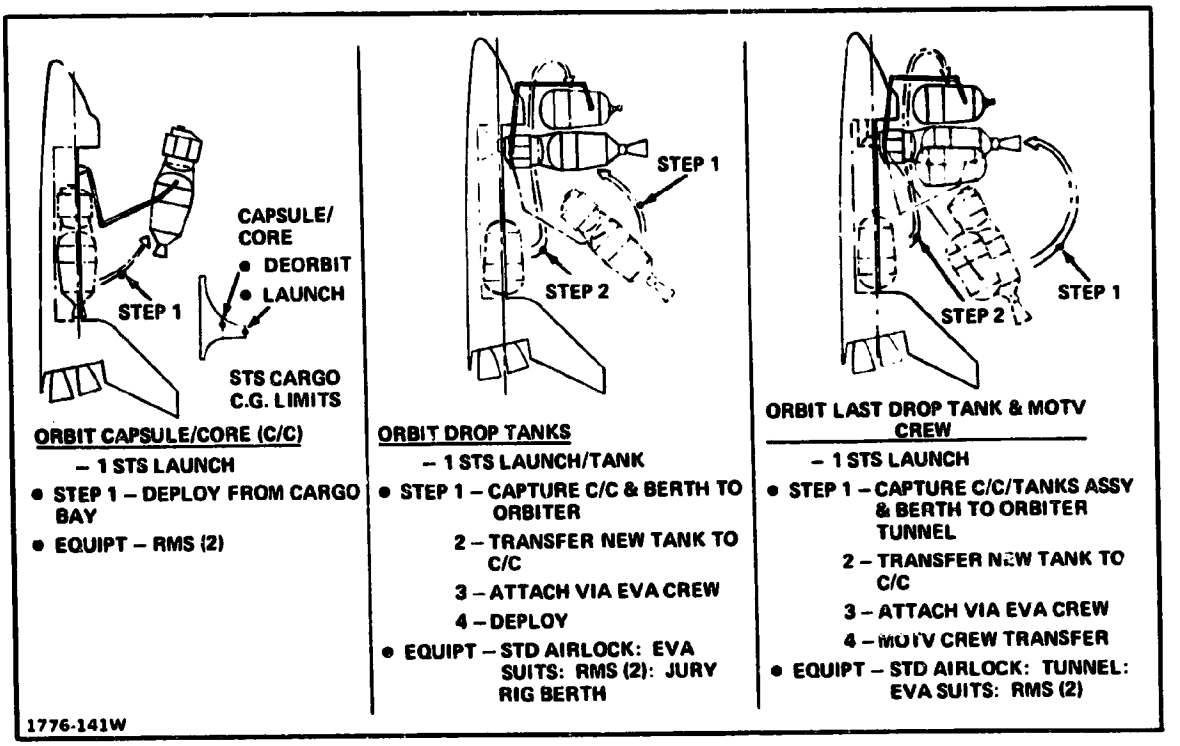

Fig. 6-13 Mission S1: MOTV Launch & Assembly Sequence – Capsule/Core First

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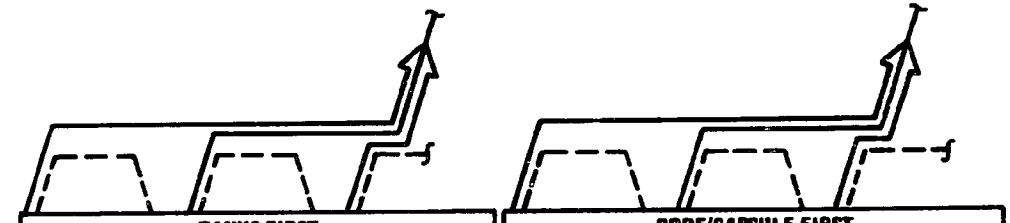

	TANKS FIRST			CORE/CAPSULE FIRST		
	TANK 1 • DEPLOY	TANK 2 • DEPLOY & TEMP MOUNT • RDVS T1 • TEMP MATE	CORE/CAPSULE • DEPLOY & TEMP MOUNT • RDVS T1 & 2 • INSTAL TANKS • CREW TX • C/O DEPLOY	CORE/CAPSULE • DEPLOY	TANK 1 • RDVS & TEMP MOUNT • DEPLOY T1 • ATTACH • DEPLOY	TANK 2 • RDVS & TEMP MOUNT • DEPLOY T2 • ATTACH • CREW TX • C/O DEPLOY
ORBITER PAYLOAD REDUCTION (Kg)						
– RENDEZVOUS		360	360		360	360
– JURY STRUTS	100	100				
– CARGO BAY MOUNT PAD		500			500	500
– TURNTABLE MOUNT			600			
– CREW TX TUNNEL			100			100
TOTAL	2120			1820		

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Fig. 6-14 MOTV LEO Assembly Sequence Options

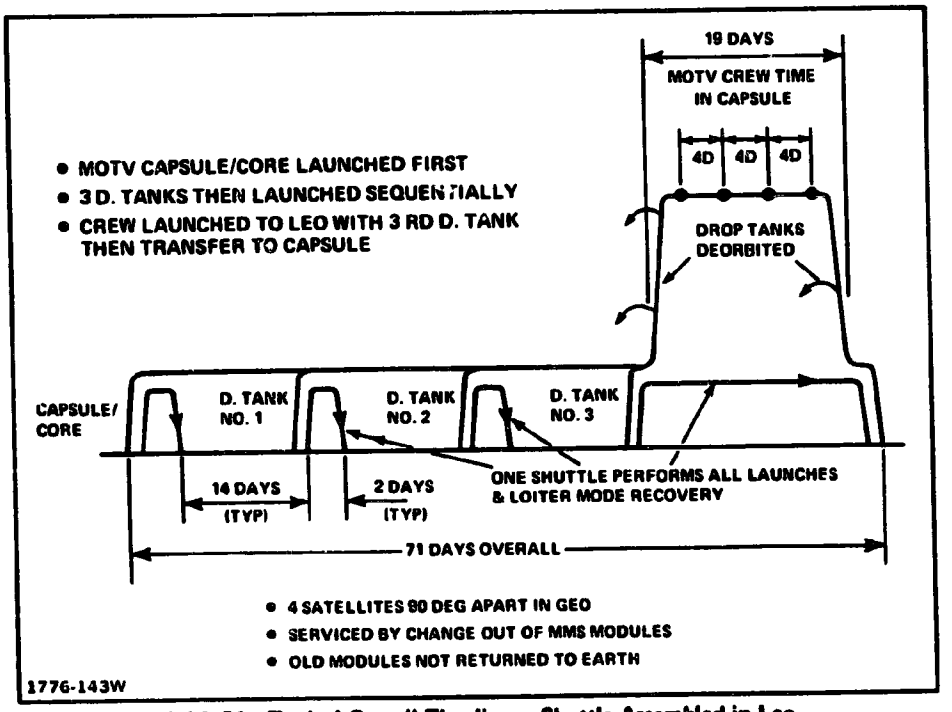

Fig. 6-15 S1: Typical Overall Timeline – Shuttle Assembled in Leo

6.4 WEIGHT CONTINGENCIES AND PERFORMANCE

Weight breakdowns for the capsule and for the assembled MOTV are given in the Mission Handbook. Contingencies of 25% were used for the crew capsule and mission equipments, 15% for propulsion. Performance data in the form of weight and ΔV budget for the mission phases is also contained in the Handbook.

Costs are presented in the Mission Handbook, in the program plan, and in the Final Presentation of the study.

The preceding concept definition discussions dealt mainly with S1, the DRM. It is, in general, equally applicable to the other generic missions. The crew capsule is changed in layout for some of the other missions, as described in the Commonality and Sensitivity section (Section 8) under "Impact of Varying Crew Complement." The MOTV overall configuration will vary in as much as the number of drop tanks differs with mission: some missions carry a solar array while others do not; passenger missions do not carry manipulators or grappler and replace the capsule berthing ring with a docking ring; equipment externally mounted on the crew capsule varies with mission. The Mission Handbook shows the MOTV configuration for each generic mission, together with weights and performance data.

Groundrules governing performance were:

- Propulsion Module
  - 1½ Stage APOTV with two RL10 type engines
  - Drop tanks deorbited and STS compatible
  - Flight performance reserves = 2% total ΔV
  - λ' Core stage, 17,500 Kg Prop. = 0.826
  - λ' Drop tank 27,270 Kg Prop. = 0.941
  - Propellant boil-off 19 Kg/Day/LOX HYD Tank
- Mission Operations
  - STD STS launch to 200 x 200 n mi, 28.5° Orbit
  - STS loiter mode recovery
  - STS turnaround = 14 days typical
- Payload (crew capsule, on-orbit mission hardware hdwr, mission equipment) is mission-dependent.

6.5 SUBSYSTEMS

In phase 1 of the study, we were primarily concerned with analysis of those subsystems which had a gross impact on configuration. Electrical power (EPS), with its possible solar array requirements and fuel cell reactant storage, and environmental control (ECLS), with its radiator requirements, impose limitations on a configuration. Also of interest was the distribution of subsystems throughout the MOTV to determine c.g. and access. Figure 6-16 shows the initial locations of the subsystems, bearing in mind c.g. requirements, reliability, access for maintainability and on-orbit repair, and the desire to minimize relocation when flying unmanned.

6.5.1 EPS

Considering EPS, in phase 1 we defined energy requirements for 20 generic missions as listed in Fig. 6-17. The differences are due, primarily, to varying mission durations and crew size. The totals includes a four-day reserve.

Power requirements for the various MOTV electrical equipments are listed in Fig. 6-18. Two general categories exist:

1) equipment which will vary as a function of crew size, and

2) equipment which is common for most missions.

Two items, electrolyzer and power conditioner, are part of the electrical system and used only in conjunction with a solar array when installed. Main issues of concern in the EPS, together with the options, are:

- Power Source
  - Fuel cells
  - Fuel cells + solar array + regenerative system
- Location of Power Supply
  - Crew capsule
  - Separate module
  - Propulsion core
- Peaking Batteries
  - AgZn
  - NiCd.

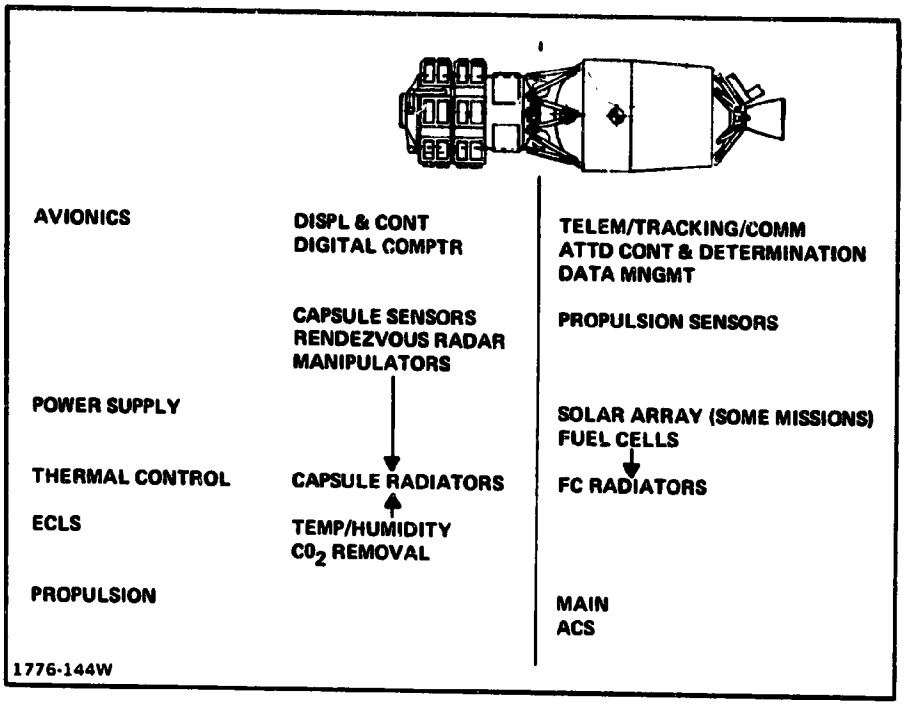

Fig. 6-16 Initial Subsystem Location Assumptions

MISSION	DURATION DAYS	NOMINAL REG-KWHR	RESERVE KWHR	TOTAL	MISSION	DURATION DAYS	NOMINAL REG-KWHR	RESERVE KWHR	TOTAL
IN1	3.8	230	143	373	DR1	8.3	544	143	687
S1	19	1131	143	1274	C1	3	157	143	300
S2	27	1637	143	1780	C2	5.6	347	143	490
S2	TBD				C3	6.0	376	143	519
ER1	3.6	208	143	351	C4	6.8	435	143	578
ER2	3.4	230	143	373	C5	5	427	143	570
R1	2.5	138	143	281		14	938	143	1081
OP1	16	1106	143	1249	C6	26	1836	143	1980
P1	4	230	143	373	UC	TBD			
P2	4	264	192	456					
P3	4	316	336	652					
P4	30	2796	173	2969					

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Fig. 6-17 Mission Energy Requirements

EQUIPMENT REQUIREMENTS RELATED TO CREW SIZE

CABIN	2-3 MEN	6 MEN	10 MEN	30 MEN
ECLS	655	850	1000	2455
WASTE MGMT/HYG	1114	1114	1114	2228
GALLEY	1500	1500	1500	1500
RECREATION	30	30	60	100
ILLUMINATION	135	200	275	675

COMMON EQUIPMENT:

CABIN	
OPTICAL SIGHT	8
CONT & DISP	287
AUDIO CENTER	10
TV (S BAND)	100
CC TV (ELEC)	20
INSTRUMENTATION	40
DATA MGMT INTERF	20
S & C CONTROL	36
MANIPULATOR	35
EPS CONT	20
EXTERNAL	
DEXTEROUS ARMS	400
GRAPPLER ARMS	150
TOOLS	250
CCTV	500
ILLUMINATION	500

PROPULSION MODULE:	
DATA MGMT	466
G & N: INTERNAL	120
LASER	35
COMM: S BAND	323
VHF	36
INSTRUMENTATION	161
ENGINE	192
THRUSTERS	112
ELEC POWER	86
RESERVE	100
ELECTROLYZER	7000
POWER COND (LOSSES)	400

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Fig. 6-18 EPS Connected Loads

For the power source, a weight trade-off is shown in the following figure (Fig. 6-19), which shows that the source should vary from mission to mission. Its location was assumed to be in the propulsion unit; this benefits c.g. and external access and is suitable for unmanned missions. The type of batteries to be used for peaking loads was assumed to be AgZn because of their light weight and modest cycle requirement. Figure 6-19 illustrates the elements which affect solar array sizing. The largest single requirement is the electrolyzer system, which requires 5.5 Kwhr per Kg of O_2 generated. The redundancy, 30%, allows continued use in the event of loss of some solar cell strings.

Variation in weight with energy requirement is shown in Fig. 6-20 for an all fuel cell system and a system which has fuel cells recharged by solar arrays. Energy levels required for each of the generic missions is shown. Fuel cell system weight is a function of the amount of reactant loaded to meet mission requirements. The solar array recharge system is sized for maximum discharge/recharge cycle requirements. Increase in energy requirements as a function of duration affects the number of cycles with the array size remaining constant. For mission requirements below 800 Kwhr the all fuel cell system is lighter but above this level the solar array recharge system presents an advantage. Subsequent work, in phase 2 of the study, revised the boil-off figures for many missions and the revised plot is shown in our final presentation.

A block diagram displaying the two systems (the all fuel cell system and the solar array recharge system) is shown in Fig. 6-21. Fuel cell and peaking batteries, which are for high power short duration pulses, are common to both systems. The all fuel cell system utilizes cryogenic reactant tanks and, in addition to its electrical output, provides potable water for the MOTV. The solar array recharge system stores water and electrolyzes it during the recharge cycle with power from the solar array. Hydrogen and oxygen are formed and stored in gaseous tanks for further recycling.

Figure 6-22 summarizes the redundancy provisions and back-up modes to cope with EPS failures.

The design concept is to continue with the planned mission in the event of a single failure, and a safe return in the event of a second failure in the same EPS section.

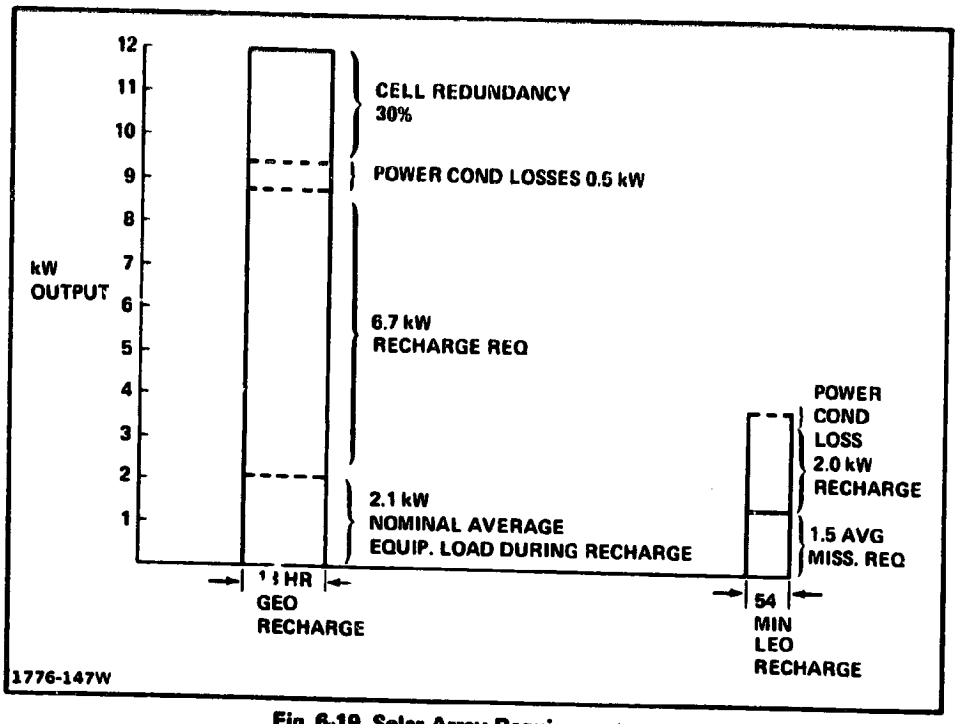

Fig. 6-19 Solar Array Requirements

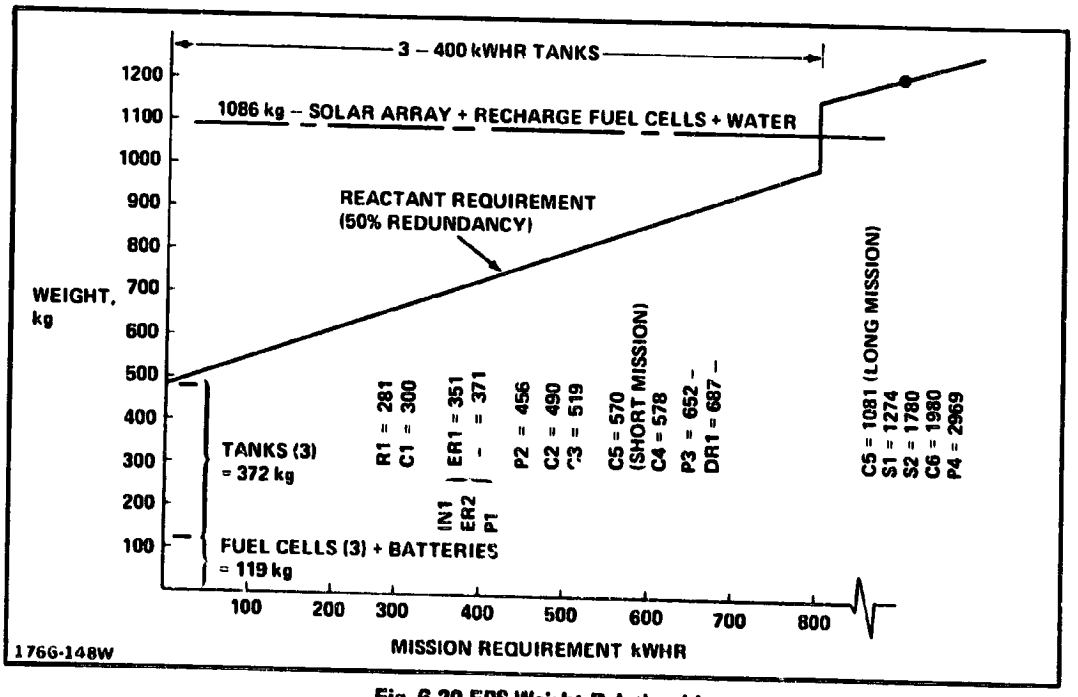

Fig. 6-20 EPS Weight Relationship

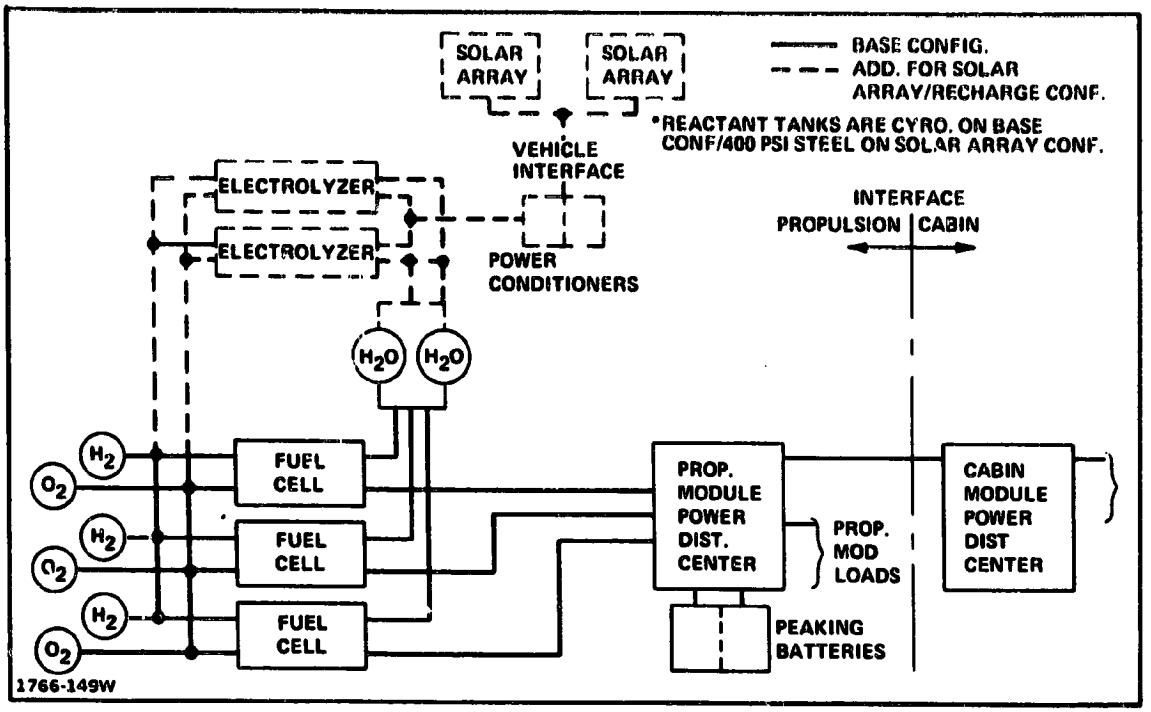

Fig. 6-21 EPS Block Diagram

	FAILURE STATUS	FUEL CELL SYSTEM (ONLY)	FUEL CELLS/SOLAR ARRAY/ RECHARGE SYSTEM
BUS/DISTRIBUTION CENTER	1ST 2ND	F.O. – ISOLATE, BY PASS & CONTINUE F.S. – ISOLATE, BY PASS & RETURN	
FUEL CELLS (3)	1ST 2ND	F.O. – 2 SUFF FOR MISSION F.S. – 1 SUFF FOR POWERED DOWN RETURN	
FUEL CELLS REACTANT TANK (3 SETS)	1ST 2ND	F.O. – 2 SUFF FOR MISSION F.S. – 1 SUFF FOR RETURN (4 DAYS RESERVE)	F.O. – 2 SUFF FOR TRANSFER/WORK F.S. – 1 SUFF FOR RETURN, ALSO ARRAY MAY BE USED AS BACKUP.
SOLAR ARRAY (2 SECTIONS)	1ST 2ND		F.O. – PROCEDURAL CHANGE – SHORTER WORK/RECHARGE CYCLES OR USE ARRAY DURING WORK PERIOD F.S. – RETURN ON F.C.
RECHARGE SYSTEM (ELECTROLYZERS (2)) 1766-150W	1ST 2ND		F.O. – 1 SUFFICIENT FOR RECHARGE F.S. – RETURN ON F.C. AND/OR ARRAY

Fig. 6-22 EPS Redundancy Design Level

6.5.2 ECLS

General design requirements used to size the ECLS system are shown in Fig. 6-23. These requirements are typical for a spacecraft ECLS system, except for the 8 psia cabin pressure. This pressure was chosen to facilitate EVA, or emergency suited operations, without pre-breathing. The basic design philosophy used is similar to that of the Shuttle, i.e., fail-safe with a 96-hour contingency. Missions can be classified as "short duration" and "long duration," and Fig. 6-24 not only identifies the issues associated with ECLS selection, it also indicates the preliminary choices made for these two classifications. (CO_2 removal trade is shown and discussed in Fig. 6-25.)

Three combinations of systems were considered for oxygen and water supply. For the short missions, where the primary electric power is supplied by fuel cells, the choice is clearly to use the fuel cell product water for drinking and carry the breathing oxygen as part of the fuel cell supply. For the longer missions, where primary power will be from solar cells and rechargeable fuel cells will be used instead of an open fuel cell, a system that recycles waste water is desirable. The choice is between a system that recycles waste water for all uses, including drinking, and carries gaseous oxygen, vs a system that carries potable water and reclaims waste water for all uses except drinking, and uses electrolysis for O_2 generation. Of these two systems, the "recycle waste water to potable/carry O_2" system has a weight advantage but requires a water sterilization and quality monitoring system to provide drinking water. Also, the physiologic and psychologic acceptance of drinking reclaimed water for extended periods has not been totally accepted or demonstrated. Therefore, we have tentatively selected the "carry potable/generate O_2" system for this phase of the study. A single fluid heat transport loop was baselined instead of the typical dual fluid system, for simplicity and weight considerations. This approach is reasonable since the coolant circuits within the cabin are of limited volume and present small damage potential.

Figure 6-25 compares a typical regenerable CO_2 removal system (solid amine) with the classic LIOH expendable system. In general, systems sized for large crews and/or long duration missions favor regenerable systems, and small crews/short duration missions favor the expendable systems. As can be seen from the curve, the number of missions falling above the breakpoint curve (regenerable) are about equal to the number falling below (expendable). This plot optimizes the ECLS system, but if the EPS for a particular mission has solar arrays to recharge the fuel cells, then water is

GENERAL:

- PROVIDE SHIRTSLEEVE ENVIRONMENT FOR 2 TO 30 MEN
- MISSION DURATIONS UP TO 30 DAYS

CABIN CONDITIONS:

• TEMPERATURE:	NORMAL	18-24 C	64-75 F
	EMERGENCY	10-32 C	50-90 F
• HUMIDITY:	DEW POINT	10-16 C	50-60 F
• PRESSURE (O_2/N_2)		0.5-0.6 bar	7.5-8.5 psia
• CO_2 pp		3.5 mm Hg.	
• CABIN LEAKAGE		1 kg/DAY	2.2 LB/DAY

CREW:

• AVG METABOLIC RATE	11,200 BTU/MAN DAY
• CO_2 PRODUCED	0.96 kg/MAN DAY
• CONDENSATE	1.58 kg/MAN DAY
• O_2 REQUIRED	0.83 kg/MAN DAY
• POTABLE WATER REQD	2.35 kg/MAN DAY
• URINE PRODUCED	2.06 kg/MAN DAY

REDUNDANCY PHILOSOPHY

- FAIL SAFE WITH 96 HR SURVIVAL PROVISIONS
- ADDITIONAL REDUNDANCY FOR LESS RELIABLE COMPONENTS (E.G., PUMPS, FANS, REGULATORS)

1766-151W

Fig. 6-23 ECLS Requirements

ISSUES

- CO_2 REMOVAL & HUMIDITY CONTROL
  - REGENERABLE SOLID AMINE
  - VS
  - EXPENDABLE LIOH & COND HX

  TRADE

- POTABLE WATER & ATMOSPHERE SUPPLY

	SHORT MISSIONS	LONG MISSION
– POT. WATER FROM FUEL CELL/CARRY O_2	✓	
– CARRY POT. WATER/RECLAIM O_2 FROM WASTE	—	✓
– RECYCLE WASTE TO POT./CARRY GASEOUS O_2	—	

- HEAT TRANSPORT LOOPS
  - DUAL FLUID – WATER IN CABIN LOOP/F-21 IN RADIATOR LOOP
  - SINGLE FLUID F-21 ✓

OTHER SUBSYSTEM FEATURES

- N_2 STORAGE – HIGH PRESSURE GAS
- AVIONICS COOLING – AIR/LIQUID HX (REDUNDANT FANS & FLUID LOOPS)
- CABIN COOLING – AIR/LIQUID HX (REDUNDANT FANS & FLUID LOOPS)
- HEAT SINK – HEAT PIPE RADIATOR – REDUNDANT FLUID LOOPS

1766-152W

Fig. 6-24 ECLS Subsystem Issues & Baseline Definition

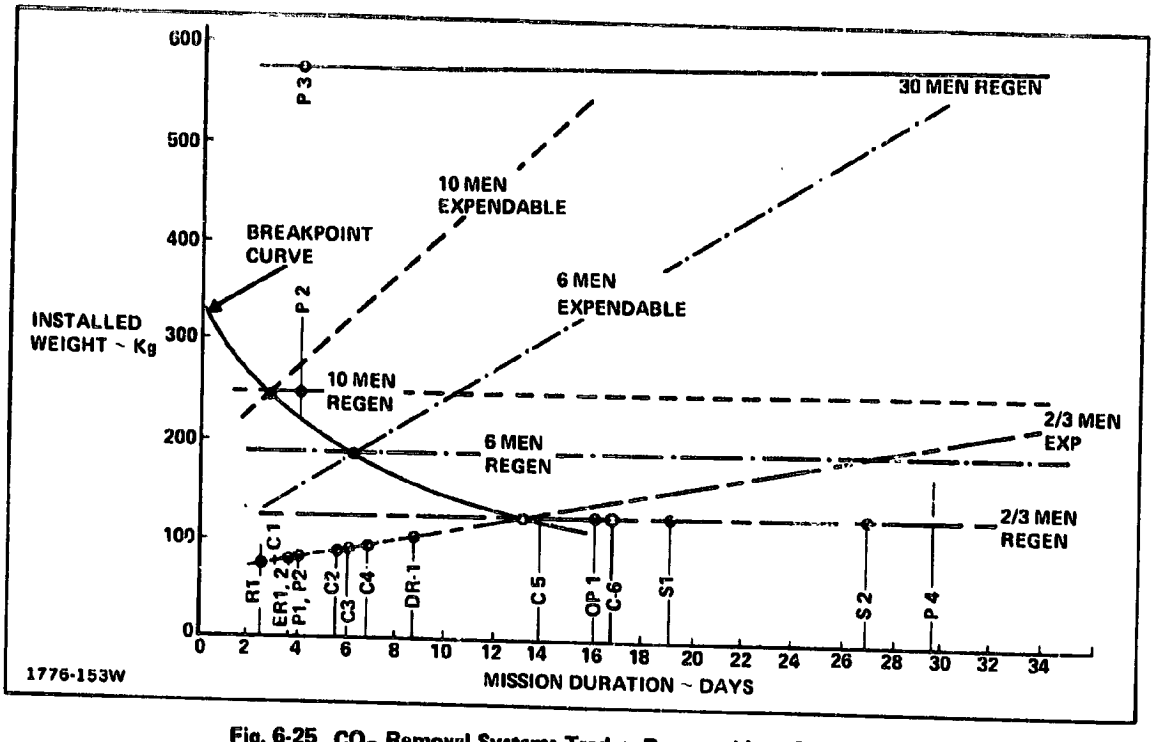

Fig. 6-25 CO_2 Removal Systems Trade: Regenerable vs Expendable

not available for the crew and a regenerable ECLS will be required. The section of this report dealing with Commonality and Sensitivities (Section 8) shows development and unit cost for converting from expendable to regenerative system.

Figures 6-26 and 6-27 show simplified schematics of the two CO_2 removal systems. The major differences are:

Short Missions

- Expendable LIOH system for CO_2 removal
- Store all water
- Store all oxygen

Long Missions

- Regenerable solid amine for CO_2 removal.
- Reclaim waste water for all uses except drinking
- Store potable water
- Electrolysis of reclaimed waste water for O_2 generation

6.5.3 Thermal Protection

To size radiators for the crew capsule, a three-man and a 10-man capsule were looked at to span the likely range. Heat loads used are given in Fig. 6-28, together with the inlet and outlet temperatures. Assumptions were:

- One variable conductance heat pipe radiator
- Radiator capacity loss per meteorite strike = 10%
- Freezing temp below -130°C (-202°F)
- Strip heaters for thaw-out; thaw-out time not to exceed one hour.

Radiator sizes for these loads are 8.3 m^2 for the three-man and 15.8 m^2 for the 10-man capsule. A 30% growth factor is included.

For the fuel cell radiator which is attached to the propulsion core, heat loads assumed are given in Fig. 6-29. Assumptions were:

- One pumped loop radiator for each (of three) fuel cells
- Double redundant fluid circuits; radiator capacity loss per meteorite strike = 20%
- Radiators to accept FC-40 direct from fuel cells
- Strip heaters for thaw-out; thaw-out time not to exceed one hour.

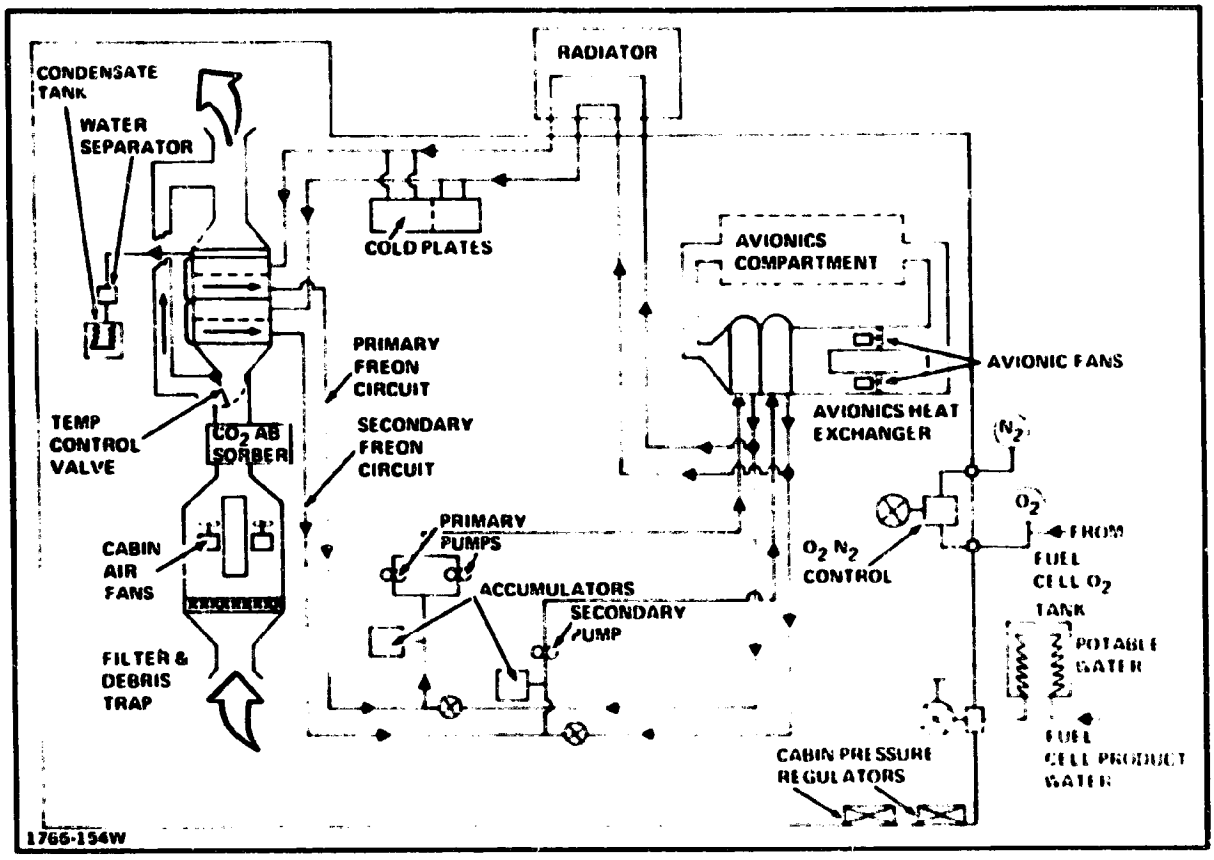

Fig. 6-26 ECLS Expendable System for Short-Duration Missions

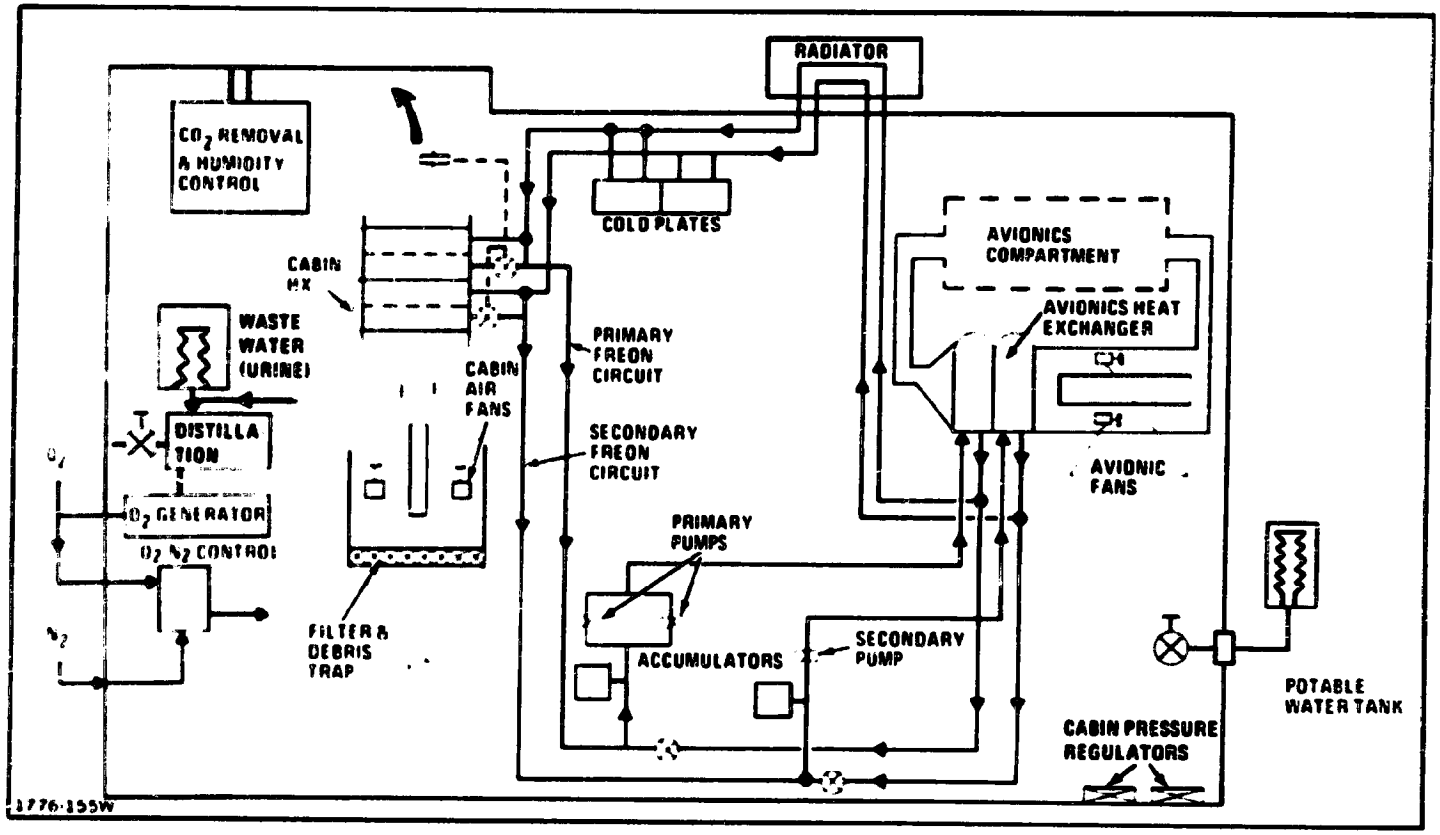

Fig. 6-27 ECLS Regenerable System for Long-Duration Missions

REQUIREMENTS	3-MAN MODULE MAX	3-MAN MODULE MIN	2-MAN EVA	10-MAN MODULE MAX	10-MAN MODULE MIN
HEAT LOADS					
AVIONICS	800 W	300 W	645 W	920 W	400 W
ECLS	655 W	655 W	355 W	1000 W	1300 W
METABOLIC	410 W	210 W	137 W	1377 W	880 W
STRUCTURE	-41 W	-41 W	N.A.	-41 W	-41 W
NET LOAD	1824 W	1124 W	1137 W	3256 W	2539 W
TEMP LIMITS					
INLET	66°C	30°C	30°C MIN	66°C	30°C
OUTLET	16°C	10°C	10°C MIN	16°C	10°C

1776-156W

Fig. 6-28 Crew Capsule Radiator Requirements/Sizing

REQUIREMENTS	3-MAN MODULE MAX	3-MAN MODULE MIN	2-MAN EVA	10-MAN MODULE MAX	10-MAN MODULE MIN
GENERATED LOAD	4.57 kW	1.80 kW	2.64 kW	5.30 kW	2.1 kW
FUEL CELL HEAT REJECTION	2.44 kW	0.93 kW	1.39 kW	2.83 kW	1.1 kW
TEMP LIMITS					
INLET	97 C	23 C	45 C MIN	97°C	23°C
OUTLET	60 C	-43°C	-23 C MIN	60 C	-43 C

1776-157W

Fig. 6-29 Fuel Cell Radiator Requirements/Sizing

Radiator sizes for these loads are 4.4 m^2 for the three-man and 5.1 m^2 for the 10-man mission.

6.5.4 Avionics

This subsystem is located throughout the MOTV. The following placements were assumed for the best functional locations, maintainability and reliability, on-orbit repair, and unmanned flight:

- Display and control: 100% crew capsule
- Data management: 30% capsule; 60% core; 10% drop tank
- Navigation and guidance: 30% capsule; 70% core
- Tracking, telemetry, and command: 15% capsule; 85% core
- Rendezvous radar: 100% capsule.

The display and control subsystem block diagram is shown in Fig. 6-30. Circuit breaker panels control DC and AC power. Caution and Warning display will identify malfunctions. Closed-circuit TV (CCTV) display is provided for IVA and the manipulator controls. Computer CRT display is used with a keyboard for calling up data stored in the CPU. A Data Distribution Center routes the various electrical signals throughout the MOTV. Color TV is available from a hand-held camera for EVA display in the cabin, and for transmission to the ground along with CCTV pictures. A RR display shows the range and bearing data from targets during rendezvous operation.

Data management, Fig. 6-31, accepts status inputs from the various subsystems, signal conditions and converts them from analog to digital read-outs. These inputs are electronically sampled and converted to a data stream by the PCM electronics. This data stream is sent, via the data distribution center, to the ground. Bio-Med and ECLS inputs are also displayed and transmitted to the ground. A tape recorder is provided to work in conjunction with the data distribution center for recording data and voice, and to playback to the ground as required.

Figure 6-32, navigation and guidance, shows the IMU providing attitude information to the CPU via the DIU. Star Scanners are used to provide star-angle measurements for alignment of the IMU. A Horizon Sensor is used to provide redundant attitude information to the CPU. Crew navigation displays and controls are used for automatic and manual control capability for all mission phases. Guidance commands, that drive control loops, are generated to actuate valves in the main propulsion system and the RCS.

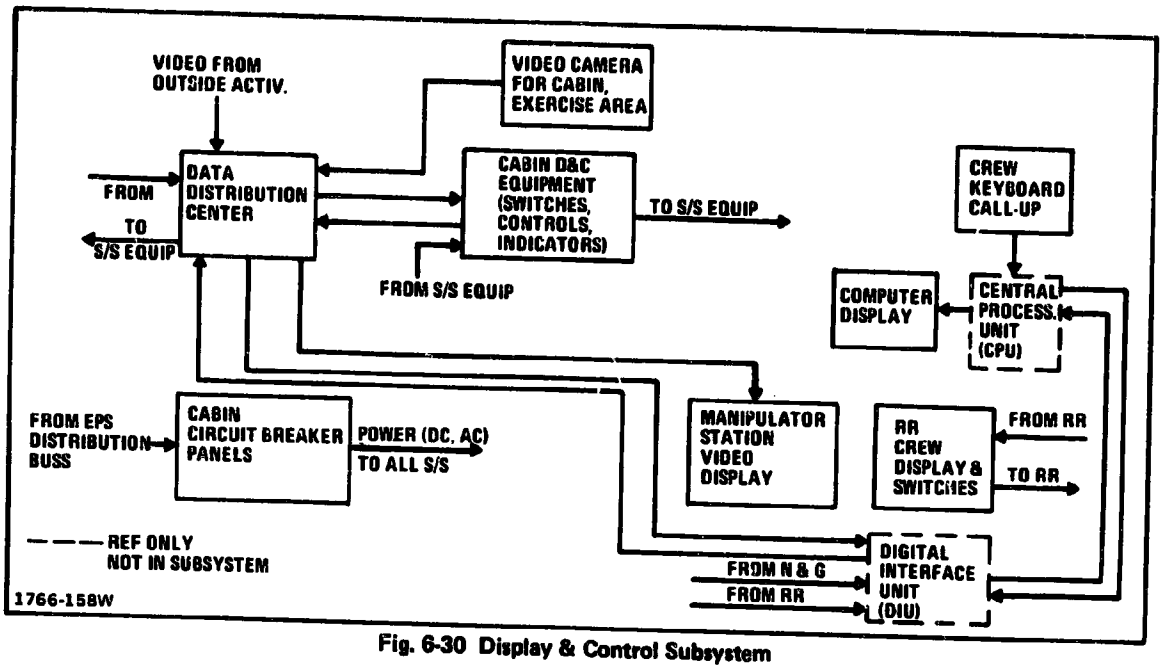

Fig. 6-30 Display & Control Subsystem

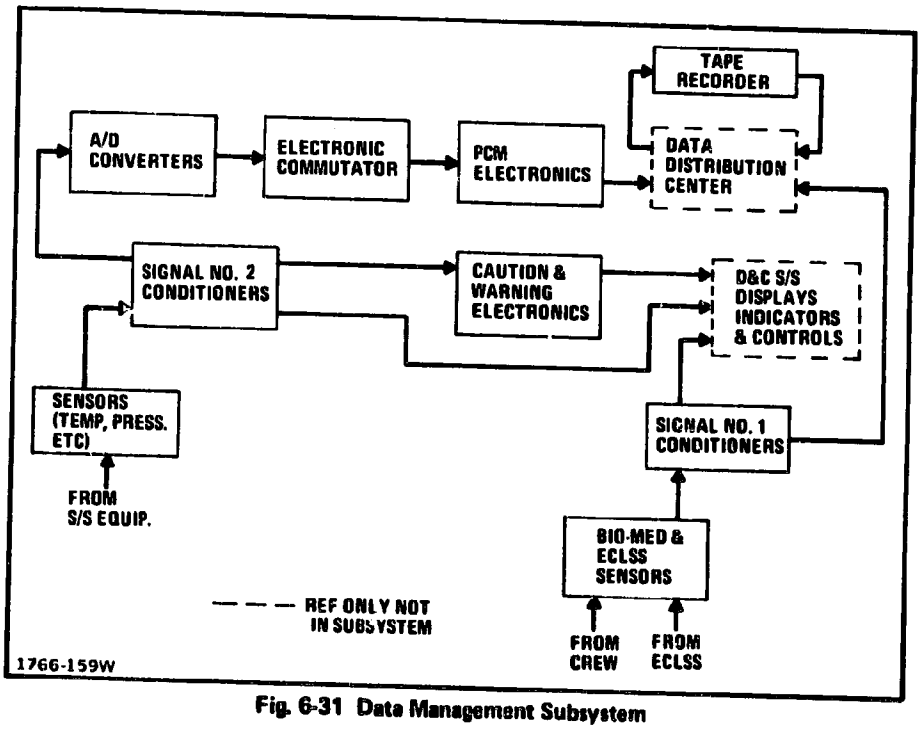

Fig. 6-31 Data Management Subsystem

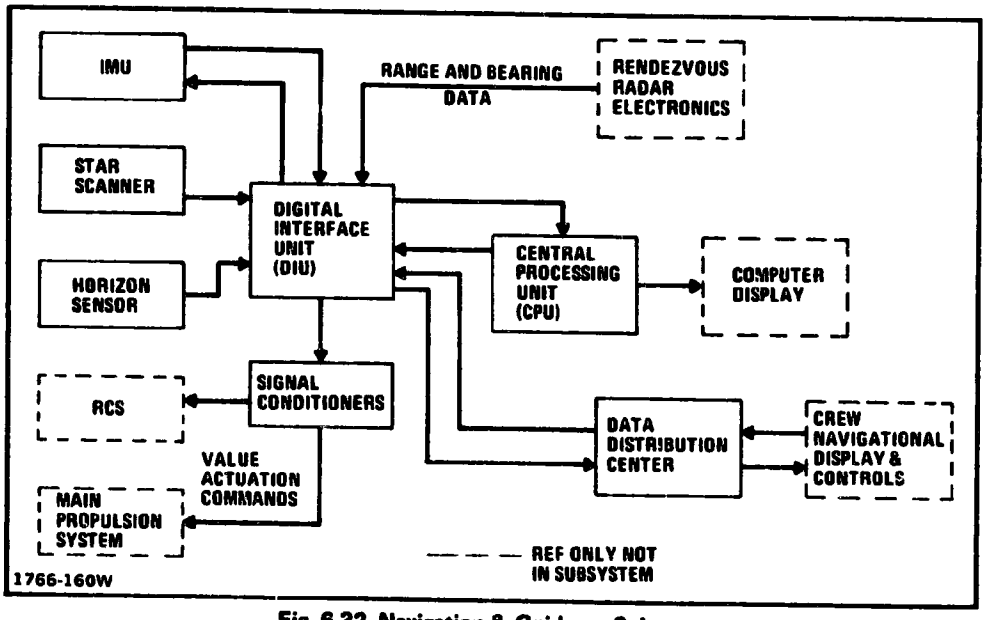

Fig. 6-32 Navigation & Guidance Subsystem

Tracking, telemetry, and command, Fig. 6-33, provides voice communication among crew stations and to outside manned activities. There is an uplink and downlink S-Band transmission between the MOTV and the ground via the steerable antenna. Voice, Spacecraft Status, Crew Health, Color TV, CCTV, and Received Command Data are carried on the S-Band. A turnaround ranging signal is provided for tracking by ground stations using the S-Band Link Carriers. Also, two X-Band Carriers are used with a steerable antenna to carry the same kind of intelligence as the S-band Carriers, but at wider bandwidths and higher data rates. KU band would have been desirable for communication since it is common with the Shuttle, provides wider band widths and higher data rates and it could be used with TDRSS. However, TDRSS is operable to only 3000 n mi altitude, whereas X-Band can be used to geosynchronous orbit with ground spacecraft tracking and data network. It was decided, therefore, to use X-Band to back-up the S-Band downlink.

For Rendezvous radar, Fig. 6-34 block diagram shows the KU-Band transmitter and receiver working in conjunction with a steerable antenna. Inputs are received from Navigation and Guidance to steer the antenna. Range and bearing data are sent to the CPU via the DIU, and to the crew display. Laser beam was considered for this function since it requires less average power than the KU-Band, it is smaller, and it weighs less. KU-Band, however, carries lower risk, is compatible with STS, is less costly, requires less pointing accuracy, has better echo return probability, and is more reliable for autotracking and lock-on function.

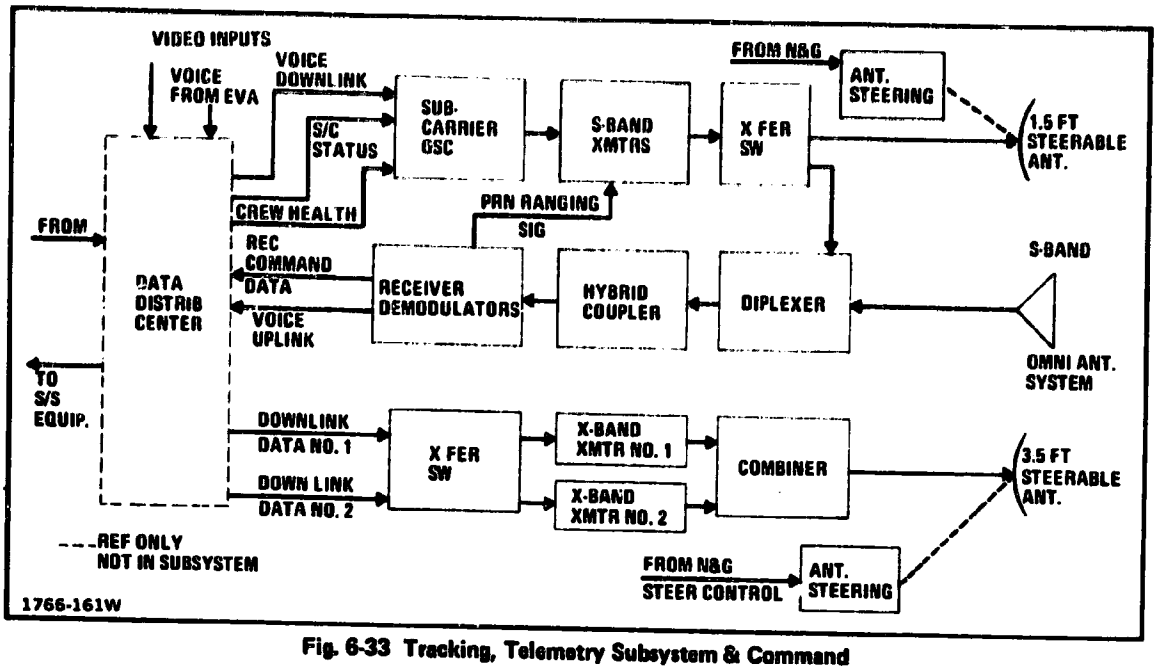

Fig. 6-33 Tracking, Telemetry Subsystem & Command

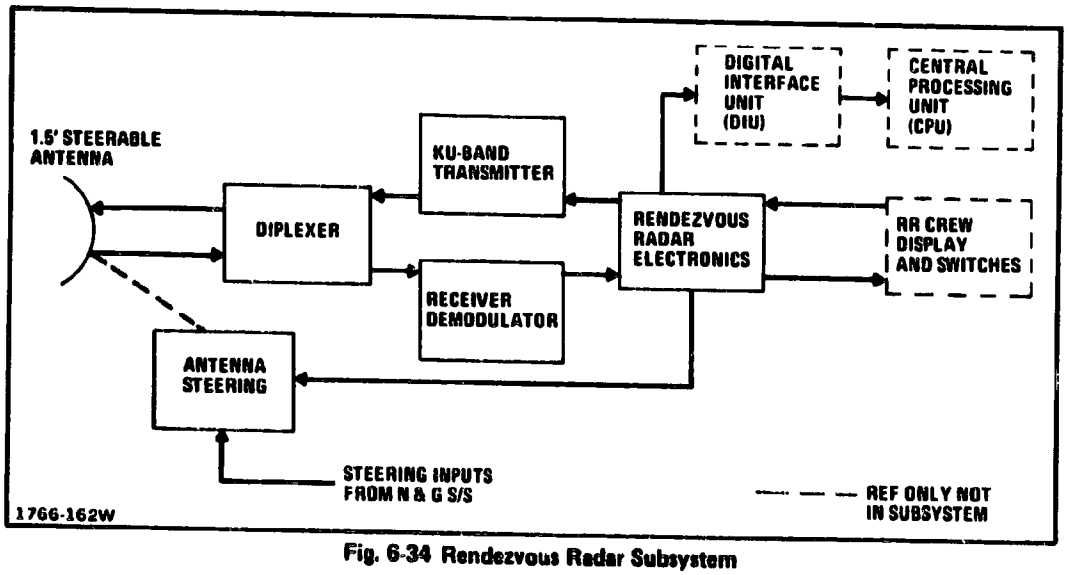

Fig. 6-34 Rendezvous Radar Subsystem

7 - MISSION SUPPORT AND TURNAROUND

This subject is reported fully in the "MOTV Turnaround Analysis Study" document, issued with this report. It summarizes work performed, and already reported on, for Ground and Shuttle-Tended Turnaround of the MOTV, and it reports on recent studies carried out for SOC turnaround.

In summary, the turnaround/maintenance analysis indicates the following:

- The recommended turnaround scenario starts out with ground turnaround because it utilizes in-place facilities, has the flexibility to deal with contingencies which will occur during the operational shakedown period, and provides a benign environment in which to gain experience, work out procedures, and refine support equipment requirements
- SOC turnaround at 200 n mi provides a viable alternate because it decouples the turnaround operations from the STS support flights and saves approximately $11M per mission. SOC turnaround, however, requires a significant investment in facilities, support equipment, and MOTV maintainability features, approximately $330M. Payback takes about 15 years, assuming an MOTV flight rate of six/year. The SOC option should be retained until the appropriate program milestone, when the following can be resolved
  - SOC operational attitude of around 200 n mi rather than the current assumption of 265 n mi
  - definitive costs of facility, MOTV design, and support equipment costs
  - portion of the initial investment for facilities which are chargeable to institutional improvements or other programs.

If the decision at the appropriate program milestone is to proceed with SOC, then the ground turnaround period of two to three years would be followed by an STS-tended LEO turnaround which would be used to qualify and refine the SOC equipment, procedures, and personnel. The final phase would utilize SOC on a progressive basis until the required operational capability was reached.

8 - COMMONALITY & SENSITIVITIES

8.1 TRADE ANALYSES

In considering commonality and the resultant sensitivities, analyses in the form of trades were performed in these areas:

- Number of decks for the Design Reference Mission crew capsule
- Number of capsules to capture all generic missions
- Impact of varying crew complement
- Introduction of emergency "direct return" to earth
- ECLS: Expendable vs regenerative system
- EPS: All fuel cells vs fuel cells + solar array recharge
- Upgraded engine performance
- General sensitivities

8.2 DESIGN REFERENCE MISSION CAPSULE: ONE vs TWO DECKS

The S1 Design Reference Mission (DRM) crew capsule must accommodate three crew for 19 days. Layout of the capsule to minimize weight and cost, yet maximize crew well being and growth potential, is a sensitivity issue. We laid out a capsule which has a single, longitudinal deck and a capsule which has a transverse deck to provide two distinct areas. Figure 8-1 shows the free volume/person curve on which the living volumes were based. It is Frazer's "Tolerance curve," which we were directed to use by NASA.

Figure 8-2 shows the layout of the longitudinal deck capsule by taking a horizontal section through it to show the flight and mission station area, the personal hygiene and waste management area and in the center, the living quarters. This capsule has a burn-out weight of 4495 Kg and is fully described in Section 6.

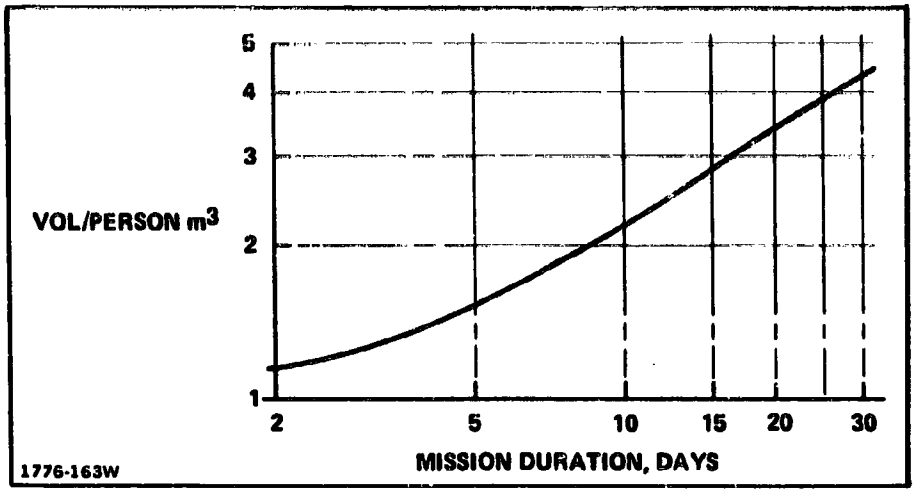

Fig. 8-1 Free Volume Per Man

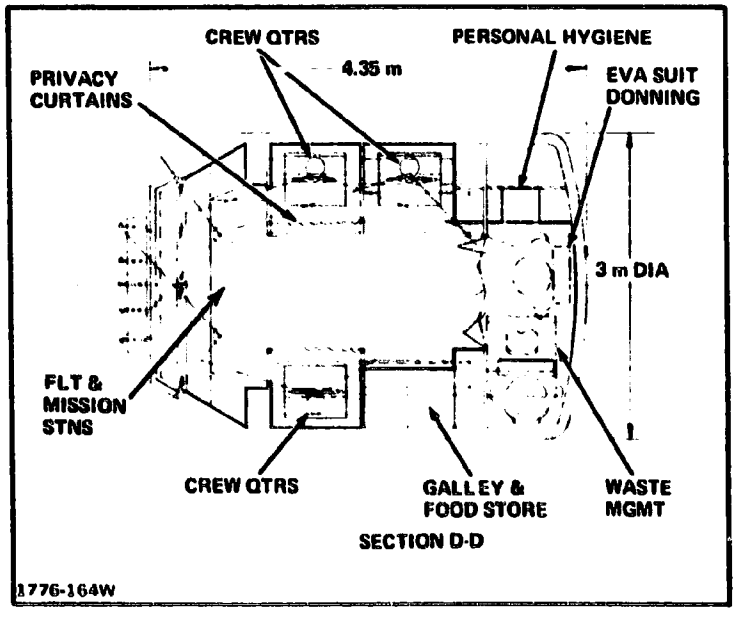

Fig. 8-2 DRM 3-Man Crew Capsule: Longitudinal Deck

The transverse deck layout is shown in Fig. 8-3. It has a living volume and a working volume separated by subsystems stowage, thus providing two decks. The floor orientation for each deck is different. Living facilities are similar to those for the single deck. Waste management cannot be accommodated on the "living" deck and is therefore located in the working deck volume. This has some benefit in that an "on duty" crew member has immediate access to the waste management. This layout has less feeling of spaciousness than the longitudinal deck concept but is shorter in length. Burnout weight is 4327 Kg.

Comparison of the two layouts, together with the criteria used, is shown in Fig. 8.4. When considering only the objective criteria, the two deck version comes out better, but its superiority is only marginal and is, it is felt, outweighed by the subjective criteria and by the growth potential of the single deck. Because of this potential and the better viewing, access, and general spaciousness found in the single deck, the latter is the preferred configuration and is the capsule for the DRM.

8.3 CAPSULES TO CAPTURE ALL GENERIC MISSIONS

Based on the volumes information contained in the preceding paragraphs, Fig. 8-5 shows the total cabin volume required for each generic mission. The three-man DRM capsule encloses 24 m^3, which captures all of the generic missions except for P4 and S2.

P4 is sized for six people but since it uses a storm shelter to house some of these crew, the basic crew capsule for this mission is still the 24 m^3 baseline capsule.

S2 mission is for three people for 27 days. It is the free volume required for this long duration, which pushes the total volume requirement to just over 26 m^3. Reducing this volume to 24 m^3, thus using the baseline capsule, would have little effect on crew comfort because that capsule is sized for three people, 19 days, and it is a subjective judgement as to when the free volume/man curve should flatten out.

Six of the missions can be accommodated in a 17 m^3 total volume capsule. This capsule was laid out and, as shown in Fig. 8-6, it has a combined flight/mission station with room for the second man to be in tandem for backup and mission monitoring. Aft of this work station, on the "starboard" side of the vehicle, are subsystems, EVA suits, and the galley. The "port" side of the capsule has two crew quarters and hygiene and waste management. Volumes for these latter facilities follows the three-man baseline crew capsule. At burn-out, this capsule weighs 3281 Kg. Costs were $426M DDT&E plus $96M for two production capsules and their derivation is shown in Fig. 8-7. To

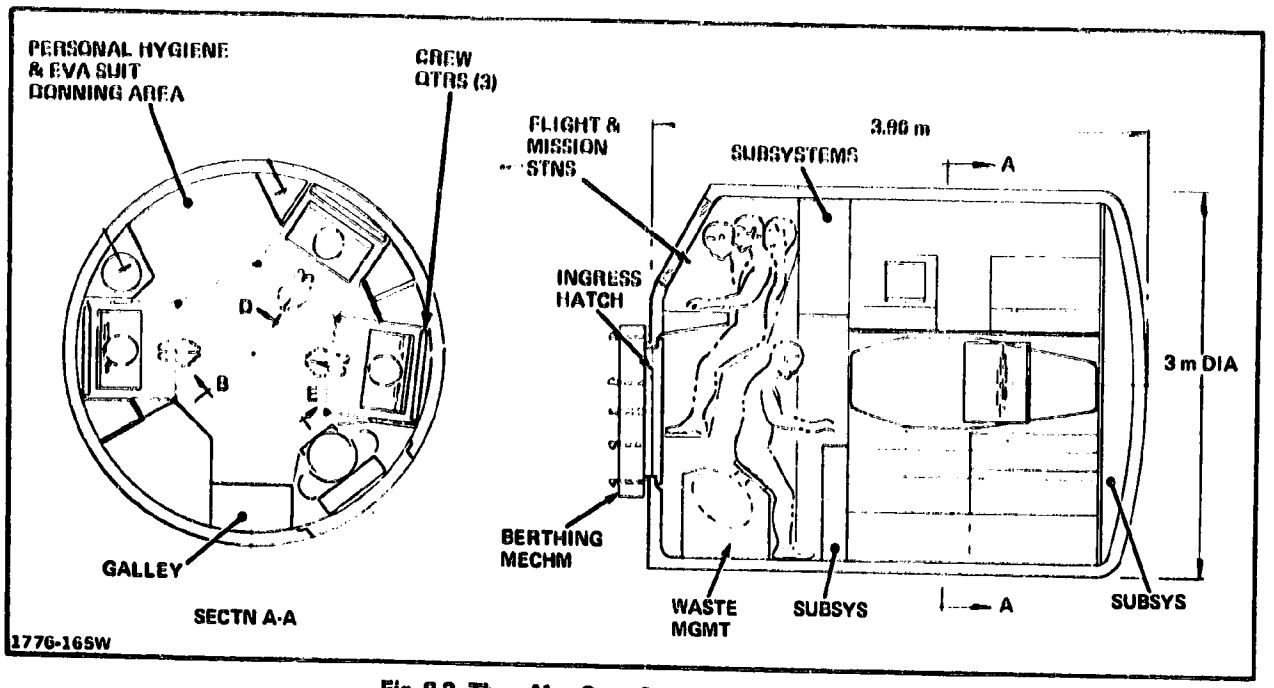

Fig. 8-3 Three-Man Crew Capsule: Transverse Deck

CRITERIA	SINGLE DECK	TWO DECKS
OBJECTIVE:		
• LENGTH	4.35 m	3.90 m √
• PENETRATION INTO WORK AREA	√	
• VOLUMES		
– Δ GROSS	+ 2 m^3	√
– CREW		
– HYGIENE	APPROX SAME	
– GALLEY		
– W/M		
– SUBSYS (2.5 m^3 REQD, 19 DAYS)	4.25 m^3	2.5 m^3 √
• GROWTH CAPABILITY TO 30 DAYS	√	+92 kg
• Δ WEIGHT	+ 168 Kg	√
SUBJECTIVE:		
• WORK STATION VIEWING	√	
• CREW SENSIBILITIES		
– SPACIOUSNESS/FREE VOL	√	
– W/M PRIVACY	√	
– ACCESS TO SUBSYS	√	
– ACCESS TO WORK STATION	√	
√ = PREFERRED		
1776-166W		

Fig. 8-4 Three-Man Crew Capsule Options

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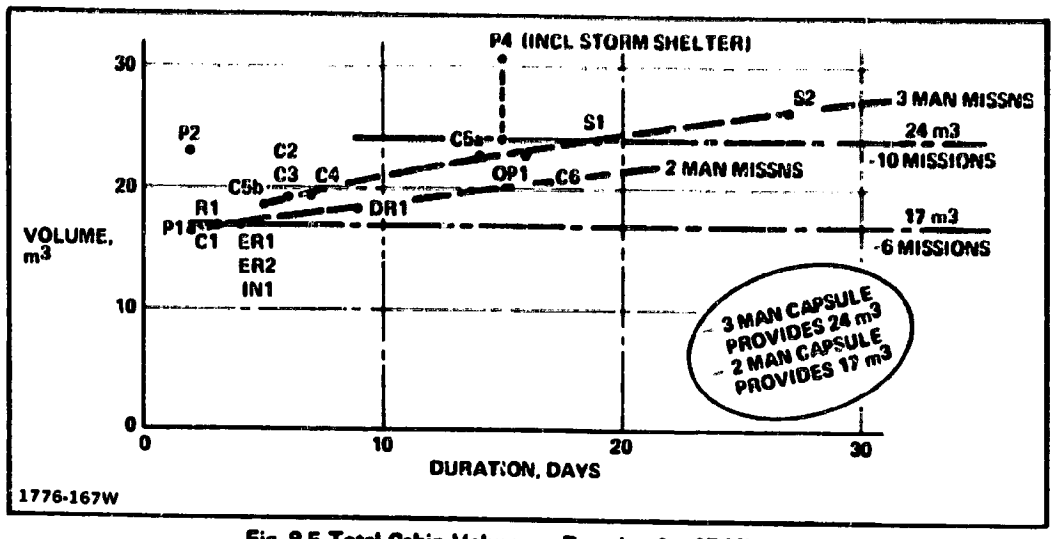

Fig. 8-5 Total Cabin Volume vs Duration for 17 Missions

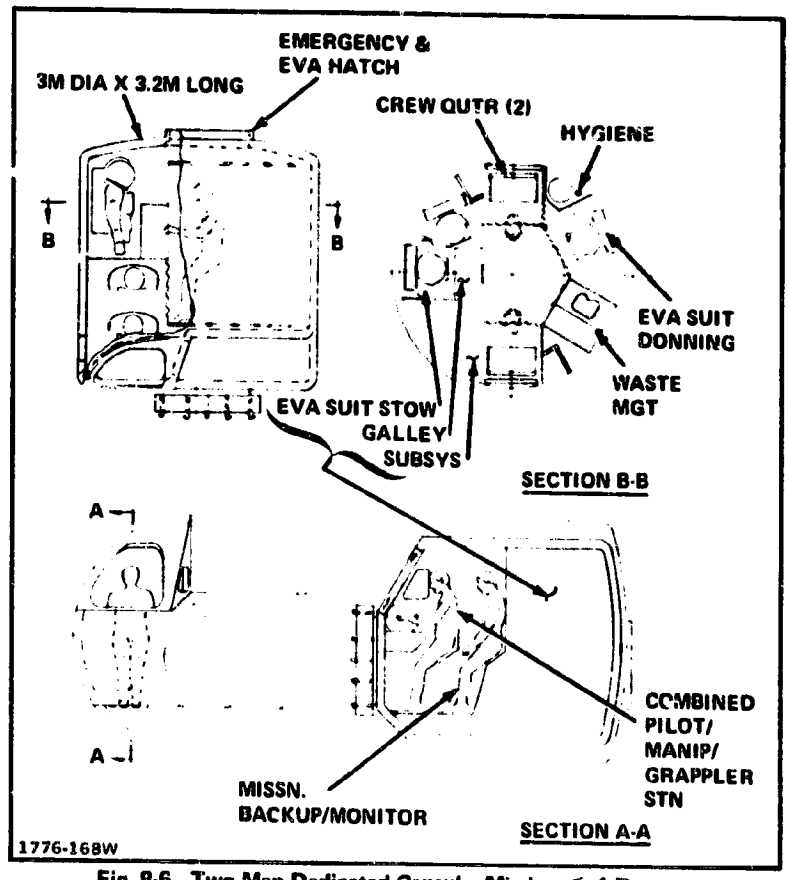

Fig. 8-6 Two-Man Dedicated Capsule, Mission ≤ 4 Days

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	DDT&E	PROD (2)	TOTAL
PROJECT MANAGEMENT	34	7	41
SE & I	28	–	28
STRUCTURE	89	14	103
INSULATION	5	1	6
MECHANISMS	26	14	40
EPS	4	1	5
AVIONICS	42	7	49
GN/C	18	3	21
INSTRUMENTATION	6	1	7
COMM/DATA HNDLG	3	1	4
CONT & DISPLAYS	15	2	17
ECLS	113	23	136
CREW ACCUM	71	15	86
SOFTWARE	5	–	5
SYS TEST & EVAL	9	–	9
INST ASSY CHECKOUT	–	14	14
1776-169W	426	96	522

Fig. 8-7 Two-Man Dedicated Crew Capsule Dev/Prod. Cost (Constant 1979 $ M)

determine whether it pays to develop such a capsule, in addition to the three-man baseline crew capsule design, a payback analysis was prepared to find the traffic rate necessary to payback the additional investment. Shown in Fig. 8-8 are the results of an analysis which assumes that the 17 m^3 capsule is a completely new development with no benefit from the three man capsule development, and a more probable case which assumes that both capsules are developed concurrently. The results show that a traffic rate of at least four flights of short duration, two-man capsule missions are necessary each year to amortize the parallel program investment in 10 years. At this stage, it was decided to discontinue study of the smaller 17 m^3 capsule because of the payback analysis and because of the seeming impracticability of growth.

8.4 IMPACT OF VARYING CREW COMPLEMENT

Although none of the generic missions presently calls for a crew of four, we investigated how to put a fourth man into the DRM three-man capsule, as a sensitivity issue. As minimum modification to the DRM capsule, an additional crew quarter was located as shown in Fig. 8-9. This fourth crew quarter is similar to the other three except that the sleeping position is vertical and contained within the quarters. The additional quarter is a bottleneck in the fore and aft traffic flow, but there is sufficient room for people to get by. Compared to the DRM, three-Crew Capsule, this four-man capsule carries a weight penalty of 531 Kg for additional ECLS, crew accommodations, crew and crew consumables.

The P2 and the P4 missions call for crews of six and eight, respectively. In the interests of commonality, we investigated using the DRM three-crew capsule for these missions. P4 is for crew rotation of six men and resupply of a manned facility located in deep space. This requires a one-way trip approaching 15 days. It is assumed that, by the mission date, there will be a reliable long-range prediction capability for forecasting solar events. Referring to Fig. 8-10, the assumption that this mission can be then scheduled for a period of low radiation, the capsule will be the design reference mission capsule, modified to accommodate four people as described in the preceding paragraph and shown in Fig. 8-9. The remaining two crew will be accommodated in a storm shelter which, because of the long-duration trip, should be provided in the event of an unpredicted solar flare. The shelter is sized to accommodate six men in cramped quarters, riding out the storm. When not in use as the shelter, it provides two crew quarters for the men not accommodated in the main capsule. The shelter is mated to the capsule in LEO during MOTV assembly and attaches to the capsule emergency exit provided for normal use. An emergency exit is provided in the shelter.

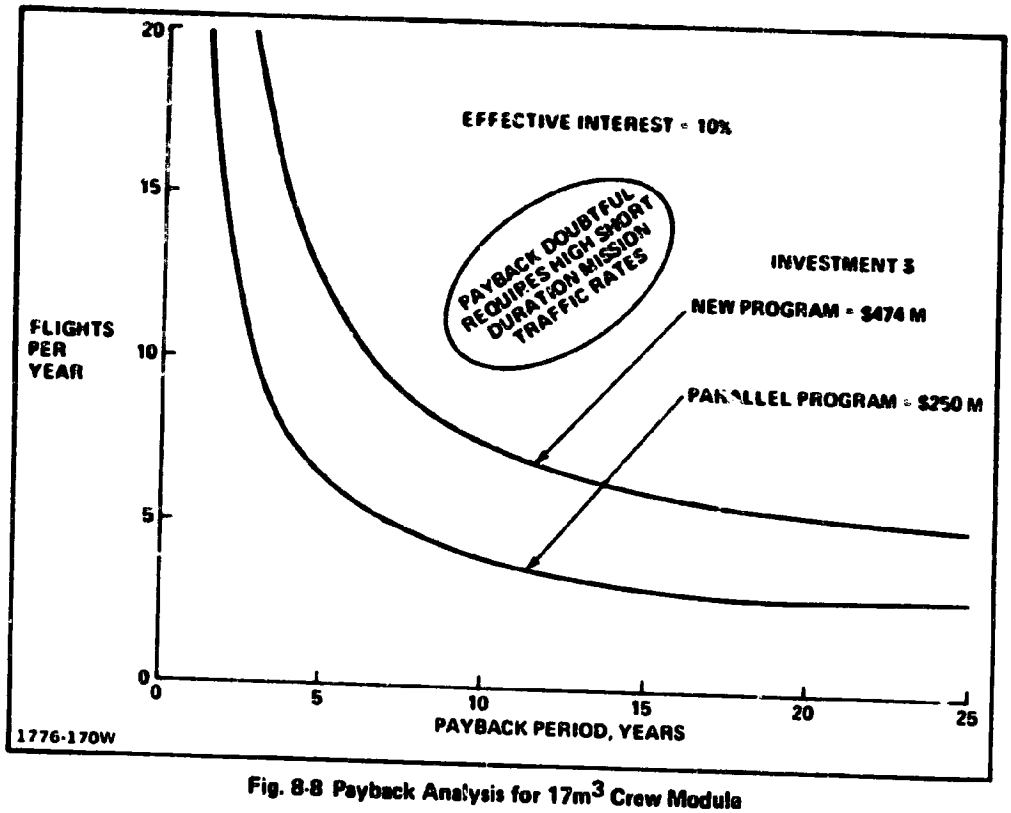
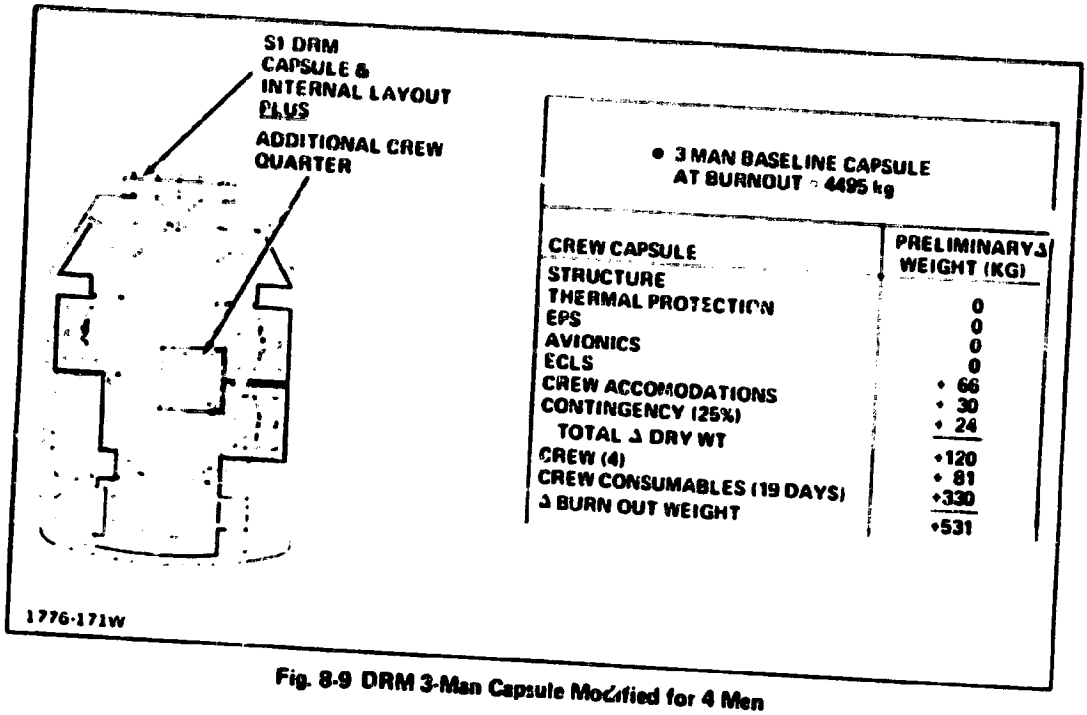


Fig. 8-8 Payback Analysis for 17m^3 Crew Module



• 3 MAN BASELINE CAPSULE
AT BURNOUT ≈ 4495 kg

CREW CAPSULE	PRELIMINARY Δ WEIGHT (KG)
STRUCTURE	0
THERMAL PROTECTION	0
EPS	0
AVIONICS	0
ECLS	0
CREW ACCOMODATIONS	+ 66
CONTINGENCY (25%)	+ 30
TOTAL Δ DRY WT	+ 24
CREW (4)	+120
CREW CONSUMABLES (19 DAYS)	+ 81
Δ BURN OUT WEIGHT	+330
	+531

Fig. 8-9 DRM 3-Man Capsule Modified for 4 Men

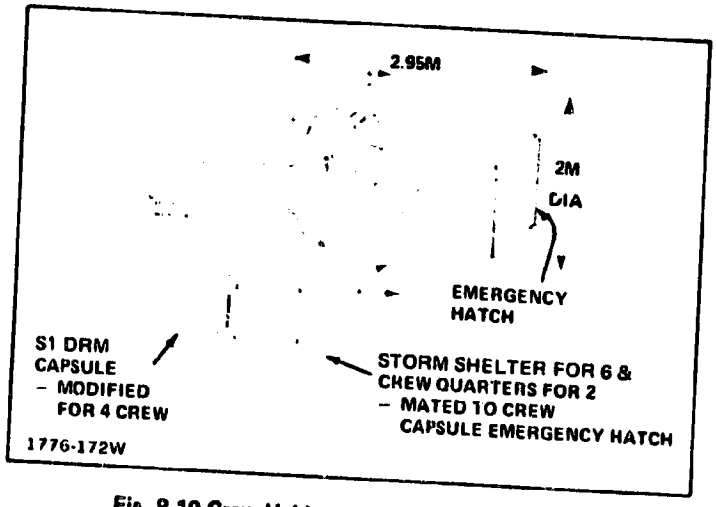


Fig. 8-10 Crew Habitat for 6 Men P4 Mission

Compared to the DRM capsule weight, the penalty for P4 was 120 Kg for the four-man capsule plus 350 Kg for additional ECLS to cater for two more crew. The storm shelter dry weight is 2317 Kg, which reflects a structural shell designed to give 10 g/cm^2 of aluminum equivalent thickness. This provides shielding to reduce the proton skin dose to 24 rem. With the crew and consumables, the total burn-out weight penalty for P4 crew habitat of capsule and shelter is 5284 Kg.

An eight-man capsule is required for mission P2, crew rotation to GEO. The people are only in the capsule for about a day and do not, therefore, require separate quarters or cooking facilities. As shown in Fig. 8-5, the volume for this mission can be provided by the DRM crew capsule. Referring to Fig. 8-11, minimum modification to this capsule retains the forward work station, which is occupied by two of the crew, and the aft station for personal hygiene, waste management, and EVA suit stowage/donning. The center section of the capsule is changed by removing crew quarters and the galley, then replacing them by six seats. Thus, eight crew are accommodated. A docking ring replaces the berthing ring.

Comparing the weight of this capsule to that for the DRM capsule, there is less structure and crew accommodations due to removed quarters and galley. ECLS has increased to cater for the additional men but, because this is a shorter mission than the DRM, consumables are less. Thus, 248 Kg are saved.

To accommodate two men in the three-man DRM capsule requires, at the least, no change; but weight may be saved by removing one of the crew quarters. The Δ cost is negligible. For four men, the addition of one crew quarters, with some extra ECLS and consumables, costs $1.5 M extra per unit. The six-man habitat is the four-man capsule with the same $1.5 M additional cost. Over and above that are the costs for a storm shelter, required by the mission, which also houses two crew members. The combined DDT&E plus the cost per unit amounts to $152 M for the shelter. More extensive work is required to modify the baseline capsule to accommodate eight men, and this carries a cost per unit of $3 M. There is, therefore, little cost penalty to the DRM capsule when adapting it to accomoodate all missions.

8.5 INTRODUCTION OF EMERGENCY "DIRECT RETURN" TO EARTH

Our baseline concept for the APOTV assumes that emergency return from GEO will be to LEO for rendezvous with a loitering STS. This is discussed in the Section 3. There are, however, some opinions that a capability should be provided for direct return from GEO to earth.

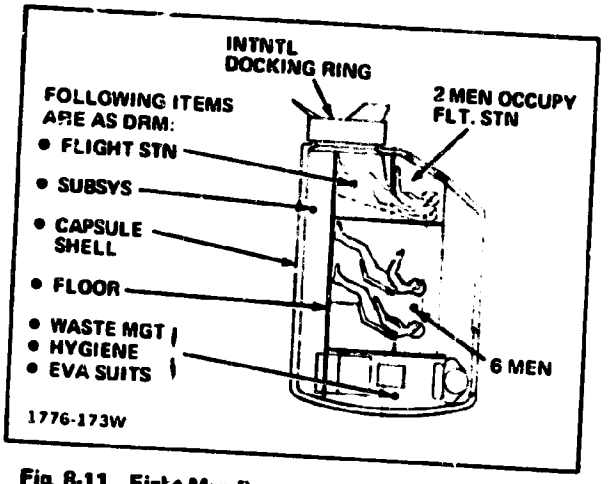


Fig. 8-11 Eight-Man Passenger Transport Capsule (P2)

One option has a MOTV with "minimum development" crew accommodation and mission operation equipment. The three-person crew is housed in an Apollo CM, the capsule in which the crew can return directly to earth in the event of an emergency. The on-orbit mission tasks are performed by a crewman operating a free-flying Manned Remote Work Station Module (MRWS), which is a vehicle under consideration for future study in Grumman's MRWS Study contract. This module is docked to the Apollo CM for shirtsleeve transfer of the operator. The vehicle may be flown as an APOTV, with the CM for emergency return, or it may be flown as an AMRV with the propulsion and MRWS expended on each mission. The cost per mission for this minimum development "direct return" capability is $40.05 M plus DDT&E costs amortization, and it carries a weight penalty of 4000 Kg for the two crew capsules compared to the DRM capsule.

Considering the DRM, which has three crew for 19 days, the Apollo CM is not only restrictive for a mission of that duration but it also does not cater for a mixed gender crew because it provides no privacy and has personal facilities which are relatively primitive. As an alternative to that "minimum development," Fig. 8-12 shows a configuration which utilizes a small capsule to house two of the crew in privacy. The capsule also provides the mission station with manipulator and stabilizer controls, personal hygiene and waste management with privacy curtains, food preparation, and subsystems. A second module serves as quarters for the third crewman, probably the commander, and also has the MOTV flight control subsystem. This module is the emergency escape vehicle and can accommodate all three crew for emergency return. It has Apollo CM shape and capabilties and, although presently conceived as being a little smaller than the CM, it could in fact be a CM shell. The vehicle may be flown as APOTV or AMRV. The cost per mission for this "direct return" capability is $35.4 M plus DDT&E costs amortization, but it incurs a weight penalty for the two capsules of 3600 Kg compared to the DRM capsule.

8.6 ECLS: EXPENDABLE vs REGENERATIVE SYSTEM

In the concepts definition section (Section 6), we show that, as a function of installed weight, the number of generic missions favoring a regenerable (solid amine) system is about the same as the missions favoring an expendable (lithium-hydroxide) system. The simplified schematics, illustrated in Fig. 8-13, show the expendable system and that part of the system which will be replaced when converting to a regenerable system. Development costs amount to $2.9 M with the average unit cost being $0.25 M.

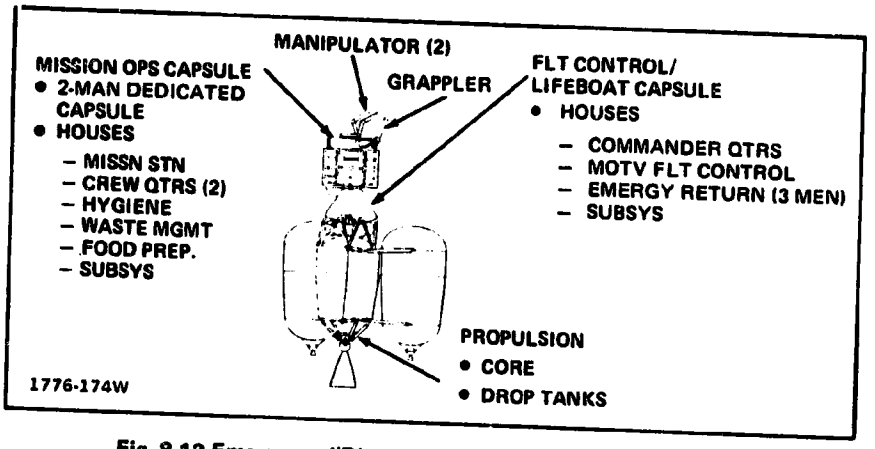


Fig. 8-12 Emergency "Direct Return" – 2 Dedicated Capsules

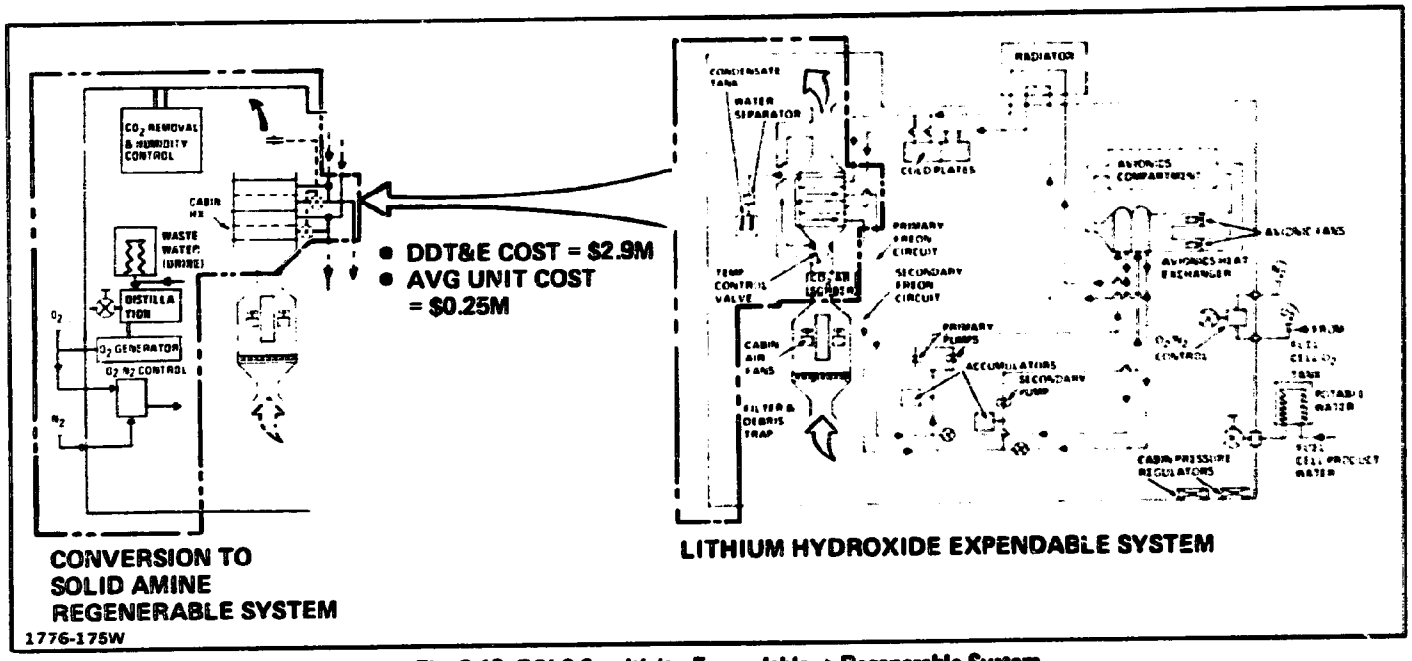


Fig. 8-13 ECLS Sensitivity Expendable → Regenerable System

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8.7 EPS: ALL FUEL CELLS vs FUEL CELLS AND SOLAR ARRAY RECHARGE

Studies conducted and reported in Section 6 show that, at mission energy requirement above 800 KWhr, the addition of a solar array recharge system to the basic fuel cell EPS has an advantage. Figure 8-14 shows, with dotted lines, the added equipment for conversion to solar array recharge. The H_2 and O_2 tanks require change out from cryo, for the all fuel cell system, to gaseous when solar array recharging is introduced. Costs penalties are $16.96 M for DDT&E and $10.01 M for TFU (two sets).

8.8 UPGRADE ENGINE PERFORMANCE

A costs sensitivity trade to replace the two RL10 Derivative IIB type engines is shown in Fig. 8-15. The characteristic of each of the engine and their costs are summarized.

A payback analysis for upgrading the engine is shown in Fig. 8-16. Based on an estimated differential cost of $58M (1979 constant $) for development, and an effective interest rate of 10%, a payback period was determined as a function of MOTV's to GEO per year. At two flights per year, the cost of development could be amortized in five years.

8.9 GENERAL SENSITIVITIES

Some general sensitivity figures that have been used in this study are:

- Savings for upgrading the engine I_{sp} from 459 seconds to 473 seconds are $8.4 M per mission
- Cost for adding each crew member is $1.5 M
- Increasing the round trip payload costs $0.01 M per kilogram
- Increase in mission duration costs $0.75 M per day.

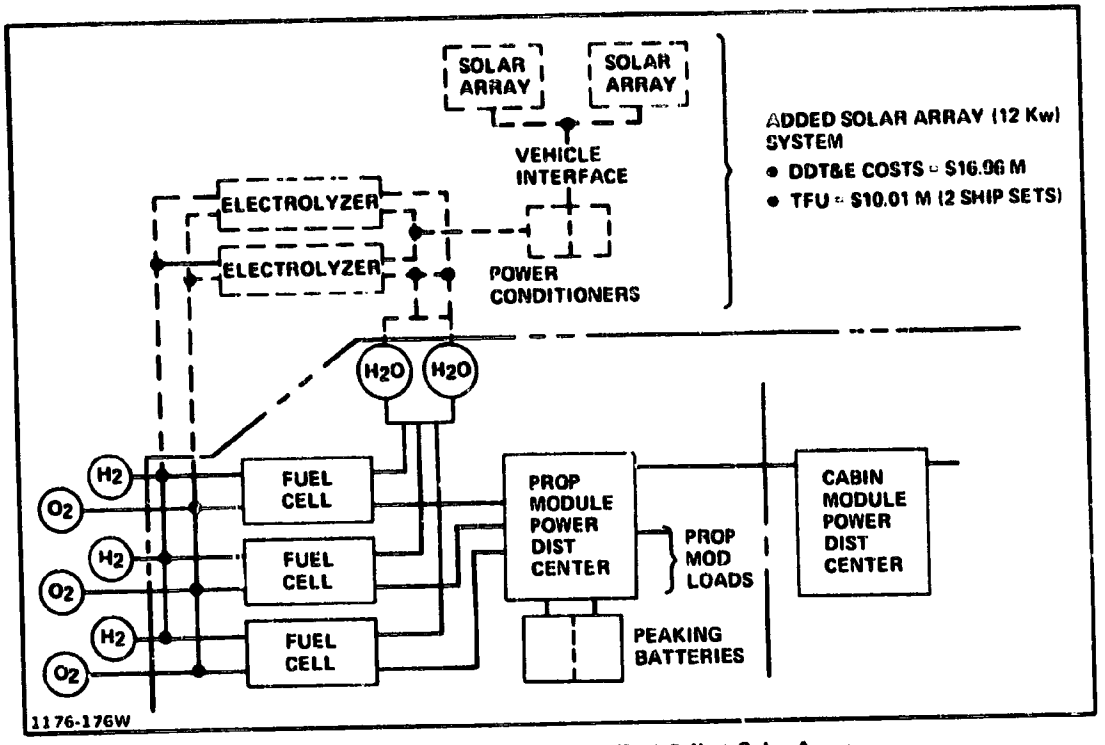

Fig. 8-14 EPS Sensitivity Fuel Cells → Fuel Cells + Solar Array

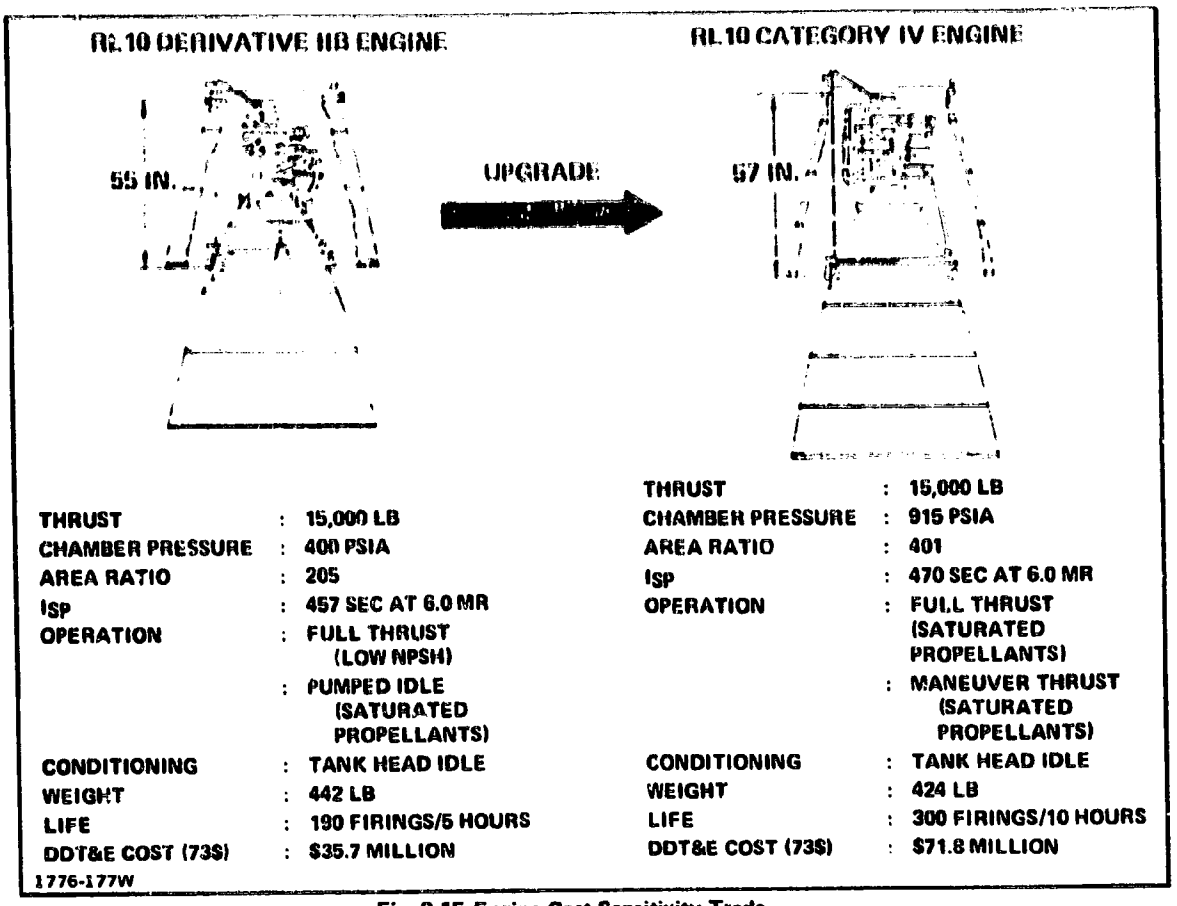

Fig. 8-15 Engine Cost Sensitivity Trade

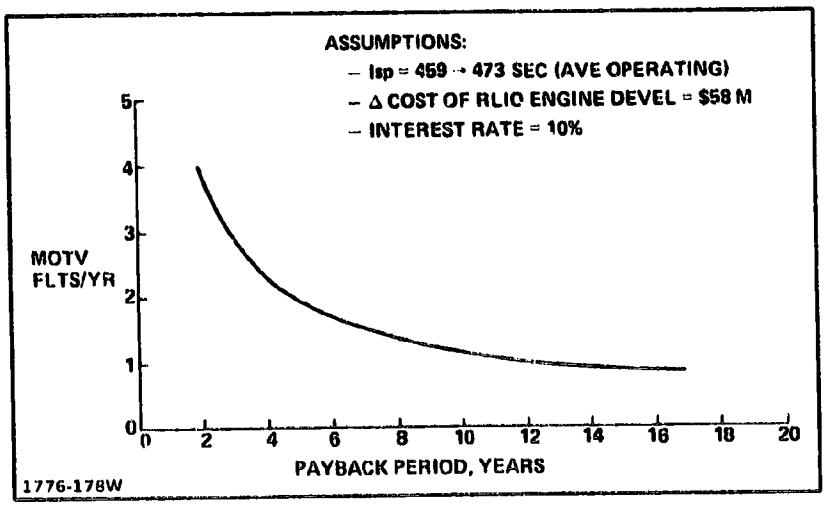

Fig. 8-16 Payback Analysis for Upgrading Engine Performance

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9 - COSTS

9.1 GROUNDRULES AND ASSUMPTIONS

Costs figures are provided in the Mission Handbook, the Program Plan, and the Final Presentation of this study. As background to those figures, the following groundrules and assumptions were made in their derivation:

- General
  - Costs data are in 1979 constant dollars
  - Mission rate is four missions/year for 10 years
  - All turnaround is on the ground
  - Crew salaries and space suits are excluded
- DDT&E
  - Three STS flights are used to develop mission capabilities in LEO
  - One set of equipment is included in Ground Support equipment costs
  - Non-replicated weights are used for computing costs
  - Tools are Government Furnished Equipment
  - One set of equipment is included in Flight Support Equipment costs
  - Manipulators are off-the-shelf
- Operations
  - Drop tank cost per mission is based on the average production unit cost of 60 units at 85% learning.

9.2 METHODOLOGY

Figure 9-1, presents the costing methodology of those WBS items which were estimated as a percent of certain WBS cost aggregations (noted in parentheses). The systems test and evaluation category represents primarily testing labor. The SE&I category is considered non-recurring while the IACO category is considered as recurring cost.

The approach used throughout the study for cost analysis is illustrated in Fig. 9-2 with supporting WBS example shown in Fig. 9-3 and 9-4, which elaborates on level 5. Figures 9-5 and 9-6 show costs comparisons between the LM Ascent Stage and the

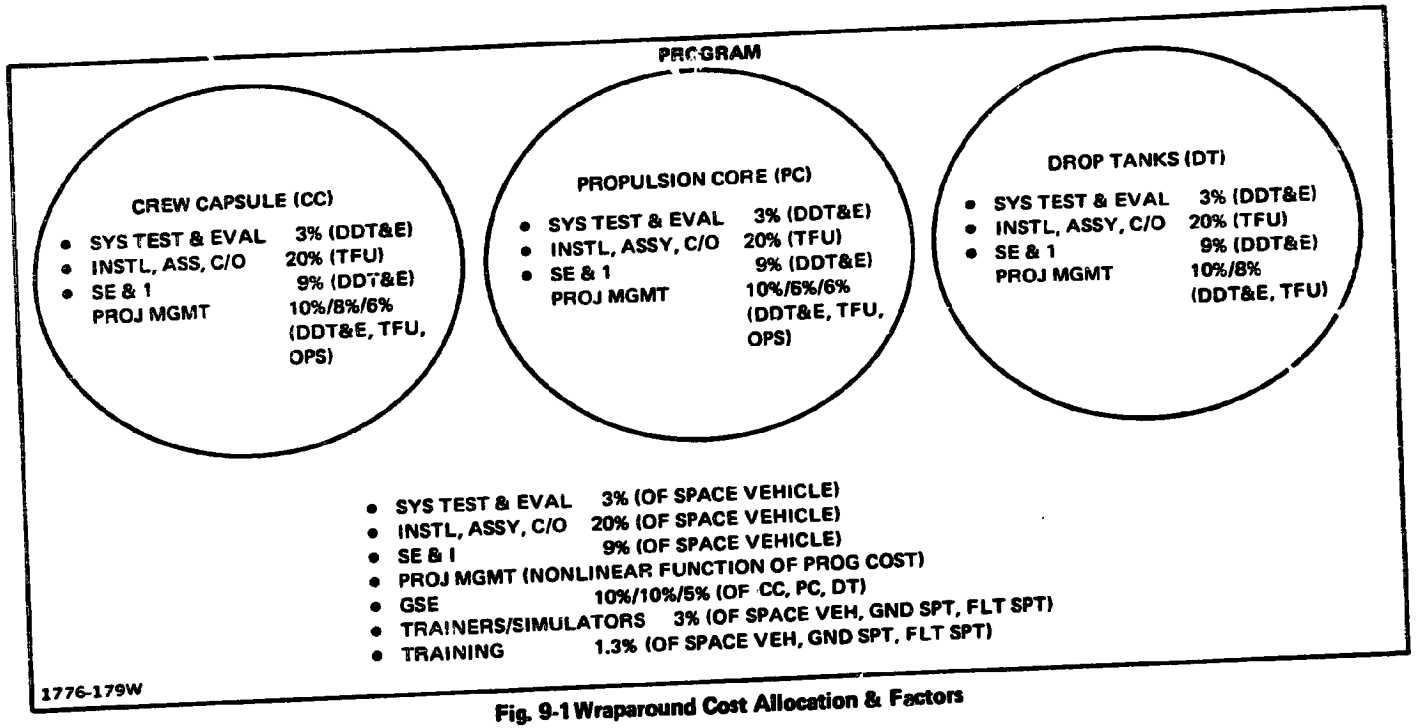

Fig. 9-1 Wraparound Cost Allocation & Factors

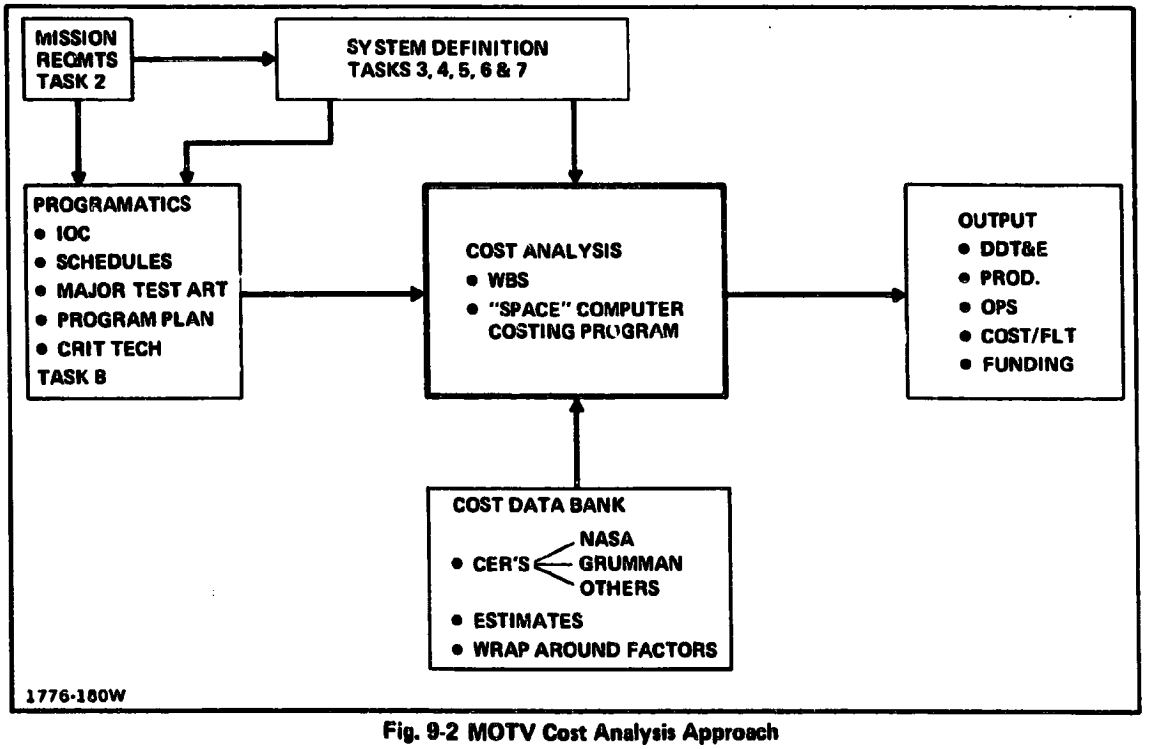

Fig. 9-2 MOTV Cost Analysis Approach

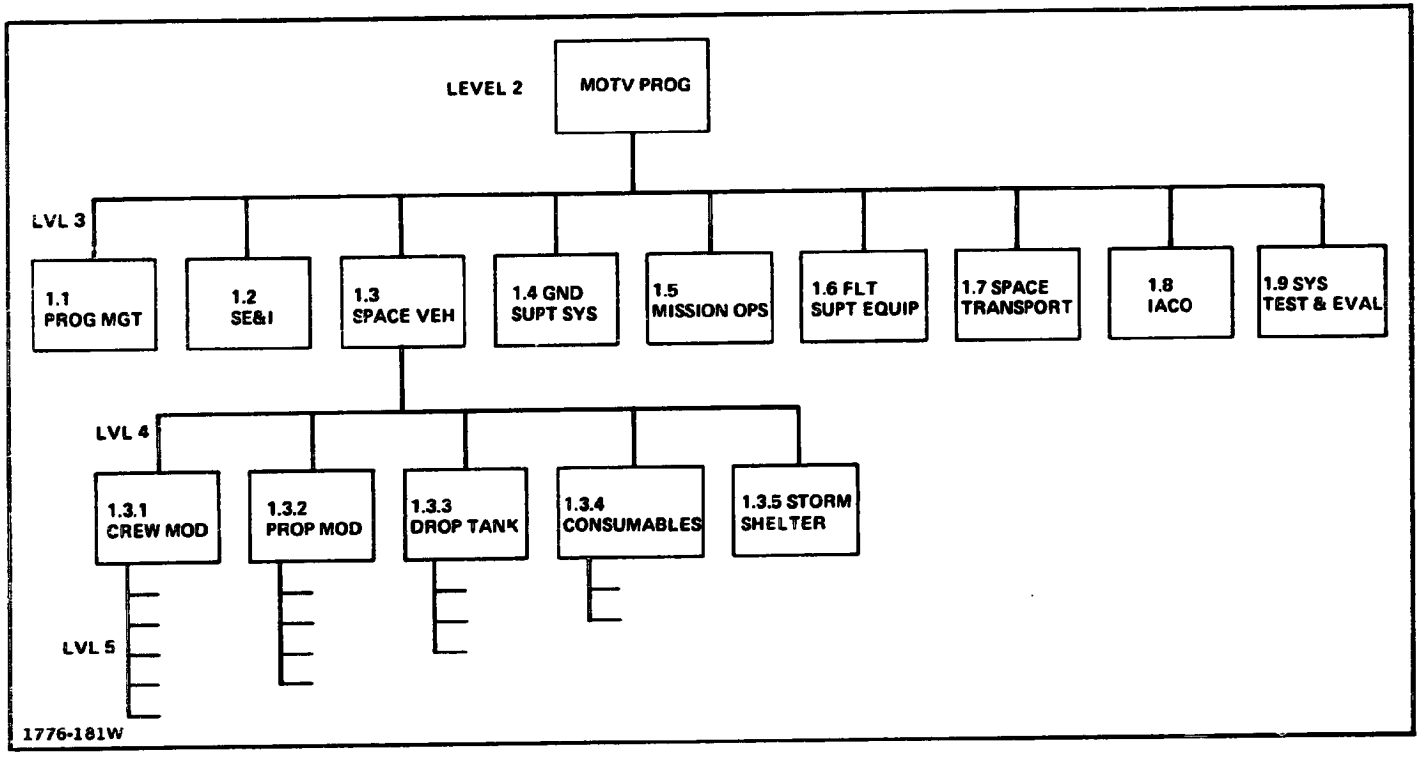

Fig. 9-3 MOTV WBS

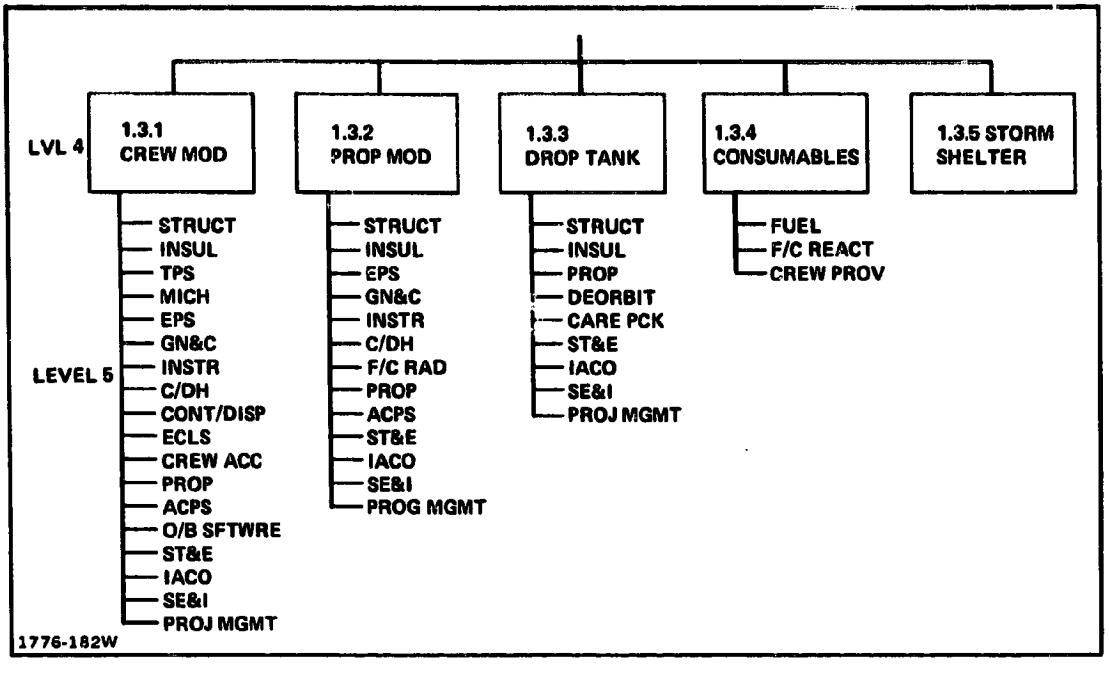

Fig. 9-4 MOTV WBS Detail

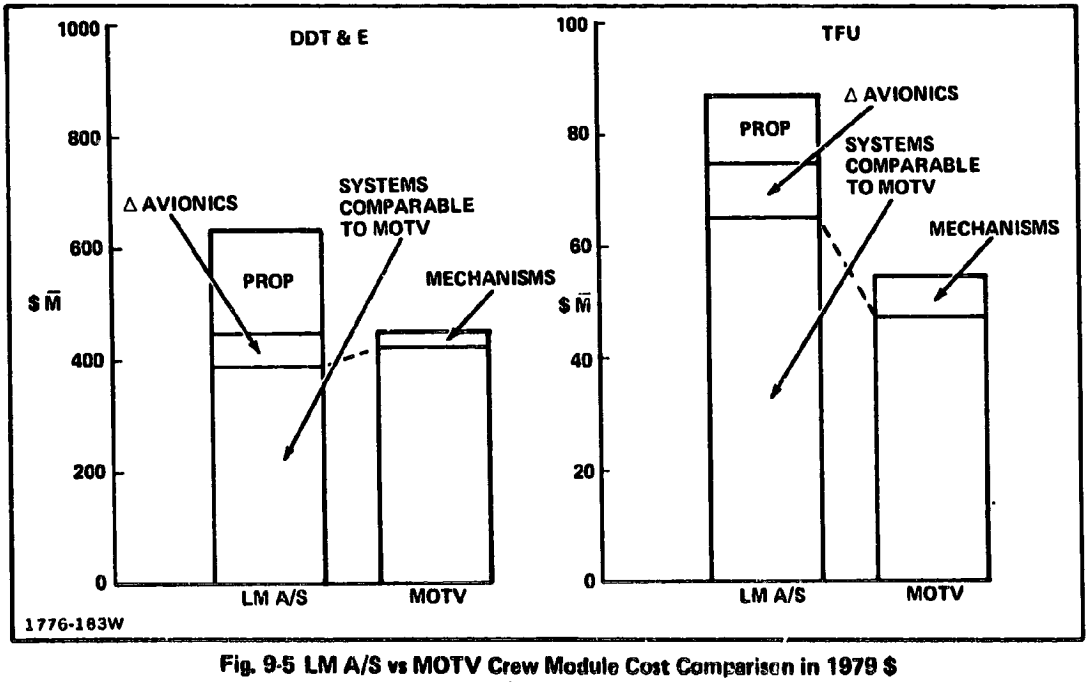

Fig. 9-5 LM A/S vs MOTV Crew Module Cost Comparison in 1979 $

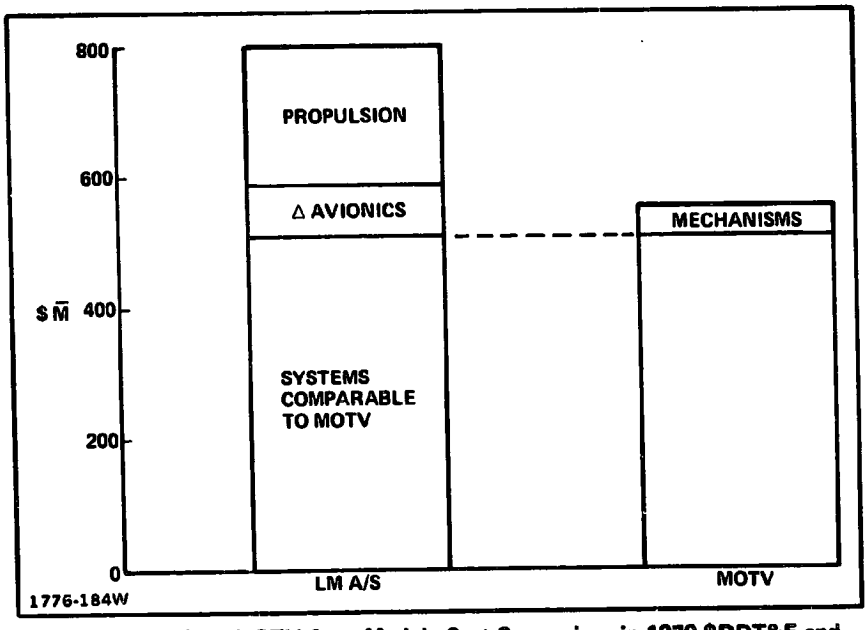

Fig. 9-6 LM A/S vs MOTV Crew Module Cost Comparison in 1979 $DDT&E and 2 Production Units

MOTV crew capsule for comparable sections of the vehicles. These two figures illustrate the viability of our MOTV costs by comparing them with an operational vehicle.

The weight-cost relationship for the MOTV Propulsion Module is presented in the bar graphs in Fig. 9-7. It shows clearly that weight is not always a driver of DDT&E cost. Avionics subsystem hardware, for instance, is relatively lightweight, but highly complex. Although its weight is a mere 5.6% of the total weight, its cost is over 25% of the total DDT&E cost. Our Cost Estimating Relationships (CER's) account for these sensitivities in the various subsystems.

The curve on the extreme right of Fig. 9-7 shows the basis for the MOTV structural CER.

Various crew capsules DDT&E costs plotted against capsule weight are presented in Fig. 9-8. When differences in subsystem requirements are adjusted for (e.g., entry and landing systems for the Mercury, Gemini, and Apollo crew capsules), the data still show considerable scatter. These costs variations are thought to be primarily due to differences in number of test articles, mission peculiar subsystem requirements, and subsystem reliability/redundancy/complexity for man-rating. Nevertheless, the MOTV costs estimates seem to be in the right ballpark compared to these other programs.

Costs for propellant tanks and the propulsion module are plotted in Figs. 9-9 through 9-11 as a function of usable propellant.

Following our costs studies, these observations can be made:

- Historical program costs show considerable scatter when plotted against vehicle weight; cost variations are primarily due to
  - differences in number of test articles
  - mission peculiar subsystem requirements
  - subsystem reliability/redundancy/complexity for man-rating
- Projected MOTV cost uncertainty are due to
  - standard error of estimate for CER's
  - errors in system definition
- Adjusted LM costs correlate well with projected MOTV crew capsule production costs and are within 15% of LM Ascent Stage, adjusted for differences in the number of test articles and mission peculiar subsystem requirements.

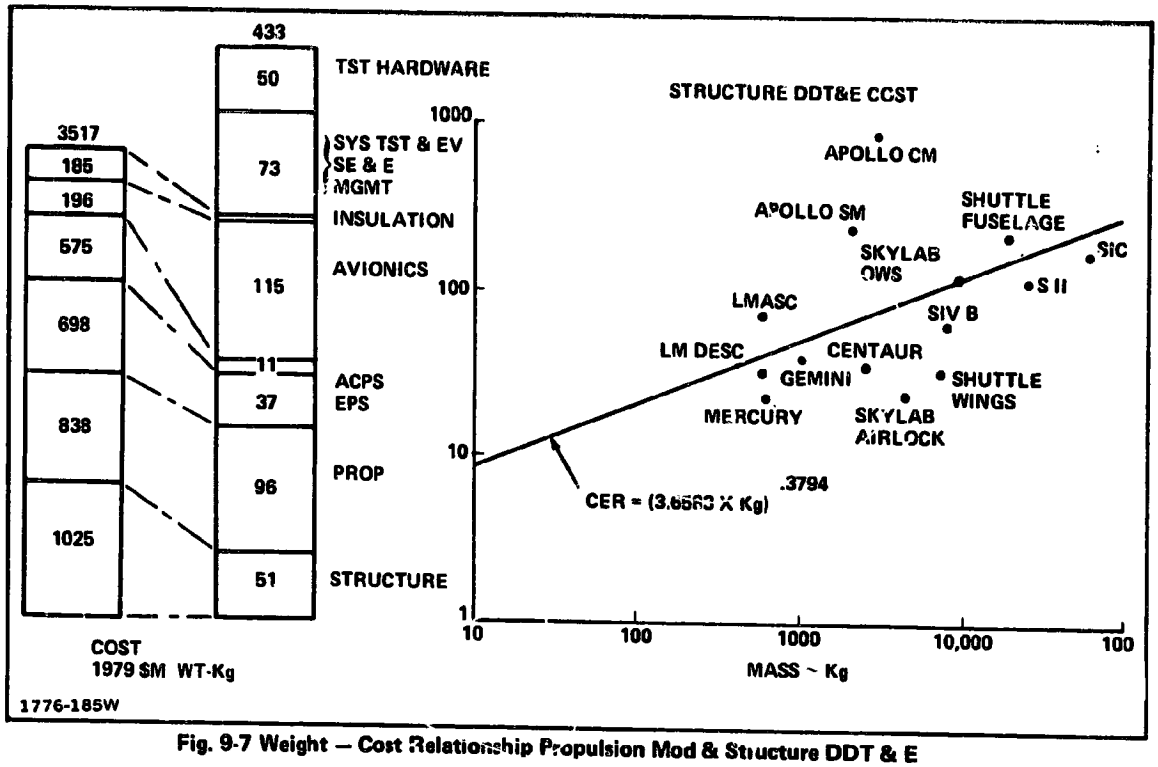

Fig. 9-7 Weight — Cost Relationship Propulsion Mod & Structure DDT & E

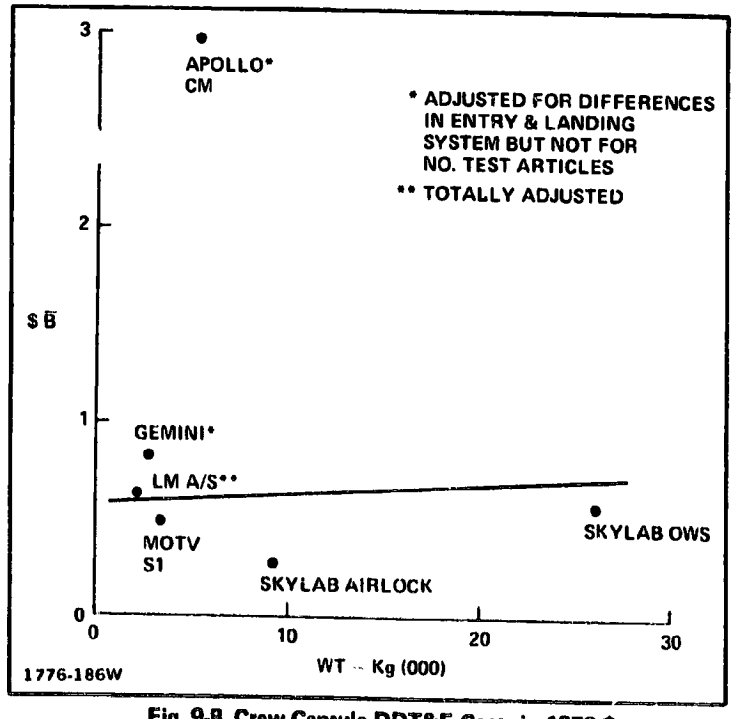

Fig. 9-8 Crew Capsule DDT&E Costs in 1979 $

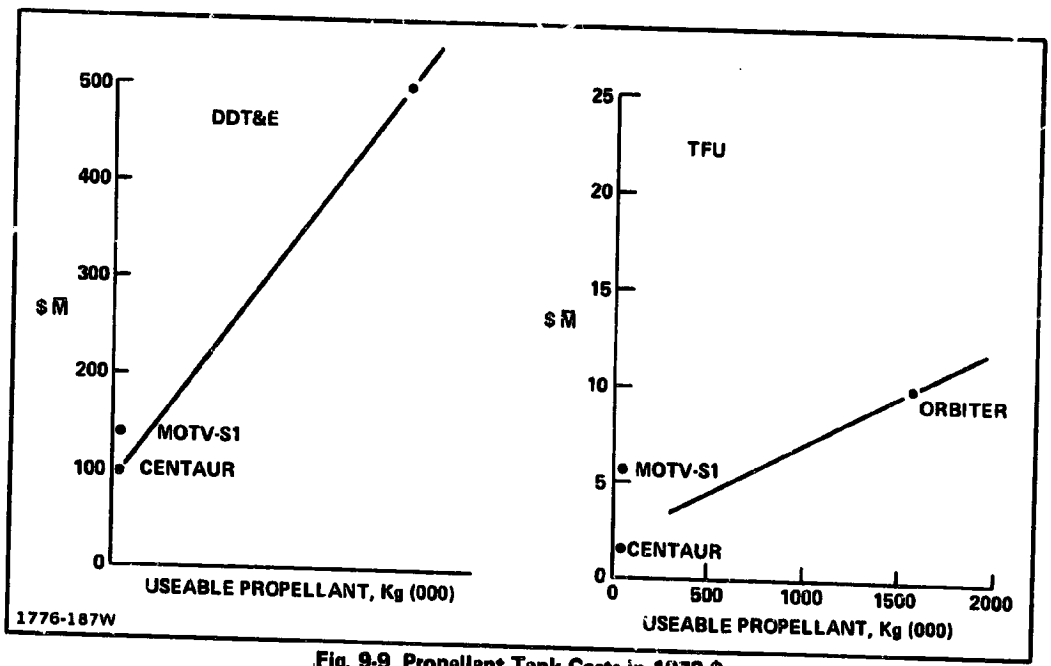

Fig. 9-9 Propellant Tank Costs in 1979 $

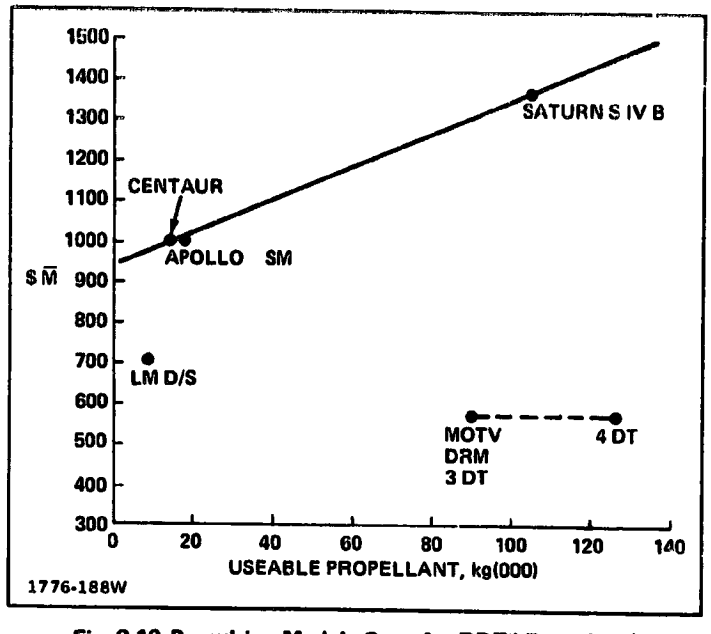

Fig. 9-10 Propulsion Module Costs for DDT&E in 1979 $

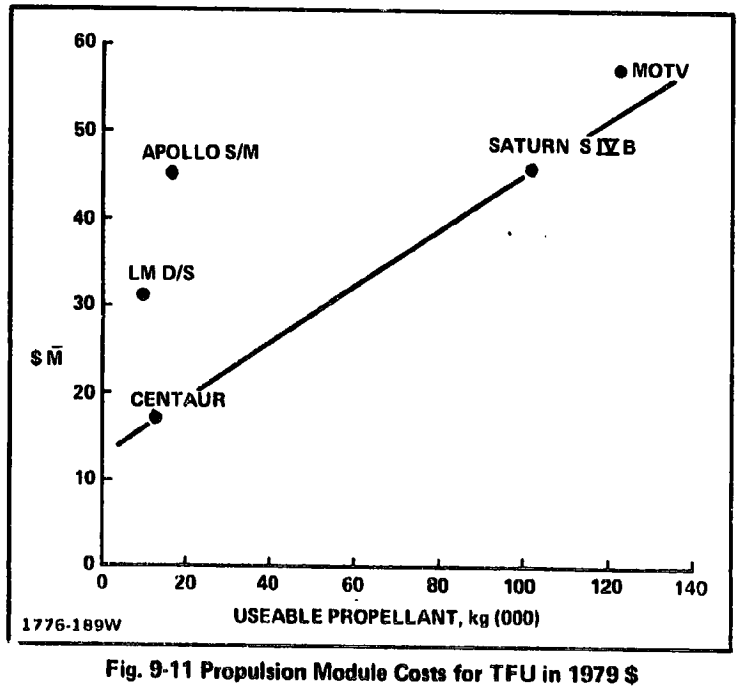

Fig. 9-11 Propulsion Module Costs for TFU in 1979 $