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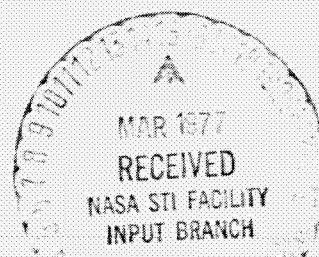
Lockheed

Missiles & Space Company. Inc.

HUNTSVILLE RESEARCH & ENGINEERING CENTER

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EXTENDED APPLICATIONS STUDY
OF AMOOS AND AMRS

FINAL REPORT

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FOREWORD

ACKNOWLEDGMENTS

SUMMARY

These studies have continued to show the potential advantages of the Aeromaneuvering Orbit-to-Orbit Shuttle (AMOOS) over the all-propulsive Orbit Transfer Vehicle (OTV). In particular, the kit concept studies and the dual fueled AMOOS studies have shown its versatility and option potential over the all-propulsive vehicle. All of this potential of AMOOS and the Aeromaneuvering Recovery System (AMRS) depends upon the ability to control the trajectory during atmospheric flight and so use an ablative TPS. In turn, this TPS must be light weight, which can be attained by spraying a lightweight ablator (e.g., Martin Marietta SLA 561) directly onto the load bearing skin. The significant findings of each subtask performed during the contract are summarized below.

AMOOS proved more readily adaptable to the kit concept than the Cryogenic Tug. In general AMOOS outperformed the all-propulsive kit AMOOS and the Cryogenic Tug. Furthermore, AMOOS may be readily adapted to the kit concept without payload penalty.

The ablative TPS is still preferred over those using reradiative and insulative materials. The ablator yields a lighter TPS with a higher temperature range. Other materials, except carbon-carbon, are restricted to low energy missions or multiple-pass maneuvers. The latter requires many passes through the Van Allen radiation belts.

The development of the space sextant will provide autonomous navigation capability. Other systems, such as the interferometer tracker and the landmark tracker, may also provide autonomous operation. Development is required of both systems.

The preferred AMRS configuration uses an expendable solid rocket motor. The dual recovery mode of operation is feasible and carries a penalty of approximately 10% of its dry weight. The dual modes considered are Shuttle rendezvous and ground recovery. Land impact is preferred for the ground recovery mode.

The dual fueled AMOOS has a sufficient round trip payload capability to rotate a four-man crew to geostationary orbit. Its round trip payload capability is approximately six times that of the all-propulsive, dual fueled vehicle.

Six-man through 18-man crew modules may be round tripped using a Growth AMOOS vehicle and Growth Shuttle. The 24-man module requires either a single stage Growth AMOOS or staged baseline AMOOS vehicles delivered to low earth orbit by the 130K Shuttle-derived High Lift Launch Vehicle (HLLV).

The payload performance of AMOOS with the Growth Shuttle and Shuttle derived HLLVs is greatly enhanced. The OTV performance using the Shuttle-derived HLLV is further enhanced by using two AMOOS stages. AMOOS requires considerable modification for ground recovery since, in its baseline configuration, it cannot perform a horizontal landing.

Space basing may yield small payload advantages; however, the potential increases are of the order of 15 to 20% in round trip payload capability. The use of Shuttle FPRs to further augment the OTV propellant is fraught with uncertainties and potential difficulties. More study is required; however, the current mode of operation of the Shuttle appears optimum so that making the FPRs available in low earth orbit would probably decrease the Shuttle payload performance.

The above results led to the following recommended tasks to continue the advancement of the AMOOS concept.

MODEL FLIGHT TEST PLAN DEVELOPMENT AND EVALUATION

Under this task a model flight test plan will be developed in detail and thoroughly evaluated. The overall task is divided into subtasks as discussed below:

● Identify Data Required to Meet Objectives

Detailed data requirements will be established such as loss of TPS due to ablation, navigation data, aerodynamic loads, etc.

● Identify Hardware Requirement

Hardware necessary to measure and record and/or transmit the data will be identified. Sensors by type and model will be identified, if possible, together with supporting hardware and power requirements.

● Determine Flight Test Trajectory

Flight test trajectories will be determined which yield the environment, spatial position and vehicle attitude necessary to gather realistic data.

● Model Conceptual Design

From the above requirements, the flight test model conceptual design will be developed. The objective of this design will be to develop the system in sufficient detail for a meaningful cost analysis.

● Shuttle and IUS Requirements

The use of the Shuttle and, if necessary, IUS propulsion stages will be determined. The impact on the Shuttle flight will be evaluated.

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● Identification of Alternatives

Identify several model configurations and modes of operation. Identify data that can be obtained from each configuration and mode of operation. In particular, identify small models that may be used in Shuttle tether tests.

● Cost Evaluation of the Model Flight Test

The data generated above will be used to estimate the cost of the flight test program. Effort will be made to identify the most cost effective method.

● Estimate Cost Effectiveness of AMOOS

The payload performance of AMOOS will be evaluated against AMOOS costs including model flight test costs. The resulting cost estimates may be used in evaluating AMOOS as an OTV. The cost estimates for AMOOS will be generated under various assumptions: for example, use of an existing RL10 or modified RL10 engine, or a new engine such as the ASE.

The output of this study will be a detailed model flight test plan, the cost of the proposed model flight tests and a cost effectiveness of AMOOS in the form of dollars per pound of payload. The Shuttle usage will be included in these studies since AMOOS can frequently do, in one Shuttle launch, tasks which require two Shuttle launches for the all-propulsive system.

SUPPORTING TECHNOLOGY

● Split Flap Studies

The AMOOS and AMRS body flap has been designed for longitudinal trim only. It could be used for pitch control and, if split, yield the possibility for aerodynamic roll control. The specific tasks would be similar to those of side flap studies for lateral control.

● Guidance

Guidance Development: The linear regulator guidance method can be
refined by two modifications to the performance index. The first is to re-
place the bank angle term by the bank angle acceleration. This will result
in a reduced attitude thruster fuel consumption. The second modification
is to include a combination of position and velocity at atmospheric exit in
the performance index. The purpose of this modification is to minimize
the variation of the phasing time with the Shuttle.

Manual Guidance: The possibility of pilot interaction with the guidance
system is a requirement during normal operation as well as a fail safe mode
in case of a massive failure. For the normal operating system several levels
of interaction between pilot and guidance system will be identified and analyzed.
These levels of interaction will include a supervisory mode, an active inter-
action mode and a manual mode. A fail safe manual mode will be developed
to be used in case of a massive failure. A guidance technique developed
earlier and entitled "Velocity Lost Approach" may be suitably modified for
that purpose.

● Navigation Studies

The initial phase of the navigation studies consists of determining the
effects of navigational accuracy on atmospheric flight guidance and phasing
with the Shuttle. The results thus obtained will be evaluated against accept-
able phasing orbit variations to yield acceptable navigation errors. The
study will include exoatmospheric navigation (midcourse correction) as well
as navigation during the atmospheric flight. The established navigation error
budget will then be used to evaluate existing hardware, define required or de-
sirable technology and compare with that required for the Baseline Space Tug.
The end result will be a practical set of navigational accuracies, navigational
hardware and desired or required technology.

- ● AMOOS Structure — First Year

Test panels of typical integral stiffened structure designed for the
AMOOS shell will be fabricated from candidate metallic and/or non-metallic
materials. The panels will be approximately 50 x 50 cm (20 x 20 in.). Various
TPS materials will be applied to the panels and thermocouples attached. Panels
will be cycled through a typical mission environment and thermal distribution
recorded. After testing, specimens will be examined for bond line and TPS
failure and possible damage to the stiffened panel. Refurbishment of the TPS
on the panel will be performed and tests repeated.

- ● AMOOS Structure — Second Year

The most promising configuration from the preceding test series will
again be fabricated and tested with both the thermal and mechanical load
applied simultaneously. Strains, deformations and temperature distribu-
tions will be recorded. The panels will be cycled through a typical mission
environment with the corresponding thermal and mechanical loads applied.
Panels will be examined after each test for TPS and structural failure.

- ● Wind Tunnel Testing

This task is divided into aerodynamic heating and aerodynamic force
and moment tests. These are discussed separately as follows:

Aerodynamic Heating Tests: The objectives of these tests are: (1) to
determine lee side heating rates for a range of angles of attack; and (2) check
the predictive method for estimating heating rates and temperatures. An
AMOOS Stycast model will be used for these tests. The heating rate will be
determined using temperature sensitive (Tempilaq) paint. Side and bottom
view movies at speeds of 16 frames/sec will be taken of the model. Shadow-
graphs will be taken at 10 deg angle-of-attack increments for every run.

The task will include an evaluation of test facilities against test require-ments. A test facility will be chosen on the basis of meeting test objectives, cost effectiveness and availability.

Aerodynamic Force and Moment: These tests will be performed using existing and modified models to determine: (1) the effect of Reynolds number on the aero forces and moments; (2) the effects of various flap configurations; and (3) forces and moments at Mach numbers closer to flight values than those of tests conducted under a previous contract.

● Lateral Control Using Aerodynamic Surfaces

This study would be spread over two years. In the first year the use of side flaps would be studied for lateral control of the AMOOS and AMRS vehicles, whereas in the second year the trades among the various flap options and RCS lateral control, or a combination of each, would be studied.

First Year: (1) Establish flap planform options and locations on vehicle; (2) Size flaps and compute forces and moments due to flaps; (3) Perform a preliminary design of flap structure, attachment, actuating mechanism, and TPS; and (4) Perform weights analysis of flaps and related subsystems.

Second Year: (1) Perform trade studies between weight added for flap and related subsystems and weight removed through the reduced RCS require-ment; (2) Determine the differences in the guidance and control requirements for the flaps and RCS, and establish trades; (3) Determine the effect of flaps on the vehicle structure and establish trades; and (4) Evaluate flaps in com-parison with RCS for lateral control during atmospheric flight.

● External Geometry Optimization

The weight and lateral maneuvering of the AMOOS and AMRS designs are dependent upon the external geometry. The current design was selected from a family of shapes yielding a high drag coefficient with little regard for

the attendant lift coefficient. The vehicles were selected to fly at a 45 deg angle of attack. Limited trade studies and previous results have shown that the TPS weight is both external geometry and angle-of-attack dependent. Furthermore, a higher L/D ratio would yield more lateral maneuverability as well as more trajectory control. In this task it is proposed to investigate the effects of higher L/D and external geometry on vehicle performance and so optimize the external geometry, flight attitude and mode of operation during atmospheric flight.

● Optimum Dual-Fueled Operation

In this study, two modes of operation would be considered. One mode would be applicable to the short on-orbit lifetime vehicle and one to the long-lifetime vehicle. In each case an initial high-density fuel burn is followed by a cryogenic fuel burn to achieve mission orbit. After completing its mission the short-lifetime vehicle would use cryogenic propellants to return, whereas the long-lifetime vehicle would use space storable propellants. The optimum Δv values for each propellant would be determined using dry weight minimization and payload maximization as criteria.

DESIRABLE NEW TECHNOLOGY

● Lightweight, Recyclable TPS

The requirements for a lightweight, recyclable TPS material can be established from the AMOOS and AMRS thermal environments. The development of a material with a temperature range equal to that of Carbon-Carbon, recyclable at least 20 times and with a density of not more than that of LI-900 would greatly enhance the operation of an aeromaneuvering OTV. Such a material would have applications to a wide range of vehicles including the Growth Shuttle, SPS launch vehicles and planetary probes.

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STUDY RESULTS

The results of the Extended Applications Study of the Aeromaneuvering Orbit-to-Orbit Shuttle (AMOOS) and the Aeromaneuvering Recovery System (AMRS) are here summarized and discussed in more detail than in the opening paragraphs of this summary.

The kitting option studies have shown that AMOOS may be used effectively in an all-propulsive mode. The kit consists of locating a ring at the LOX tank and designing the structure to part at this point. The ablator is not sprayed on for the all-propulsive mode of operation. The resulting vehicle has a dry weight of 2359 kg (5200 lb) for the AMOOS-derived vehicle as compared to 2336 kg (5150 lb) for the cryogenic space tug. The closeness of these dry weights ensures little differences in payload performance up to the point of tankage limitation. The propellant capacity remains at 22,000 kg (48,500 lb) for the AMOOS-derived vehicle as compared to 23,133 kg (51,000 lb) for the cryogenic tug.

The payload advantage of aeromaneuvering increases with increasing mission altitude but decreases with increasing plane change. At zero degree plane change, aeromaneuvering vehicles outperform the all-propulsive vehicle to all mission altitudes above 900 km (500 n.mi.). At 28.5 degrees plane change the changeover altitude has increased to 7000 km (approximately 4000 n.mi.). If the mission includes a return payload as well as a delivery, then the changeover altitude is reduced since return payload does not penalize the delivered payload capability for aeromaneuvering vehicles to the extent it does for all-propulsive vehicles.

The TPS studies enhanced the choice of an ablative TPS over the available options. Carbon-Carbon is the only material capable of withstanding the peak temperatures of some 2000 K (3200 F) of the one-pass mission. Increasing the number of passes or decreasing mission altitude has little appeal for practical missions. The number of passes must be increased

to 20 for the geostationary mission or the mission altitude decreased to 10,000 km (5500 n.mi.) for the one-pass maneuver for other materials to be usable. Even if such restrictions were acceptable, a weight penalty of some 2000 kg (4400 lb) would be imposed, which significantly reduces the performance on most missions. A cursory study was performed on a heat pipe TPS and one consisting of boiling off liquid hydrogen. Neither system showed promise when applied to AMOOS or AMRS. In the case of the heat pipe, the average heating rate requires the pipes to be made of a superalloy, resulting in an unacceptable weight penalty. The weight of liquid hydrogen required exceeded the weight of the ablator even under the most favorable assumptions for the system. To be added to the weight of LH_2 is the tankage, plumbing, surface heat exchangers and pumps. It appears that what is required to compete with an ablator is a material with properties similar to LI-900 but with a temperature range to 2000 K (3200 F). Such a specification is suggested as a goal for TPS technology advancement.

The navigation studies revealed that the space sextant can provide sufficiently accurate position data to allow an autonomous navigation system for AMOOS and AMRS. The space sextant is currently under development and should be available in the next decade. Other systems, such as the landmark tracker, have undesirable features, are insufficiently developed or require a remote cooperative component. The space sextant would be used in conjunction with a star tracker and possibly a horizon sensor to update a hexhead IMU.

The AMOOS and AMRS option studies revealed that the optimum approach to AMRS was probably a solid motor vehicle which could be recovered by parachute to a land impact. The dual-fueled AMOOS appeared to be a viable single vehicle option to the baseline AMOOS-AMRS system. A possible alternative to the dual-fueled AMOOS is one with a wide range mixture ratio engine so that a considerable loss of LH_2 while on station does not degrade its performance to a measurable degree. Also under this task, 6-, 12-, 18- and 24-man capsules were studied. These required approximately

65, 80, 100 and 110K Growth Shuttles to provide sufficient performance from single-stage AMOOS vehicles for a round trip geostationary mission.

The advanced mission applications studies showed that AMOOS retains its performance advantage over the all-propulsive vehicle for the 80 and 100K Growth Shuttles and the 130 and 160K Shuttle derived HLLVs. For use with the latter a horizontal landing manned unit capable of transporting a five-man crew was considered. This unit would also house the main propulsion system with the propellant carried in a non-recoverable external tank. This horizontal landing vehicle may be carried in the baseline Shuttle cargo bay. Typically, the AMOOS vehicle could round trip approximately 4200 kg (9000 lb) and 5400 kg (12,000 lb) to geostationary orbit when delivered to 300 km (160 n.mi.) low earth orbit by the 80 and 100K Growth Shuttles, respectively. In each case a single stage AMOOS would be used. For the 130 and 160K Shuttle-derived HLLVs, two-stage AMOOS vehicles are recommended. The appropriate round trip capabilities are approximately 800 kg (20,000 lb) and 11,300 kg (25,000 lb), respectively. These performances cover the six-to-24-man capsules considered under the AMOOS/AMRS task.

The result of the space basing studies was generally negative toward space basing. The potential savings, namely the dry weight of the OTV, are such a small percentage of the Shuttle payload that there is not much payload on which to trade. The payload gains may be as small as 1% to low earth orbit and as high as 20% to geostationary. The use of Shuttle flight performance reserves was also considered. This study showed that the current mode of operation of the Shuttle was near optimum if the external tank is to be de-orbited. Only a fully loaded Shuttle was considered. If the Shuttle is carrying other than its full payload capability then it could possibly carry the balance in fuel for off-loading in space.

CONTENTS

NOTE ON UNITS

Throughout this report, the S.I. units are used followed by the conventional units in parentheses. Where convenient, the unit used is not necessarily the prime S.I. unit; e.g., skin thicknesses are given in millimeters and heat loads are given in Joules per square centimeter rather than in meters and Joules per square meter, respectively. The distinction is, of course, made between units of mass, kilograms, and units of force, Newtons. However, where particular word usage is deeply engraved and specific technical meaning is implied, such usage has been retained even though strictly incorrect; thus, the reader will see weights analysis of the vehicle mass and its weight in kilograms. Such inconsistencies are no worse than the use of velocity for speed or mixing notions such as in periapses.

Section 1
INTRODUCTION

The Aeromaneuvering Orbit-to-Orbit Shuttle (AMOOS) and the Aero-maneuvering Recovery System (AMRS) have been the subject of previous studies by Lockheed-Huntsville. These studies were reported by Andrews (Ref. 1) and White (Refs. 2 and 3). The objectives of these studies were to establish: (1) the general feasibility of the AMOOS concept; (2) the feasibility of specific configurations selected from those generated in the general feasibility studies; (3) the modular AMOOS configuration and performance; (4) the use of aeromaneuvering for crew transportation (AMOOS) and emergency crew return (AMRS); (5) the tractability of AMOOS and AMRS to guidance during atmospheric flight; and (6) the usefulness of a model flight test plan.

The emphasis during the performance period covered by White (Ref. 3) was on manned modular concepts, whereas, during the period covered by Andrews (Ref. 1) and White (Ref. 2) the emphasis was on unmanned integral vehicles. The performance analysis included single stage, stage and one-half and two-stage AMOOS vehicles. The all-up weights of these vehicles was commensurate with both single and multiple launches of the baseline and uprated versions of the Space Shuttle.

Initially, consumables requirements, configurations and flight environment were determined. Based upon these results, modular and integral AMOOS vehicles and a crew module were designed to perform a round trip equatorial geostationary mission. Limited analyses were performed for missions to other orbits, including planetary.

Analyses similar to the above were performed for the AMRS vehicle which is, of course, a recoverable single stage vehicle.

Sufficient aerodynamic analysis was performed to establish the static stabili·y of the vehicles over a range of center-of-gravity locations and trim angles of attack. The trajectory analyses included a brief guidance study applicable to AMOOS and AMRS during atmospheric flight.

These previous studies were directed toward an alternative to the Baseline Tug and, therefore, its missions were emphasized. Since these studies, the emphasis has changed. It is now on Space Station support. These missions are considerably different from the tug-type missions considered previously. In particular, the crew size is considerably larger (up to 24) and very heavy structures must be placed in orbit. Also not included were such important topics as operating AMOOS and AMRS in the purely propulsive mode (kit concept), determining the ranges of applicability of re-radiative and insulative thermal protection systems, hybrid engines, navigation hardware requirements, and for AMRS, ground recovery subsystems, dual modes of recovery and solid motor propulsion.

Further studies in these and other related topics were required so that AMOOS and AMRS data may be usefully applied to the current Space Station concept.

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Section 2
KIT CONCEPT

Studies previously reported (Ref. 1) showed that the payload advantage of AMOOS over the Cryogenic Tug was a function of mission altitude for the three types of missions studied. These mission types were round trip, payload delivery only and payload retrieval only. Since the Cryogenic Tug showed advantages over AMOOS on low earth orbital missions then so would an AMOOS stripped of its external TPS and operated in the all propulsive mode show an advantage over the aeromaneuvering vehicle. This advantage would, of course, be confined to low earth orbital missions as for the Cryogenic Tug. Furthermore, the performance of certain missions could require all-propulsive orbit transfers, e.g., when a large but lightweight structure is to be moved to a lower orbit, or where the vehicle is expended. The questions that arise are: (1) at what altitudes are the changeovers from aeromaneuvering to all propulsive, and (2) are modifications necessary to the AMOOS vehicle that will improve its overall performance. Typical modifications that may be considered are: (1) redesigning the AMOOS primary structure to allow more to be stripped in the all-propulsive mode, and (2) increasing the tank size to carry more propellant.

2.1 DRY WEIGHTS AND VEHICLE DESIGN

Two approaches to this task have been taken. The first approach was to strip AMOOS of the external TPS and readily removable structure and secondly to modify the Cryogenic Tug of Ref. 4 to give it aeromaneuvering capability.

These initial modifications are shown in Figs. 1 through 3. Figure 1 shows the minimum modification to AMOOS. In this configuration the hinged parts of the nose are removed and the ablative TPS (external) is not added.

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4.36 m (14.29')

9.57 m
(31.4')

3.94 m
(12.92')

4.36 m
(14.29')

3.35 m
(11')

3.66 m
(12')

3.66 m
(12')
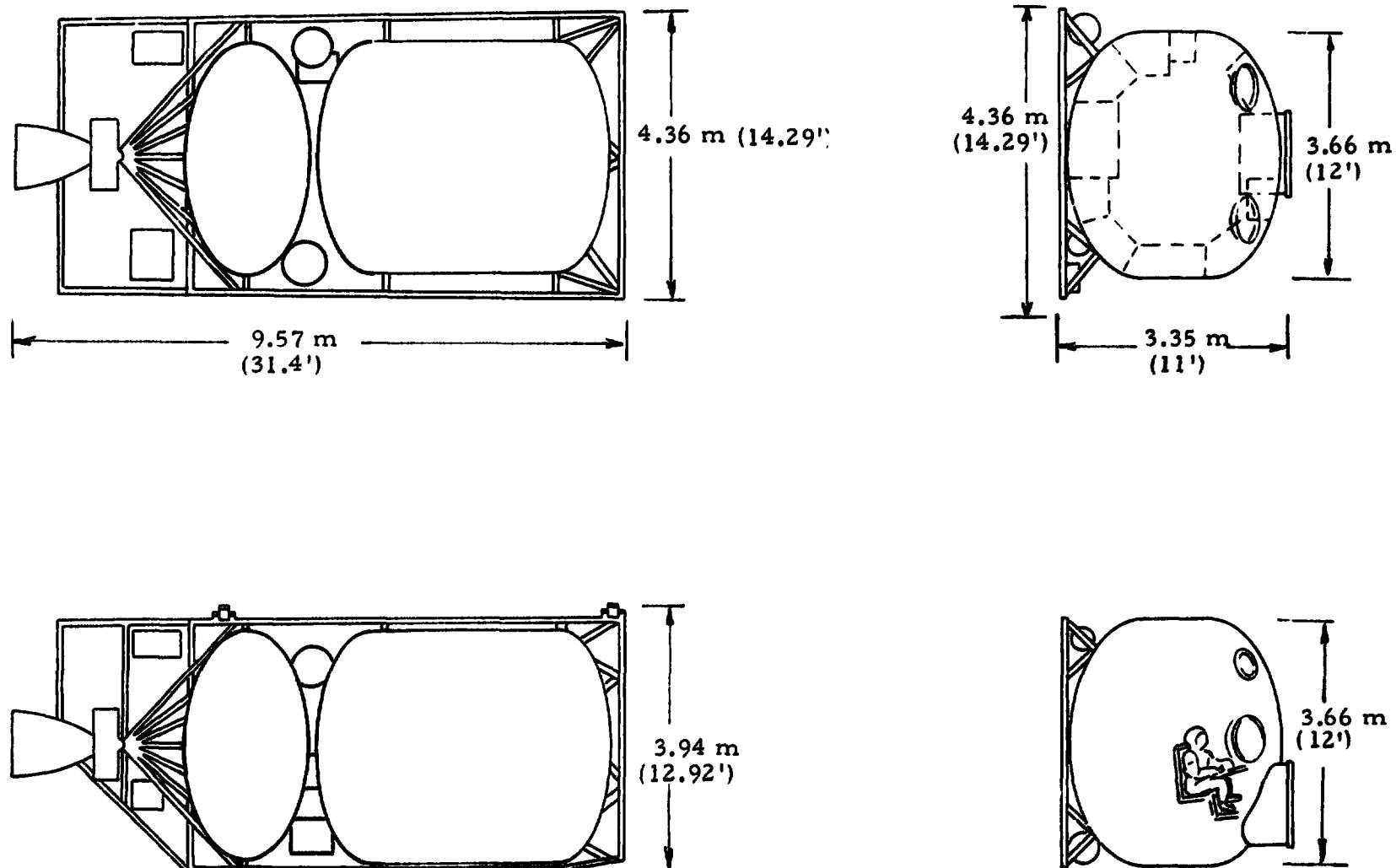
Fig. 1 - Minimum Modification Kit AMOOS and Manned Module

4.36 m
(14.29')

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(31.4')

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3.94 m
(12.92')
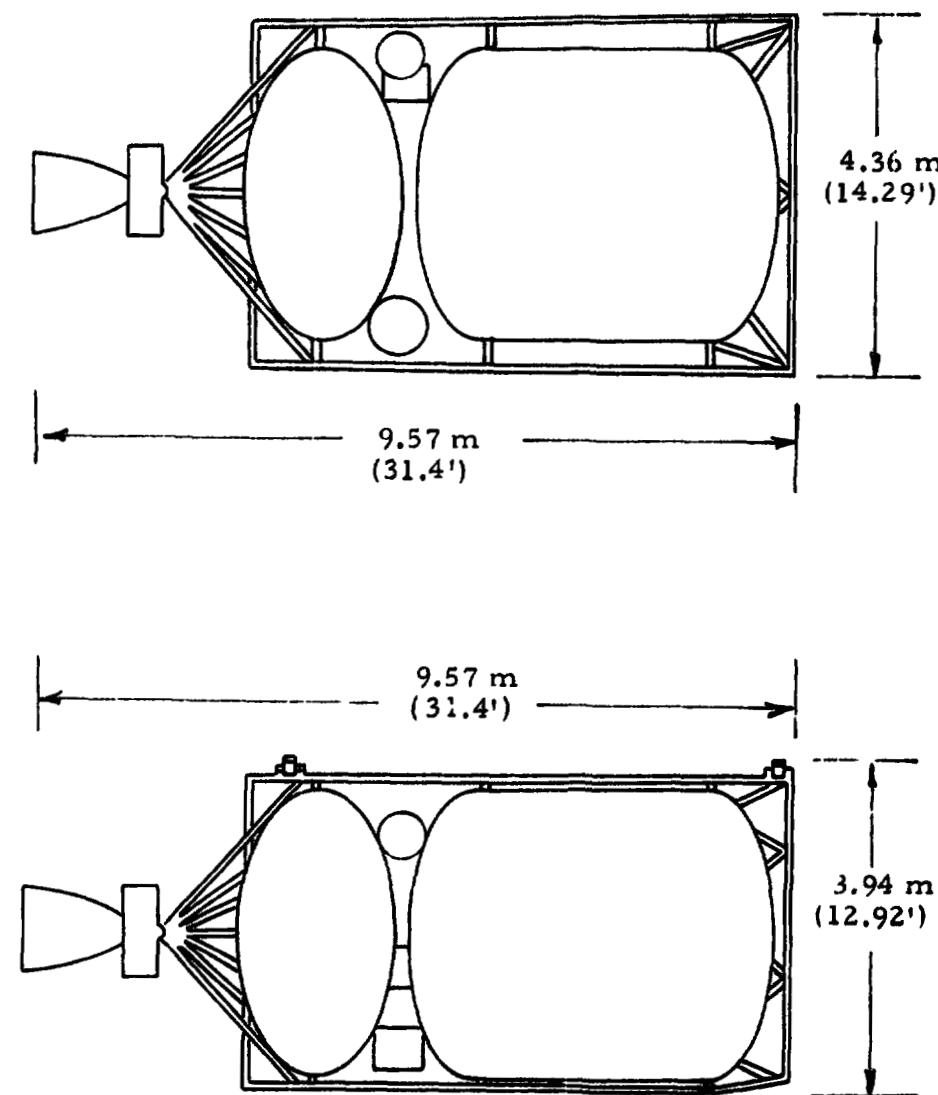
Fig. 2 - Maximum Modification Kit AMOOS Resulting in No Weight Penalty to the Aeromaneuvering
Vehicle
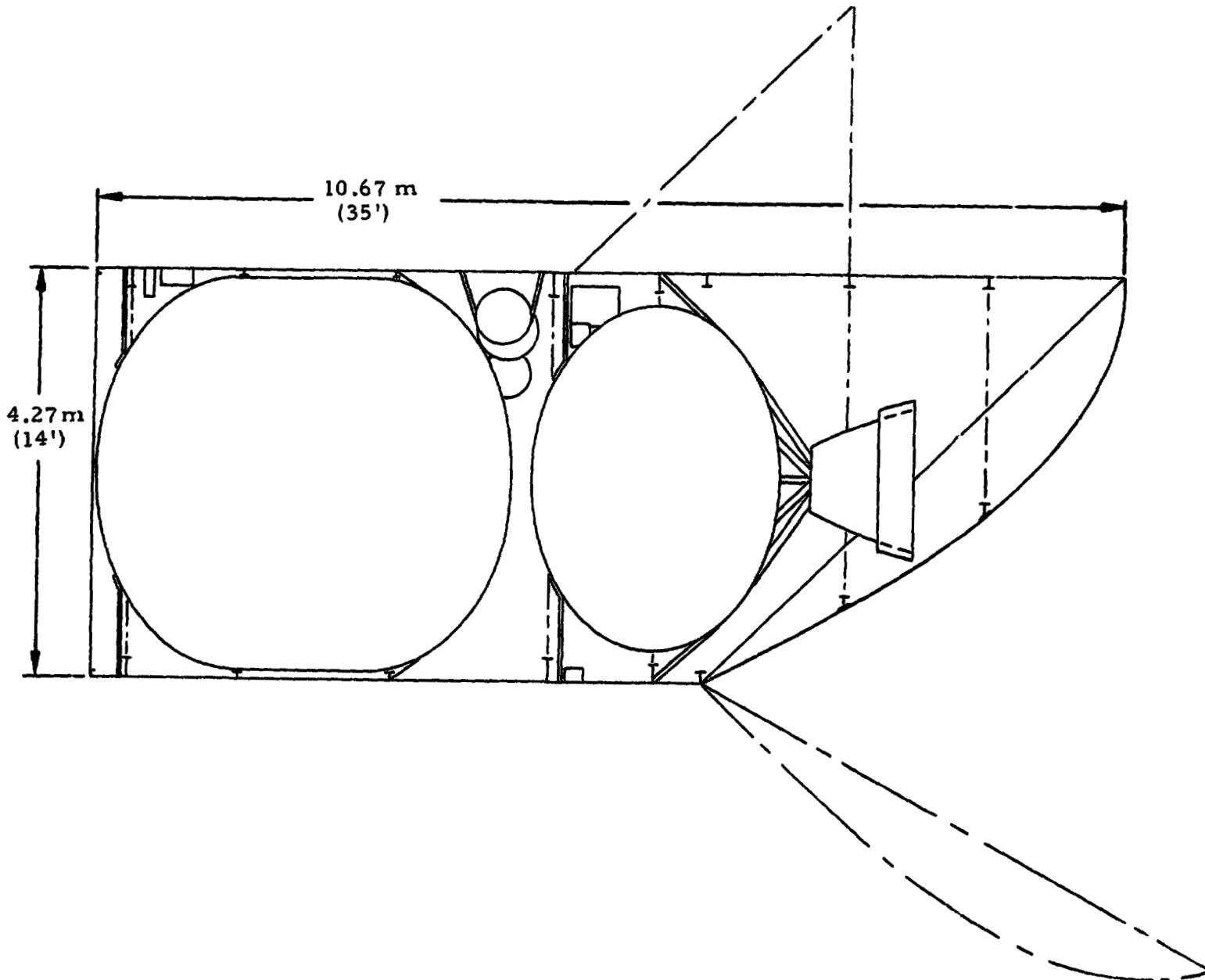
Fig. 3 - Aeromaneuvering Kit Modifications to the Baseline Cryogenic Tug

The body flap is also removed. The dry weight of the vehicle in Fig. 1 is 2427 kg (5350 lb) including the usual 10% contigency. The second step in the modification of AMOOS to the kit concept is to design the primary struc- ture forward of the oxygen tank to be removable. This concept is shown in Fig. 2. Removing this forward structure requires that a few black boxes, etc., be repositioned. This can be readily accomplished since the volume requirements for the electronic and electrical subsystems were intentionally overestimated in previous studies. Also, a primary structure ring must be positioned at the parting line. Such rings, (seven in all) were included in the weights analysis of Ref. 3, but the locations of those not at the nose hinge lines, aft end and attachment points (five total), were not specified. Choosing one at the parting point adds no weight to the AMOOS design. The dry weight of the kit vehicle in this configuration is 2359 kg (5200 lb). This is close, (22.7 kg (50 lb)) to the dry weight of the Cryogenic Tug, so that no further modification is considered necessary at this time. In comparing the kit AMOOS weights to the Cryogenic Tug, it should be recalled that the total AMOOS tankage is 22,000 kg (48,500 lb) of propellants compared to 23,133 kg (51,000 lb) for the Cryogenic Tug. Making allowance for this increased tankage would increase the dry weight of the AMOOS kit vehicle by some 45 to 68 kg (100 to 150 lb). The desirability of increasing the AMOOS tankage will be discussed later.

The AMOOS concept developed from the Cryogenic Tug is shown in Fig. 3. The modifications shown are: (1) an extension of the structure forward (AMOOS terminology) to beyond the thrust structure; (2) the addition of a hinged upper nose structure; (3) the addition of a hinged nose cap; and (4) the addition of an ablative TPS designed for a bondline temperature of 422 K (300 F). This bond- line temperature is consistent with the graphite/epoxy skins of the tug's pri- mary structure. The estimated dry weight for the AMOOS vehicle derived from the cryogenic tug is 3453 kg (7611 lb) including the usual 10% contingency.

The derivations of the dry weights from those of the basic vehicles are given in Tables 1 through 3. It should be recalled that the starting structural weights are for optimized design, and since the modifications are relatively

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Table 1

AMOOS KIT CONCEPT (FIG. 1) DRY WEIGHT ESTIMATE

Item Removed	Weight Removed	
	kg	(lb)
Nose Cap (Hinged)	39	(86)
One Ring	29	(64)
Upper Nose Section (Hinged)	25	(54)
TPS	470	(1036)
Net Total	563	(1240)
Contingency, 10%	56	(124)
Total	619	(1364)
AMOOS Dry Weight (Ref. 3)[*]	3039	(6700)
Less Wt. Removed	619	(1364)
Kit (Fig. 1) Dry Wt.	2420	(5336)

Use in performance calculations 2427 kg (5350 lb)[**]

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[*]Includes 90 kg (200 lb) of unbudgeted contingency.

[**]Weights rounded to nearest kg (lb) so that equivalences are approximate.

Table 2

AMOOS KIT CONCEPT (FIG. 2) DRY WEIGHT ESTIMATE

Item Removed	Weight Removed	
	kg	(lb)
Nose Structure Forward of Station 2.90 m (9.5 ft)	137	(301)
Structure from Station 2.90 m (9.5 ft) through at 3.81 m (12.5 ft)	25	(55)
TPS	470	(1036)
Net Total	632	(1392)
Contingency, 10%	63	(139)
Total	695	(1531)
AMOOS Dry Weight (Ref. 3)[*]	3039	(6700)
Less Wt. Removed	695	(1531)
	2344	(5169)

Use in performance calculations 2359 (5200)[**]

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[*]Includes 90 kg (200 lb) of unbudgeted contingency.

[**]Weights rounded to nearest kg (lb) so that equivalences are approximate.

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Table 3

CRYOGENIC TUG AEROMANEUVERING KIT (FIG.3) DRY
WEIGHT ESTIMATES

Item Added	Weight Added[*]	
	kg	(lb)
Nose Cap, Rings and Nose Structure	137	(301)
Actuators, etc.	50	(110)
TPS 422 K (300 F) Bondline (Structure) Temperature	828	(1826)
Net Total	1015	(2237)
Contingency, 10%	102	(224)
Total	1117	(2461)
Baseline Cryogenic Tug Dry Weight (Ref.4)	2336	(5150)
Weight Added	1117	(2461)
Total	3453	(7611)

[*]Weights rounded to nearest kg (lb) so that equivalences
are approximate.

Note: A preliminary estimate of the weight increase was made for the aero-
maneuvering kit for the Cryogenic Tug for use in the performance cal-
culations. This preliminary estimate was 3357 kg (7400 lb) which is
some 96 kg (211 lb) lighter than tne estimate above. Since at this
lighter weight, the performance, with aeromaneuvering, of the kit Cry-
ogenic Tug is inferior to AMOOS, the performance calculations were
not repeated for the relatively small change realized by the more
thorough weights analysis.

small fractions of the dry weight, the resulting weights are considered good
estimates, particularly in the case of the AMOOS-derived vehicles. Since
the TPS is the spray-on type, its inclusion or exclusion does not affect the
primary structure.

Modifications to the basic AMOOS vehicle are minor. They include
such changes as a minor rearrangement or positioning a ring allowed for in
the basic vehicle weight estimate but not positioned. The penalty for design-
ing AMOOS in a kit form is, therefore, negligible. The effect of designing
the Cryogenic Tug in a kit form was not estimated since its estimated dry
weight in the AMOOS configui .tion makes it non-competitive with the AMOOS
vehicle of Ref. 3.

At the time the AMOOS vehicle was being redesigned into kit form, a
comparable manned module was also redesigned. The resulting configuration
is also given in Fig. 1. The primary shell structure, TPS and body flap were
removed to yield an all-up weight of 2704 kg (5961 lb), a savings of 410 kg
'904 lb). The computation of the weight is given in Table 4. A further re-
design of the manned module was performed in Section 5.3, Dual Fueled
AMOOS. Such a modification could reduce the kit concept capsule dry weight
to approximately 2268 kg (5000 lb).

2.2 KIT PERFORMANCE EVALUATION

The performance of the kit concepts was evaluated over a range of alti-
tudes ranging from 1000 km (500 n.mi. approximately) through geostationary
and for two specific impulse values, namely 463 and 470 sec. These I_{sp}'s repre-
sent two stages of development of LH_2-LOX engines using conventional engine
layout technology or an advanced technology engine. In the layouts shown in Figs.
1 through 3, only conventional engines currently recommended for application
to the parent vehicle are shown. The engine configurations associated with
I_{sp} = 47C sec may or may not be comparable to the shown configurations, see
Ref. 5.

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Table 4

AMOOS KIT CONCEPT MANNED CAPSULE
DRY WEIGHT ESTIMATE

Item Removed	Weight Removed[*]	
	kg	(lb)
Primary Shell Structure (Less One Ring)	119	(262)
TPS	152	(335)
Body Flap	102	(225)
Net Total	373	(822)
Contingency, 10%	37	(82)
Total	410	(904)
Manned Capsule Dry Weight (Ref. 3)	3114	(6865)
Less Weight Removed	410	(904)
Total	2704	(5961)

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[*]Weights rounded to nearest kg (lb) so that equivalences are approximate.

The objectives of the performance analysis were: (1) to determine the crossover mission altitudes, above which AMOOS can transport more payload per Shuttle launch than the all-propulsive Cryogenic Tug or kit AMOOS, and (2) determine the design changes necessary to obtain the best compromise of performance over all mission altitudes from low-earth orbit to geosynchronous.

The performance of AMOOS, in the aeromaneuvering mode, was computed for circular mission orbits from 1000 km (500 n.mi.) through 8000 km (4000 n.mi.) altitude with a 28.5 deg plane change. The performance is plotted in Fig. 4 in the form of payload delivered versus payload retrieved for I_{sp} = 463 sec and propellant capacity of 22,000 kg (48,500 lb). The performance of AMOOS increases steadily with increasing altitude. This is due to the propulsive Δv decreasing with increasing altitude. The cause of this is the more efficient plane change at the higher altitudes. The performance is for two burn transfers. Three burn transfer Δv values were computed but showed negligible decrease in propulsive Δv requirement for 28.5 deg plane change. The delivery-only payload (AMOOS recovered) peaks at approximately 7000 km (3800 n.mi.) mission altitude (circular, 28.5 deg plane change). The round trip capability increases to a maximum at approximately 9000 km (5000 n.mi.) altitude. This round trip capability remains practically constant to geostationary altitude (Ref. 3). The delivery-only payload decreases slightly from the maximum at 7000 km (3800 n.mi.) to about 5200 to 5400 kg (11,500 lb – 12,000 lb) at geostationary. In the payload retrieval only case, the payload increases steadily to approximately 9000 km (5000 n.mi.) and then more slowly to geostationary. No maxima, in the mathematical sense occurs, for the payload retrieval case.

The performance of the aeromaneuvering kitted Cryogenic Tug is shown in Fig. 5 for the corresponding cases to the performance of AMOOS shown in Fig. 4.

The general trend is similar, however, the delivery-only case reflects the higher dry weight of the kitted Cryogenic Tug, namely, 3357 kg (7400 lb)

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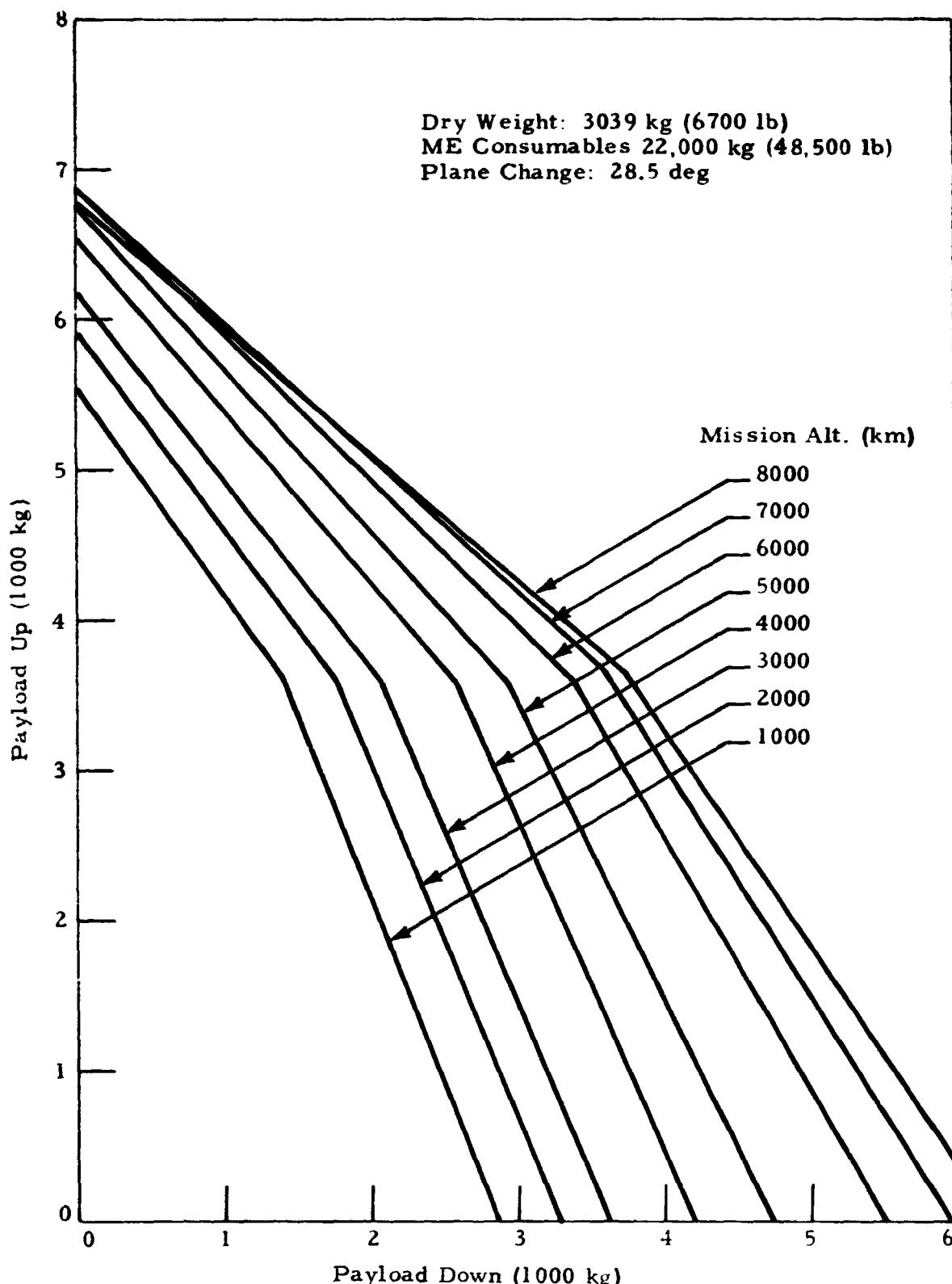
Fig. 4 - AMOOS Payloads to Low Earth Orbits, I_{sp} = 463 sec
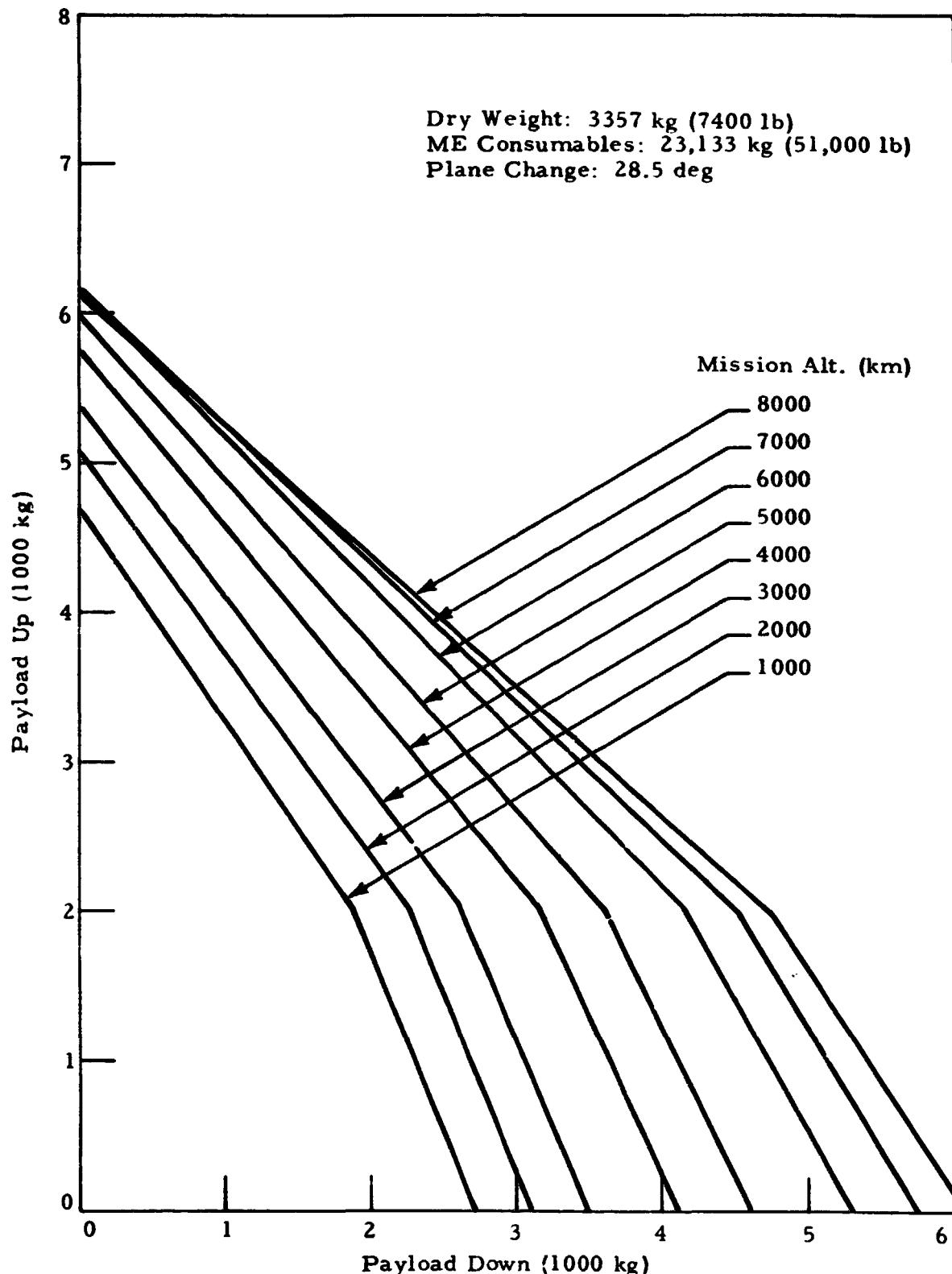
Fig. 5 - Aeromaneuvering Kit Cryogenic Tug Payloads to Low Earth Orbit, I_{sp} = 463 sec

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versus 3039 kg (6700 lb) for AMOOS by a reduced delivery capability. The round trip capability also reflects the increased dry weight by a reduction of approximately 318 kg (700 lb) in payload. On the other hand, the retrieval capability reflects the increased propellant capacity by a small increase in retrieval-only capability. The payloads to mission altitudes above 8000 km (4300 n.mi.) display the same trends as those for AMOOS. The delivery and round trip payloads continue to reflect the increase in dry weight of the kitted Tug, whereas the retrieved payloads reflect the increase in propellant capacity.

The analysis displayed in Figs. 4 and 5 was also performed for an I_{sp} of 470 sec. The dry weights and missions were unchanged. The results for AMOOS are shown in Fig. 6. As expected, the results show a small overall increase in payload capability. The corresponding cases for the kitted Cryogenic Tug are shown in Fig. 7. Again a small, but general increase in payload capability is shown.

The payload capabilities of the all-propulsive versions of AMOOS and the Cryogenic Tug were also computed for the preceding range of mission altitudes. The propulsive Δv requirement for the all-propulsive case differs from that of the aeromaneuvering case ir that, on the return to low earth orbit part of the plane change can be performed at the perigee burn. In the case of the aeromaneuvering vehicles, all the plane change on the return phase of the mission must be performed at the apogee burn. This is because the aeromaneuvering plane change is neglected for low energy missions (see Ref. 1). Because the plane change Δv is apportioned significantly between the two return transfer burns, the total Δv decreases as mission altitude increases to approximately 3250 km (1750 n.mi.) (two-burn transfer). From 3250 km (1750 n.mi.) to beyond geostationary altitude the total Δv increases with increasing altitude. At the lower mission altitude range, the plane change Δv controls the trend whereas at the higher altitude the Δv required to change altitude controls the trend.
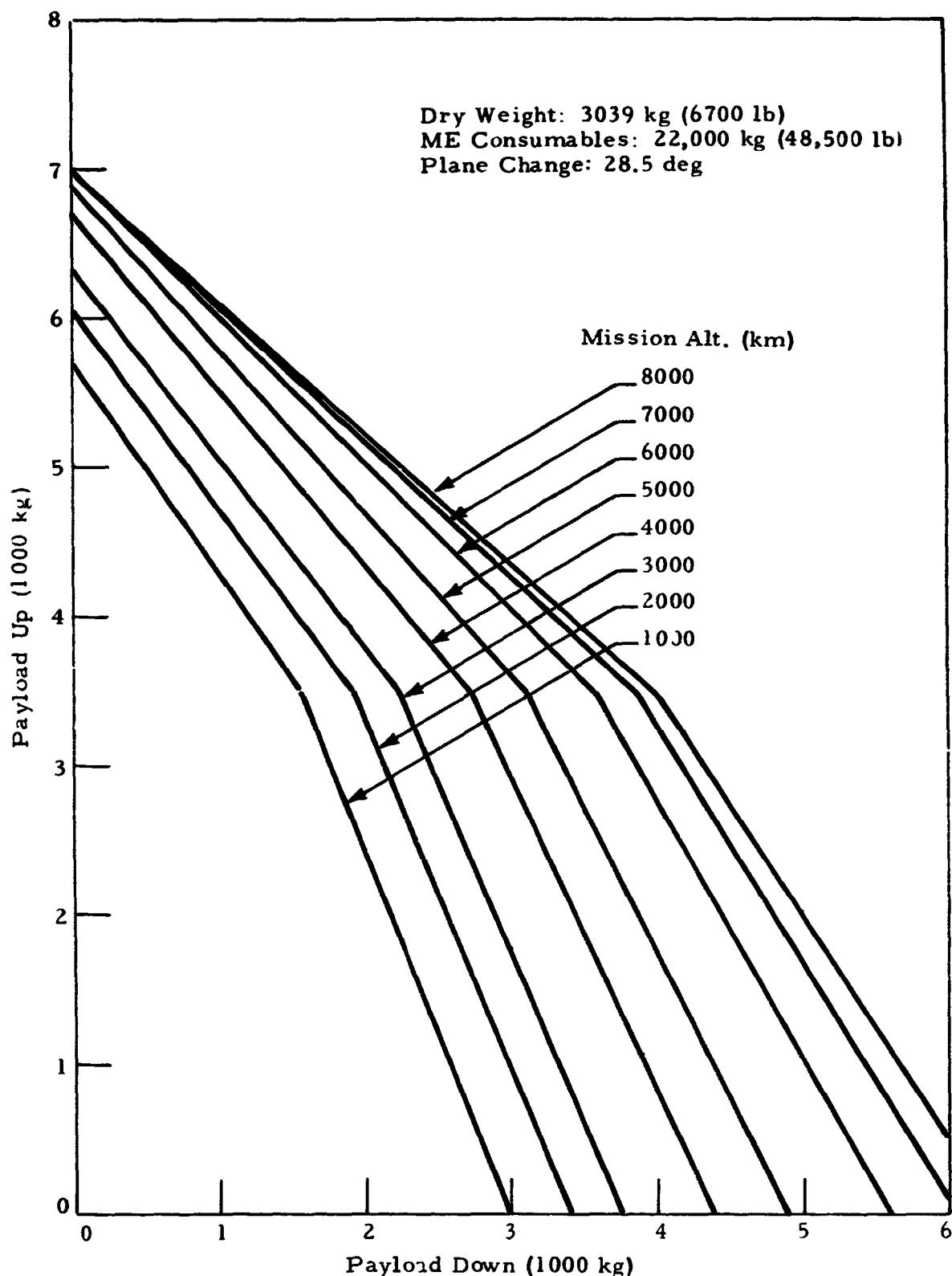
Fig. 6 - AMOOS Payloads to Low Earth Orbits, I_{sp} = 470 sec

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Dry Weight: 3357 kg (7400 lb)
ME Consumables: 23,133 kg (51,000 lb)
Plane Change: 28.5 deg

Mission Alt. (km)

8000
7000
6000
5000
4000
3000
2000
1000

Payload Up (1000 kg)

Payload Down (1000 kg)
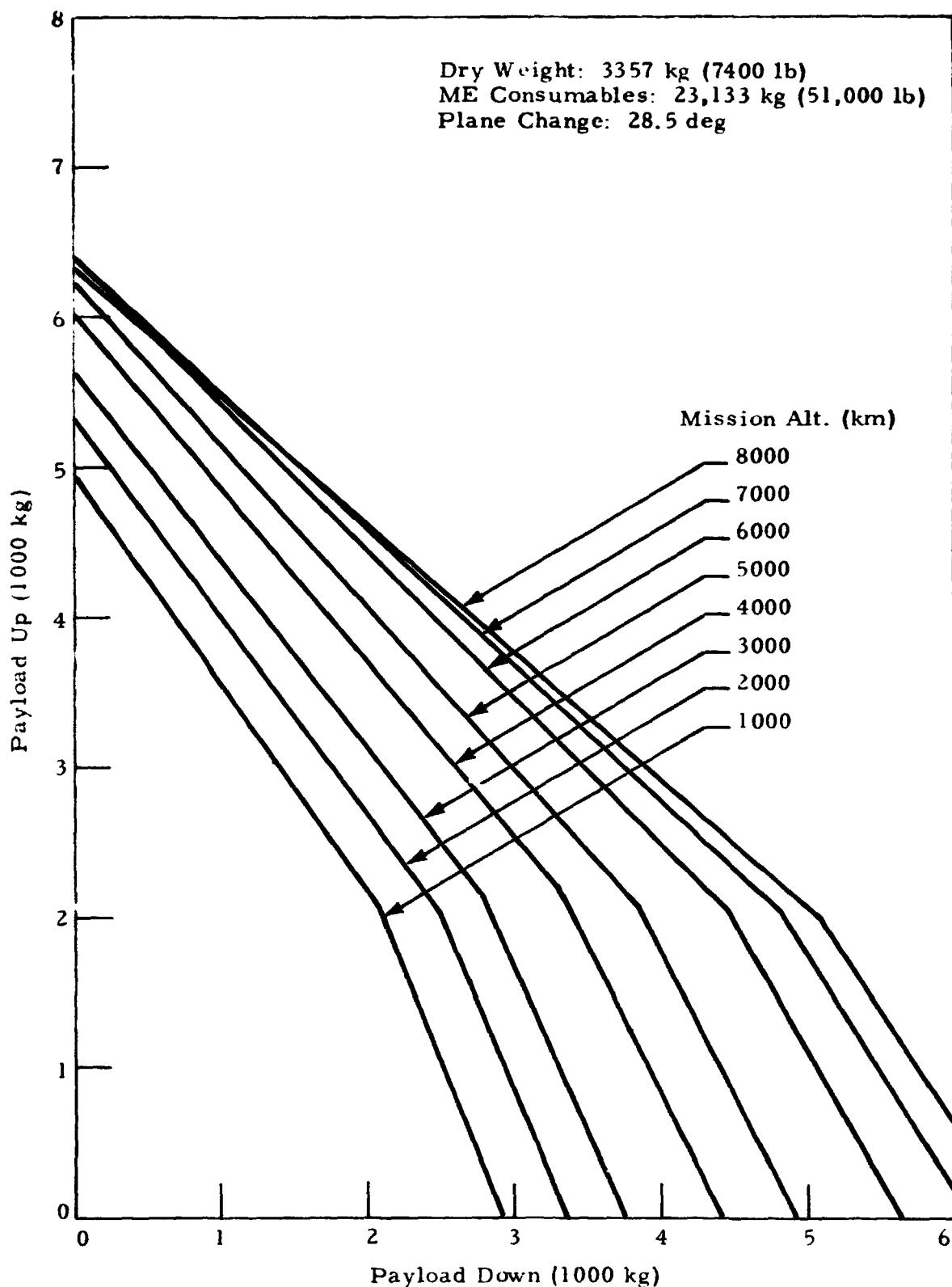
Fig. 7 - Aeromaneuvering Kit Cryogenic Tug Payloads to Low
Earth Orbit, I_{sp} · 470 sec

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The performance of the kitted AMOOS in the all-propulsive mode is shown in Fig. 8 for I_{sp} = 463 sec. The dry weight of the AMOOS in the all propulsive configuration is 2359 kg (5200 lb) with a corresponding propellant capacity of 22,000 kg (48,500 lb). The payloads reflect the trends in the total Δv requirement described above. The peak performance is to a mission altitude of approximately 4500 km (2430 n.mi.). The payloads fall with increasing and decreasing mission altitude over the range studied from the peak performance.

The payloads for the 2336 kg (5150 lb) dry weight Cryogenic Tug to the mission orbits of Fig. 8 are given in Fig. 9. Recall that the propellant capacity is 23,133 kg (51,000 lb) for this vehicle. The increase in the retrieved payload reflects this increase in propellant capacity. The delivery-only payload reflects the small decrease in dry weight as does the round trip payload.

The payloads capabilities for the all-propulsive, maximum modification AMOOS and the Cryogenic Tug to low earth orbits for an I_{sp} = 470 sec are given in Figs. 10 and 11.

The minimum modification AMOOS has a heavier dry weight than the 2359 kg (5200 lb) of the maximum modified AMOOS discussed above. Since neither modification involves a change in the aeromaneuvering vehicles' dry weight, the kit concept does not compromise the AMOOS performance to high or low mission altitudes. Since there is no compromise to the AMOOS performance by either kit concept, only the minimum weight all-propulsive configuration need be considered.

The changeover altitudes can now be determined on comparing Figs. 4 and 8 for the AMOOS derivatives and Figs. 5 and 9 for the Cryogenic Tug derivatives for a I_{sp} = 463 sec.

A comparison of the AMOOS derivatives performances to the 4000 km (2160 n.mi.) mission altitude is shown in Fig. 12. The all-propulsive vehicle

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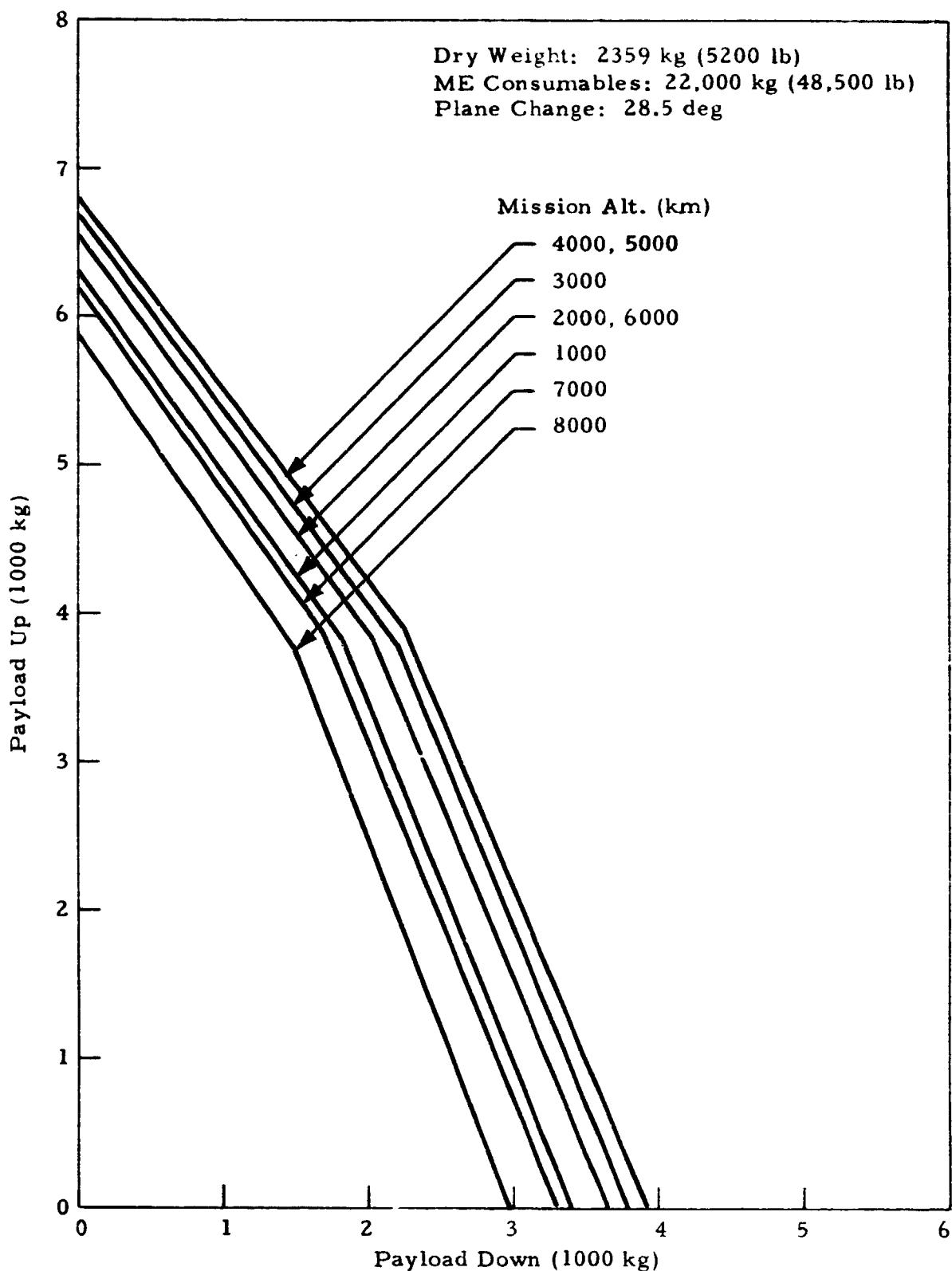
Fig. 8 - All Propulsive Kit AMOOS Payloads to Low Earth Orbit,
I_{sp} = 463 sec

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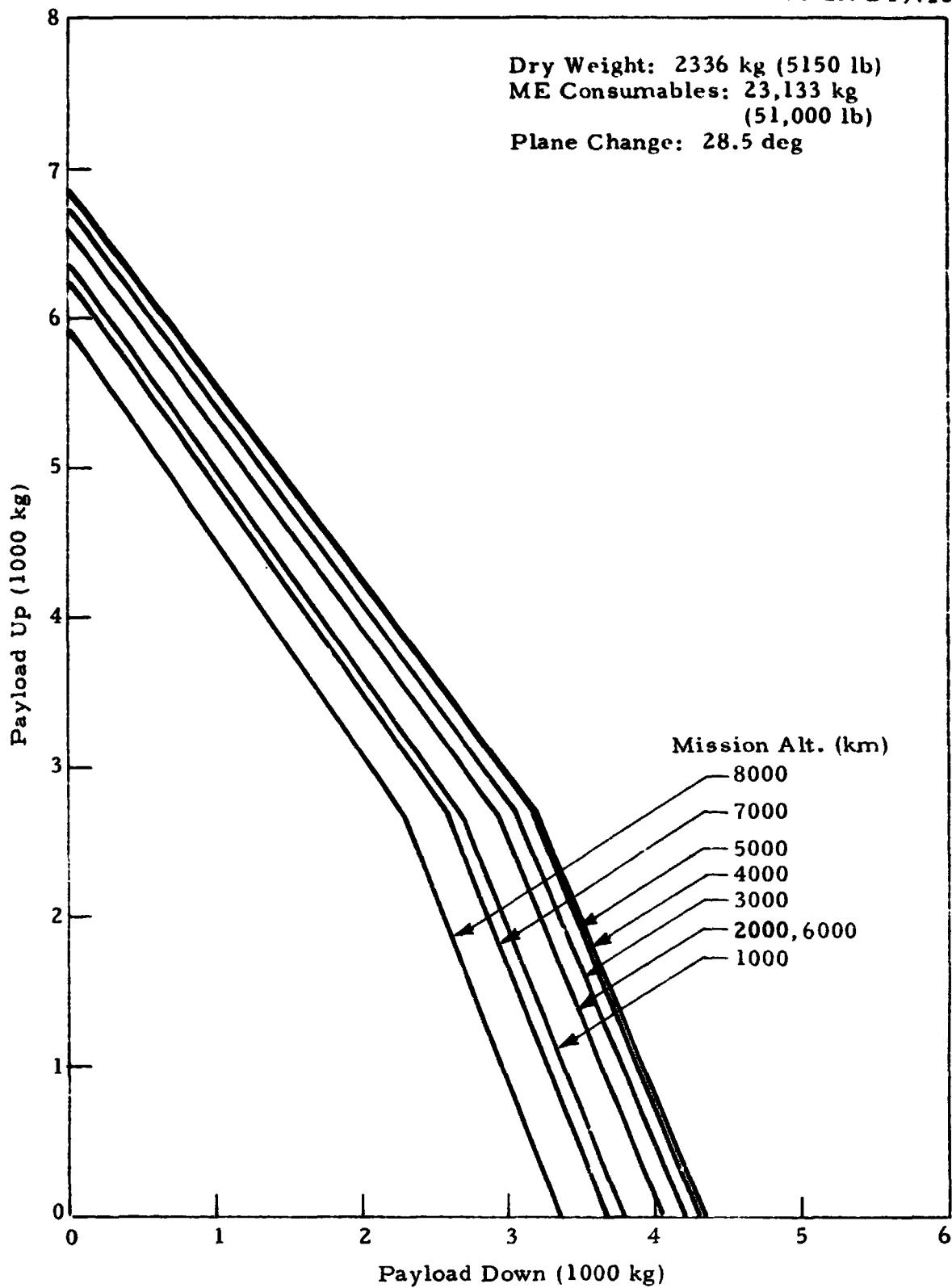
Fig. 9 - Cryogenic Tug Payloads to Low Earth Orbit, I_{sp} = 463 sec

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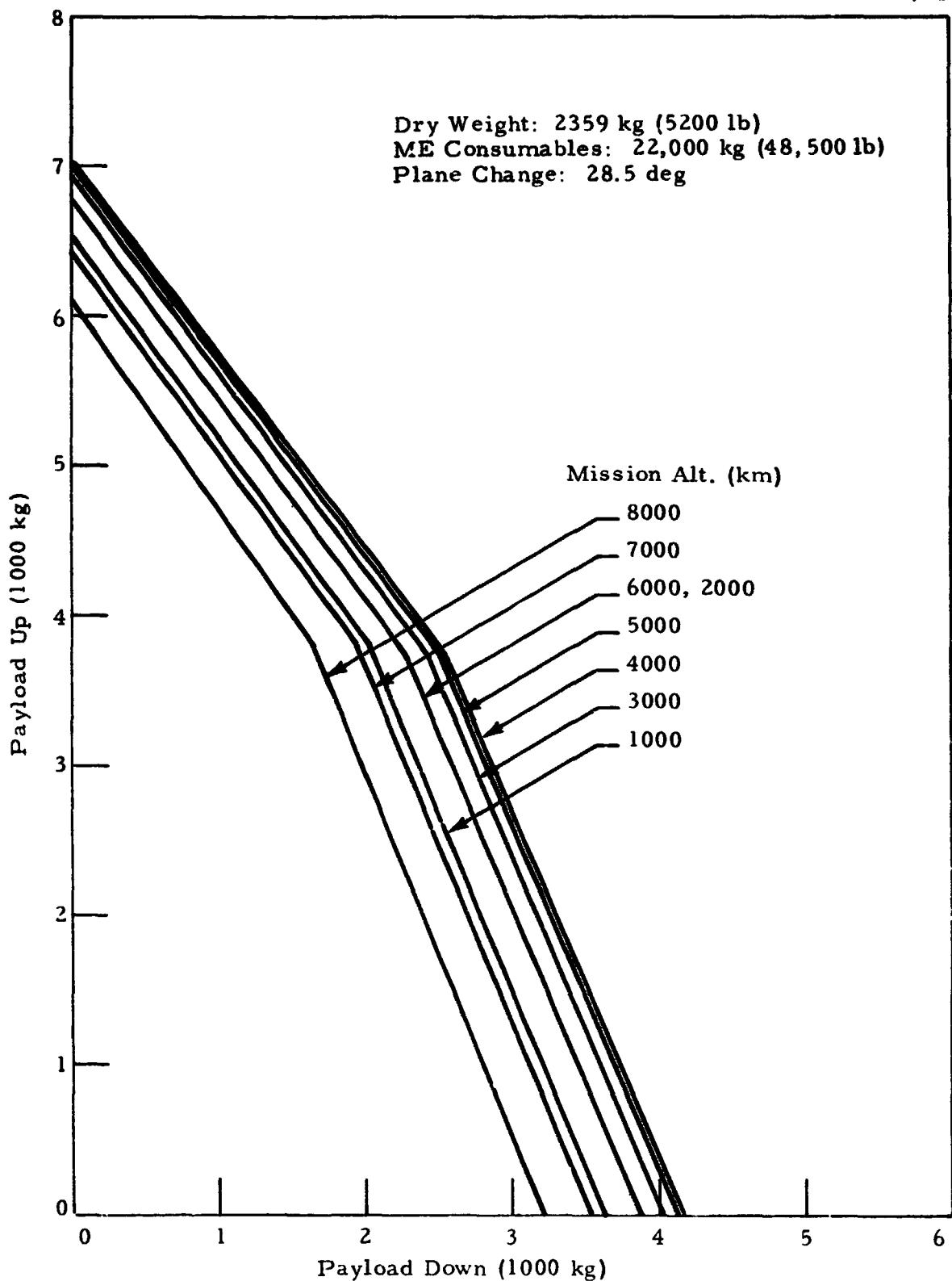
Fig. 10 - All-Propulsive Kit AMOOS Payloads to Low Earth Orbit,
I_{sp} = 470 sec

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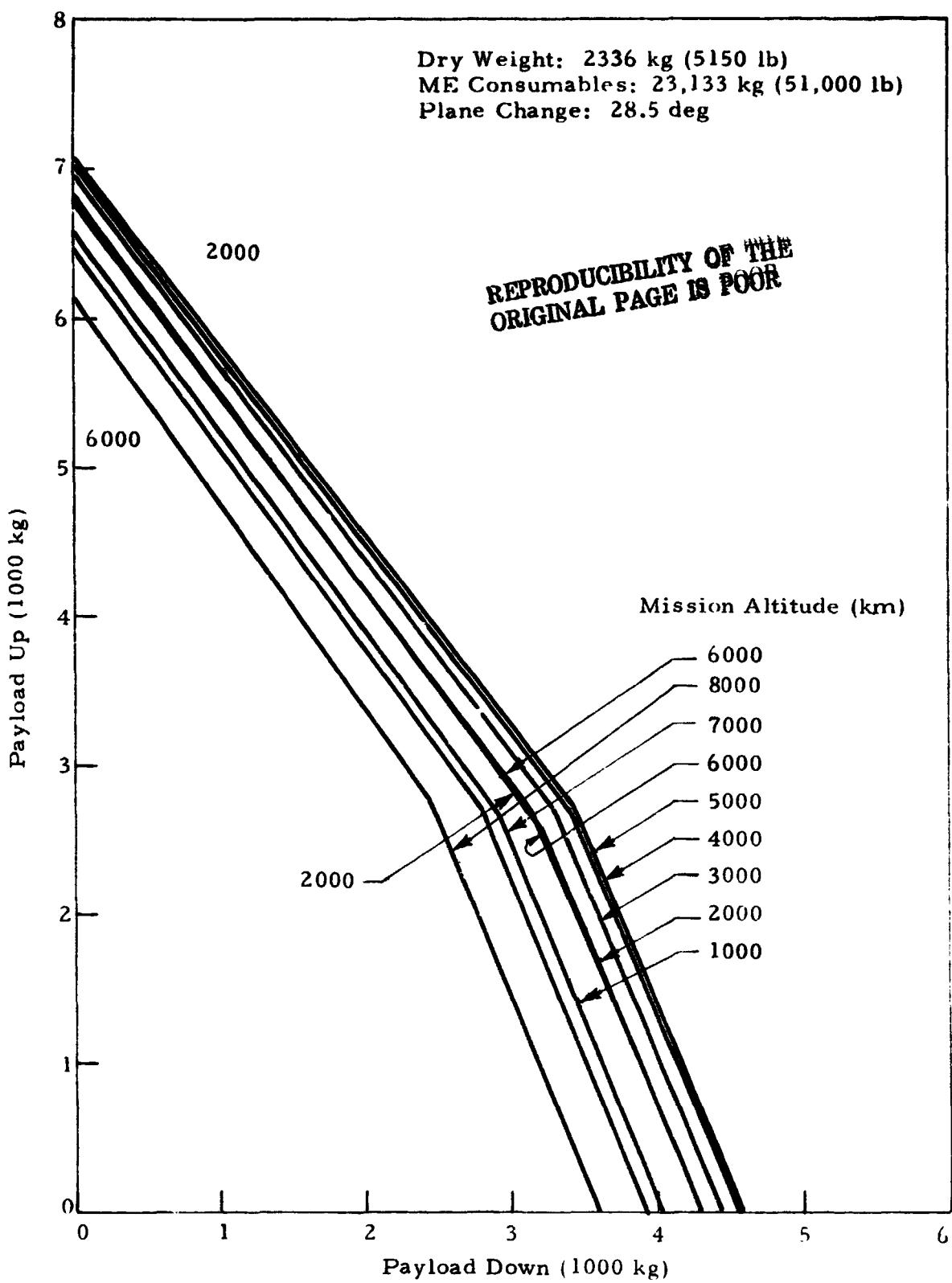
Fig. 11 - Cryogenic Tug Payloads to Low Earth Orbit, I_{sp} = 470 sec

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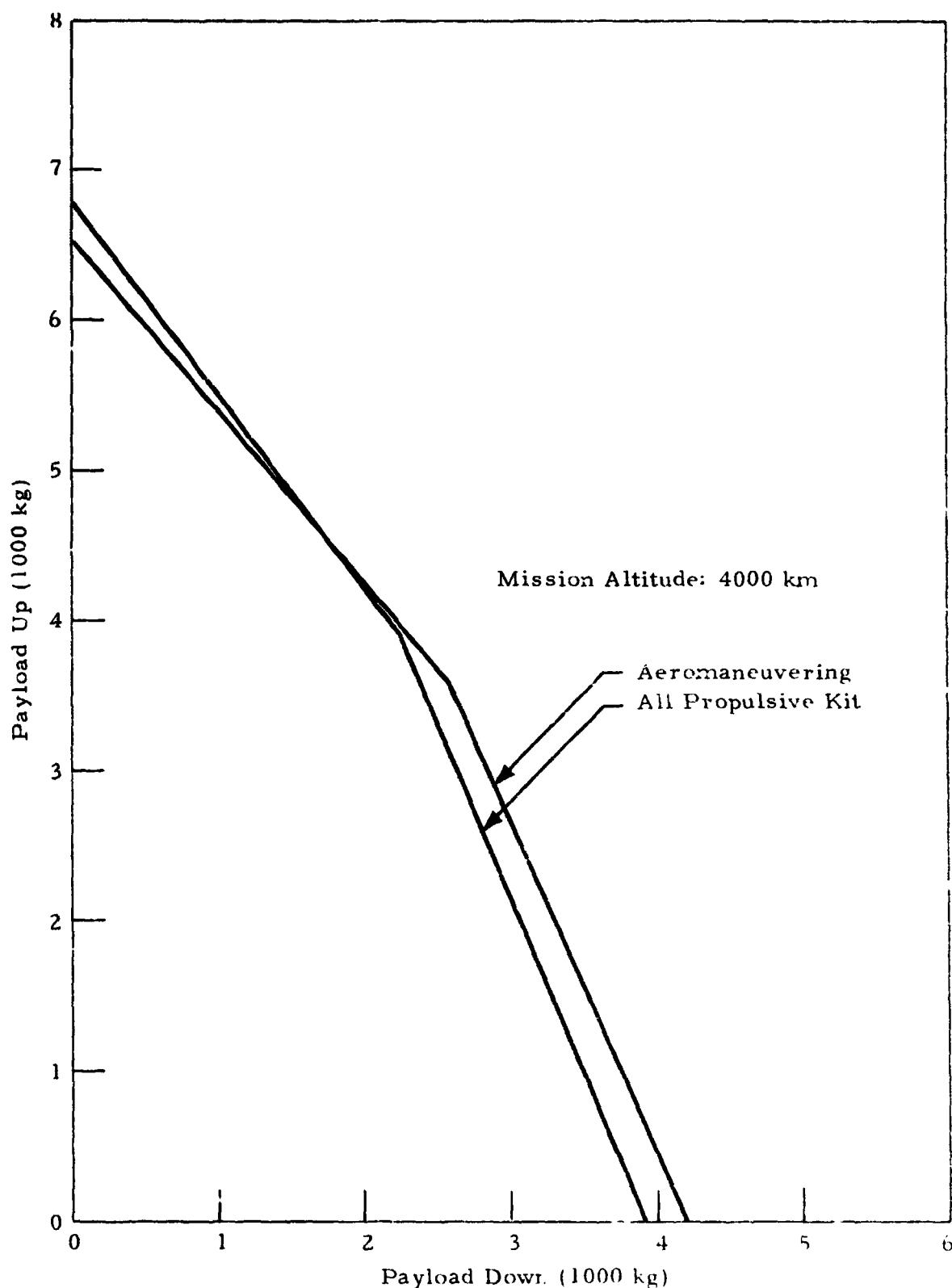
Fig. 12 - Comparison of the Performances of AMOOS and All Propulsive Kit AMOOS, I_{sp} = 463 sec

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outperforms the aeromaneuvering vehicle for all combinations of payload delivered and payload retrieved where the delivered payload is greater than 4590 kg (10,000 lb) approximately. The performance of the all-propulsive vehicle to the 3000 km (1620 n.mi.) mission altitude is only slightly inferior to that to 4000 km (2160 n.mi.). However, the aeromaneuvering vehicles performance at 3000 km (1620 n.mi.) mission altitude is significantly inferior to that at 4000 km (2160 n.mi.). Hence, a larger payload performance gap. The gap continues to widen for lower mission altitudes, cf. Figs. 4 and 8.

In Fig. 12, the payload curves intersect at approximately the 4500 kg (10,000 lb) delivered 2000 kg (4400 lb) retrieved point. For the delivery of payloads greater than 4500 kg (10,000 lb) (implies the retrieved payload is less than 2000 kg (4400 lb)), the all-propulsive configuration should be used. If, on the other hand more than 2000 kg (4400 lb) is to be retrieved with an appropriate delivery of less than 4500 kg (10,000 lb), then the aeromaneuvering configuration should be used.

As the mission altitude is increased then the crossover point occurs at a higher delivered payload. This is demonstrated in Fig. 13 where the performances to the 5000 km (2700 n.mi.) mission orbit are compared. The crossover point at this altitude is approximately 6300 kg (13,900 lb) delivered, combined with 700 kg (1550 lb) retrieved.

As the mission altitude is increased further, as in Fig. 14 for the 7000 km (3780 n.mi.) mission altitude, the aeromaneuvering vehicle outstrips all others. Also plotted on Fig. 14 the 2336 kg (5150 lb) all-propulsive Cryogenic Tug. This vehicle outperforms the 2359 kg (5200 lb) AMOOS derived all-propulsive vehicle but lowers the crossover point negligibly. The loss in performance in having a kit AMOOS perform the all-propulsive low altitude missions appears minimal. The performance of the increased propellant tankage kit AMOOS is also shown in Fig. 14. As expected, its performance parallels, but is slightly below, that of the Cryogenic Tug. This is due to its higher dry weight of 2427 kg (5350 lb) against 2336 kg (5150 lb) for the Cryogenic Tug.

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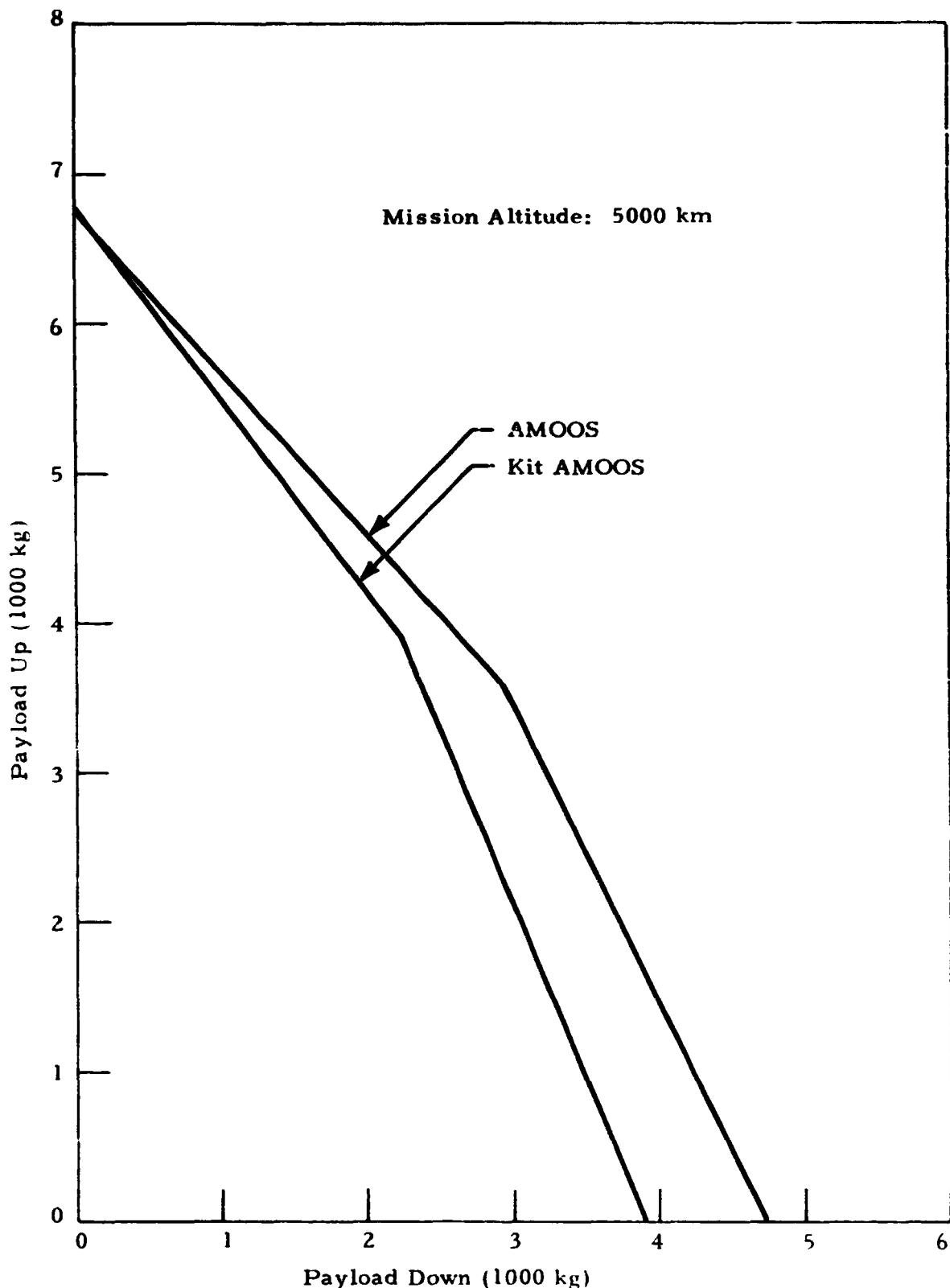
Fig. 13 - Comparison of the Performances of AMOOS and the All Propulsive Kit AMOOS, I_{sp} = 463 sec
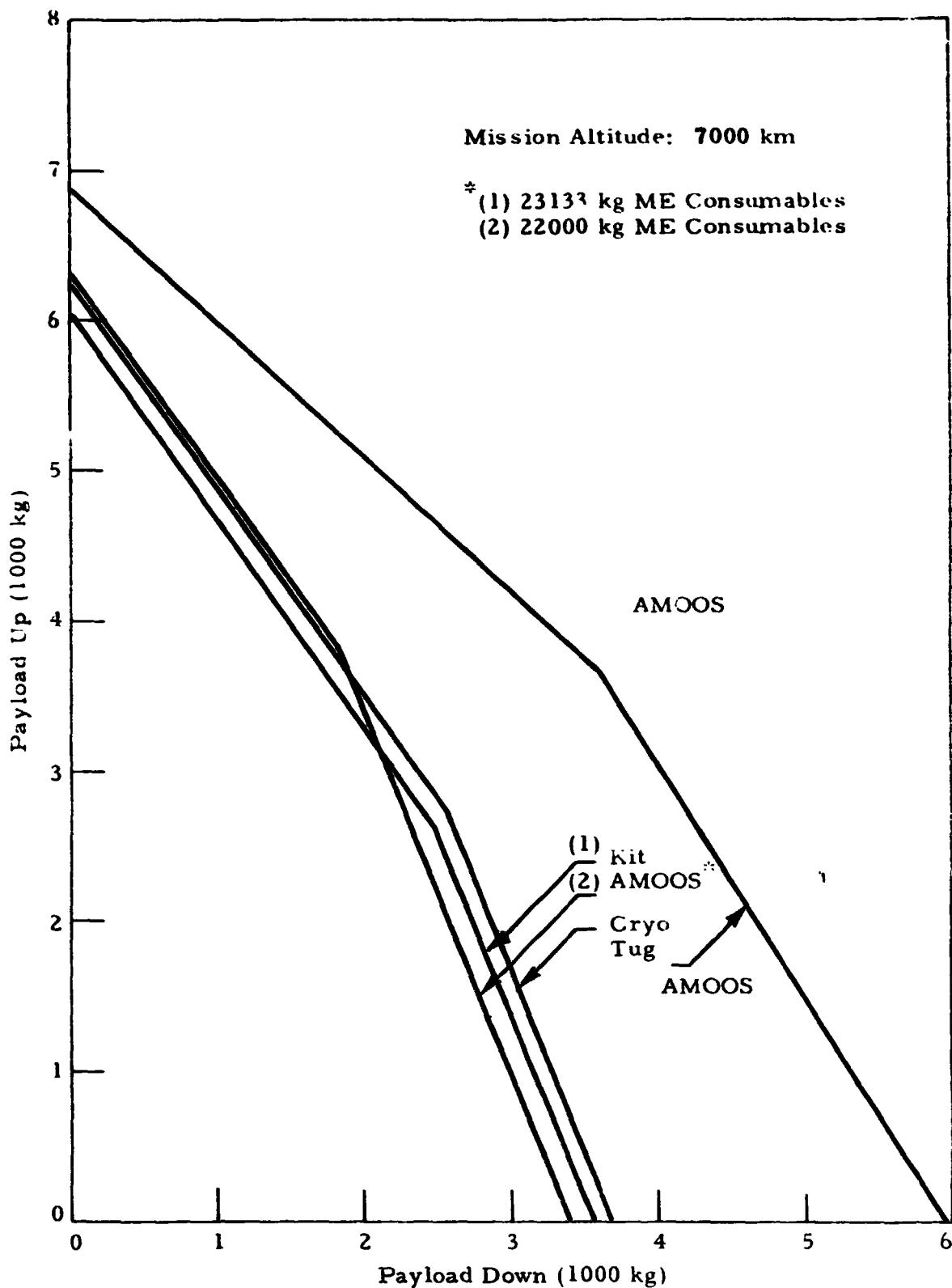
Fig. 14 - Payload Comparisons of AMOOS and All Propulsive Kit AMOOS,
I_{sp} = 463 sec

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AMOOS and its all propulsive derivatives will yield a better perfor-
mance combination than the Cryogenic Tug and its aeromaneuvering deriva-
tive. This is due to the relative dry weights of the several vehicles. The
23 kg (50 lb) excess of the all-propulsive kit AMOOS over ᵗhe Cryogenic Tug
degrades the all-propulsive performance less than does the ₃20 kg (700 lb)
excess of the aeromaneuvering Cryogenic Tug over AMOOS.

The selection from the above analysis is the AMOOS-derived vehicles,
in particular the maximum modification AMOOS vehicle with the 22,000 kg
(48,500 lb) propellant tank capacity. Two alternatives must now be considered,
the first in which the configuration is changed and the second in which the choice
of configuration is changed for a particular mission. The first alternative is
to increase the AMOOS propellant capacity to 23,133 kg (51,000 lb) total. The
second is to perform missions in one configuration for which the other is pre-
ferred, e.g., an all propulsive mission to geosynchronous altitude. Since the
low propellant capacity of the AMOOS derivatives would cause a slightly lower
round trip payload than for the Cryogenic Tug in the all-propulsive mode. it
may be desirable to increase the AMOOS propellant capacity to 23,133 kg
(51,000 lb). However, the need to operate in other than the preferred mode
must first be established.

The increase in performance at low altitudes has been shown to be
small and, in general, confined to missions in which the retrieved payload
is large compared to the maximum retrievable payload for that particular
mission orbit. In Fig. 15 the payloads are given for the all propulsive
AMOOS derivative to missions up to 8000 km (4320 n.mi.) with a 28.5 deg
plane change. The performances for an I_{sp} = 463 sec are given in Fig. 15
and for I_{sp} = 470 sec in Fig. 16. On comparing Fig. 15 with Fig. 8, it can be
seen that the payload retrieved range is extended at the expense of the pay-
load delivered. This has been discussed earlier in connection with Fig. 14.
Currently, the delivered payload capability is considered more important
than the retrieved payload capability. The importance of the round trip

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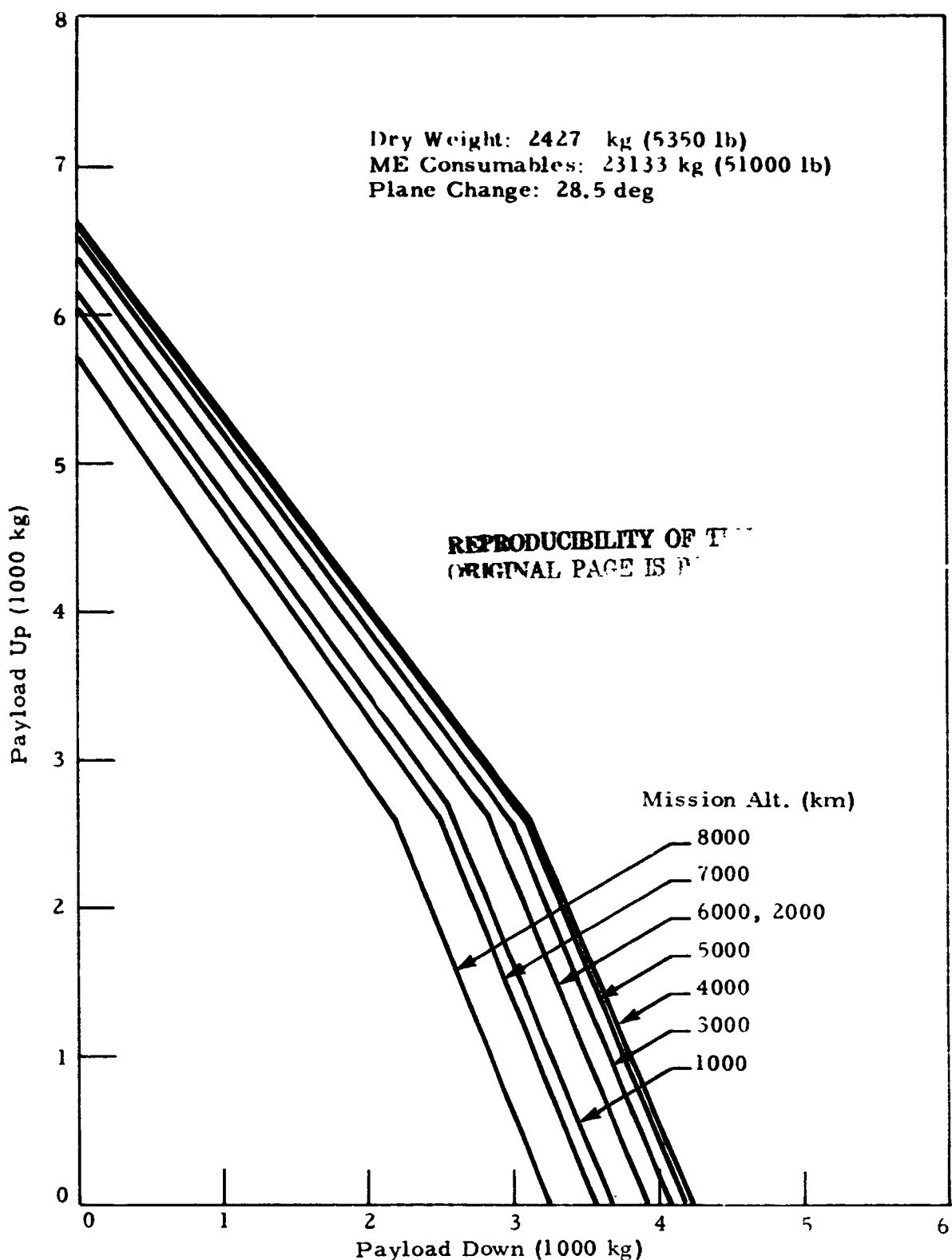
Fig. 15 - All-Propulsive Kit AMOOS Payloads to Low Earth Orbit, I_{sp} = 463 sec
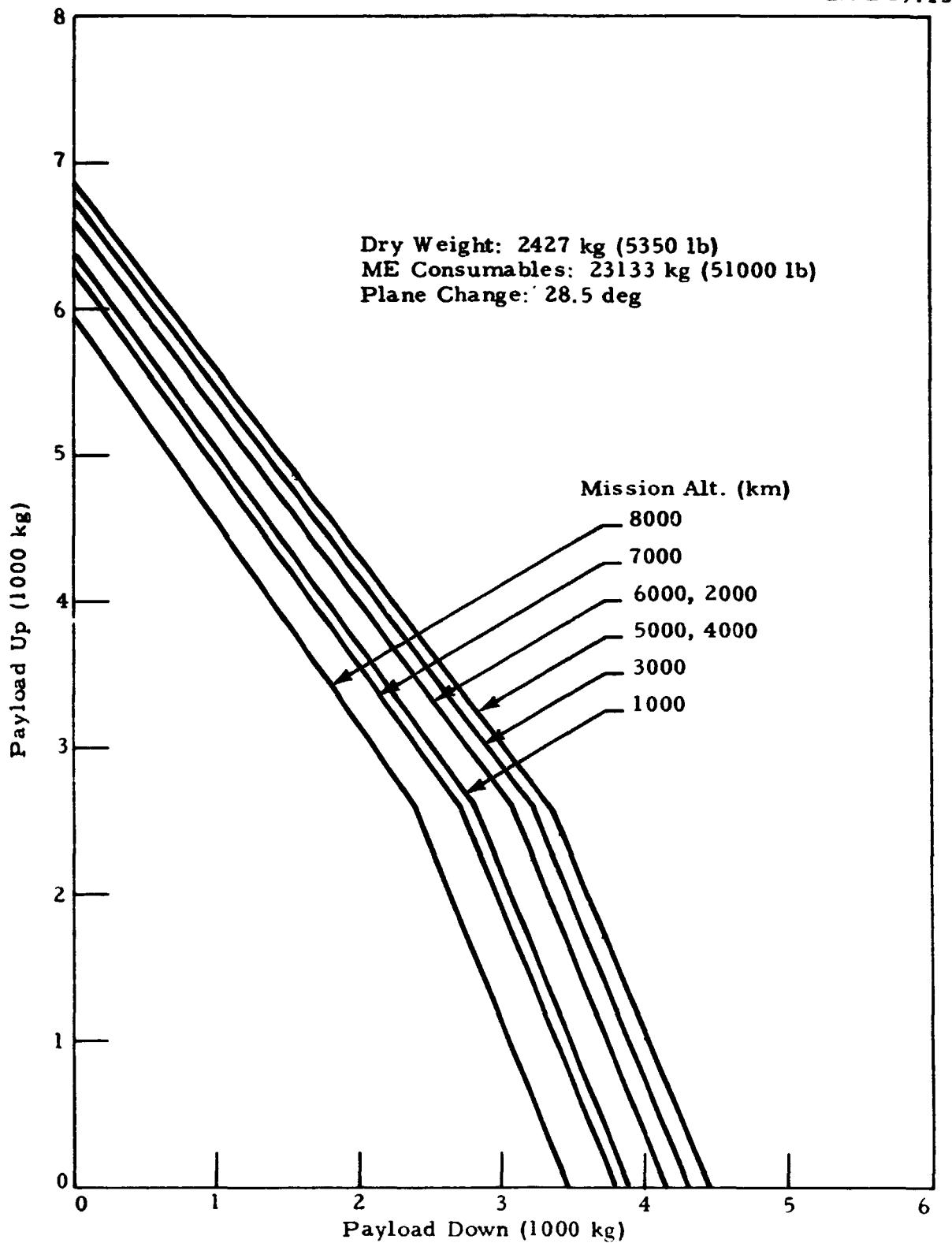
Fig. 16 - All-Propulsive Kit AMOOS Payloads to Low Earth Orbit, $I_{sp} = 470$ sec

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capability relative to the delivery capability is not clear at this time. The dry weight increase to the kit AMOOS in the all-propulsive mode was estimated to be 68 kg (150 lb). When the TPS is added for aeromaneuvering, the penalty is estimated to double (136 kg (300 lb)).

The above performance analysis was for a mission with a 28.5 deg plane change. For low altitude missions a plane change may not be called for or may not be required since a WTR launch may be used. The performances of AMOOS and the all-propulsive kit were computed for typical Shuttle delivery weights for a WTR launch. These performances are plotted in Fig. 17. Also included is an ETR launch of a no-plane change mission. At the mission altitude selected, namely 3700 km (2000 n.mi.), the difference between AMOOS performance and the all propulsive vehicle is negligible for delivery only through round trip missions with AMOOS slightly outperforming the all propulsive vehicle. If missions altitude is increased, the AMOOS will increasingly outperform the all-propulsive vehicle. Also, if the return payload requirement is increased beyond round trip requirement then AMOOS will significantly outperform the all-propulsive vehicle. The altitude of 3700 km (2000 n.mi.) represents the approximate altitude below which the all-propulsive mode would be used instead of aeromaneuvering for delivery only.

Figure 17 may also be used with Figs. 4 through 11 to determine when a WTR launch should be used instead of an ETR from payload considerations for the missions considered.

2.3 CONCLUSIONS

The AMOOS configuration appears readily adaptable to the kit concept. The dry weight of the maximum modification vehicle (Fig. 2) is only 23 kg (50 lb) above the Cryogenic Tug. This vehicle, however, has a propellant capacity of 22,000 kg (48,500 lb) as compared to the 23,133 kg (51,000 lb) capacity of the Cryogenic Tug. This modification has been investigated but is not recommended since the performance increase is small and

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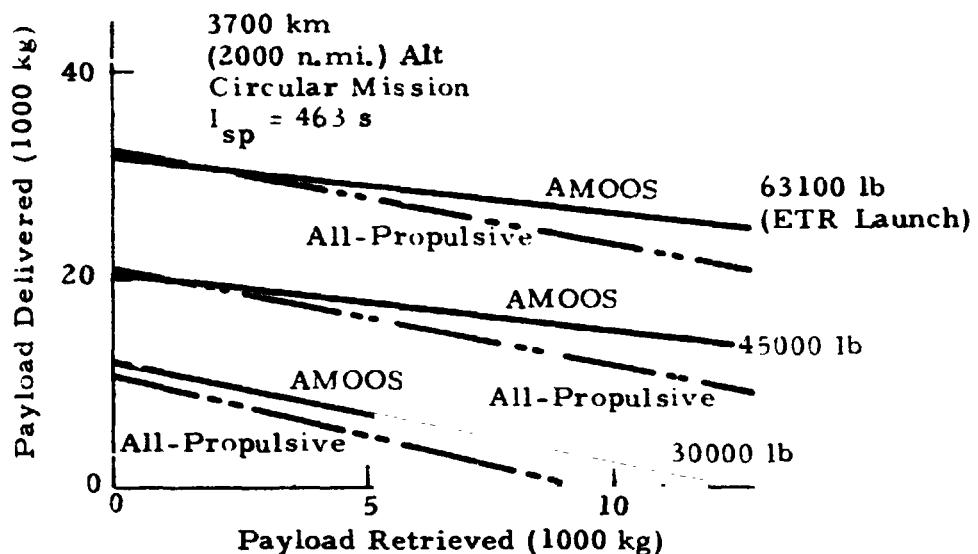
Fig. 17 - Western Test Range Shuttle Launch AMOOS and All-Propulsive
Vehicle Performance

directly affects the retrieved payloads only. The performance cost to AMOOS
in the aeromaneuvering configuration is practically zero since the design of
AMOOS allows the structure and TPS to be removed or replaced without penalty.

The effect of the increased propellant of the cryogenic tug may be seen
in the minimum propulsive Δv capability of each vehicle. The propulsion
module is expended in each case. The maximum delivered weight of 28,622 kg
(63,100 lb) to low earth orbit is assumed for each vehicle. The AMOOS has the
capability of boosting a payload of 4264 kg (9400 lb) by a Δv of 6750 m/s (22,150
ft/sec) whereas the Cryogenic Tug can boost a 3175 kg (7000 lb) payload by a
Δv of 7600 m/s (25,000 ft/sec). For equal Δvs the Cryogenic Tug's payload is
only marginally different from the kit AMOOS, the difference being the differ-
ence in dry weight, namely 23 kg (50 lb). This holds only up to the point that
the main engine consumables is equal to 22,000 kg (48,500 lb). If the kit AMOOS
propellant capacity is increased to 23,133 kg (51,000 lb) then the payload differ-
ence is still the difference in dry weights for a given Δv. In this case it is
approximately 90 kg (200 lb).

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Section 3
THERMAL PROTECTION SYSTEM OPTIONS AND TRADES

At the time of the original design of AMOOS (Refs. 1 and 2), the requirement to perform a round trip geostationary mission outweighed all others. The thermal environment resulting from this mission allowed only three distinct solutions in the design of the thermal protection system (TPS). These were: (1) an ablative TPS and hence a one-pass mission; (2) a carbon-carbon TPS; and (3) a multi-pass maneuver to reduce the heating rates and hence allow the use of superalloys or insulative materials. Various combinations of the above were also considered, e.g., ablator on the hot spots and LI 900 where the surface temperature was less than 1500 K (2300 F). A brief analysis showed that alternatives (2) and (3) yielded a very heavy TPS or, in the case of (3), violated the on-orbit lifetime of six days permissible for an upper stage if it were to be transported to and from low earth orbit in one Shuttle mission. The patching of an ablator with LI 900 poses potential problems at the junction of the ablator and the LI 900. For these reasons, the ablative TPS with a one-pass maneuver was chosen. The availability of this alternative was made possible only by the use of lift forces to control the trajectory during atmospheric flight. It is practically impossible to perform a one-pass skip maneuver accurately or precisely with a ballistic vehicle.

The advent of the Growth Shuttle concept and the Shuttle derived HLLV has lead to a reassessment of the design and operation guidelines for AMOOS. Furthermore, the demands on the Space Transportation System has changed considerably since the original AMOOS contract. This has lead to the requirement for vehicles with a larger payload capability sufficient to rotate part or all of the crew of a pilot Space Industrialization plant. The rise of the importance of such a space station has lead to the potential need

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of an emergency vehicle for crew return namely the AMRS. These develop-
ments in turn stress the need to look again at the AMOOS TPS requirements
and the methods and materials available.

Essentially, no new materials were found available although, of course,
it is known that development is continuing. Two methods were considered
that had not been considered previously. The first was the use of heat pipes
and the second was a heat sink application using liquid hydrogen. The former
was considered worthy of perusal, probably as a technology project. Current
application was not possible due to the large difference in the test heating
rates (Ref. 6), the AMOOS and AMRS heating rates being about 3.5 to 5 times
those reported in the tests. The heat sink concept was dropped due to the
significant mass of LH_2 required relative to the LH_2 used as fuel. Other
substances were not considered since LH_2 and, after boiloff, GH_2 has ex-
cellent latent heat of vaporization, specific heat and allowable temperature
rise qualities.

The only practical method of changing the heating rate on the previous
AMOOS designs is to change the number of passes required to perform the
maneuver and so allow a corresponding change in altitude and atmospheric
density. This is due to the performance and transportation requirements
allowing only minor variations in shape. The previous AMOOS configuration
(Refs. 1, 2 and 3) is probably near optimum for the shaping allowable so that
any reduction in heating rate by shaping would be minimal. This discussion
holds, of course, only if the mission altitude is invariant. The maximum or
design heating rate may be reduced by decreasing the maximum mission alti-
tude. Reducing the mission altitude reduces the entry velocity, the excess Δv
and energy that must be dissipated. Other means of reducing the heating rate
and the heat load have been considered and found impractical (Refs. 1 and 3).
These include a propulsive Δv to slow the vehicle immediately prior to atmos-
pheric entry and choosing the nominal trajectory to have a downward component
of the lift force. The former method is equivalent to reducing the mission alti-
tude whereas the latter is equivalent to increasing the number of passes per

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maneuver. In this latter method, the heating rate is reduced but the total
heat load is increased, as it is in the multi-pass maneuvers.

The reason for this increase in heat load with a corresponding decrease
in heating rate is worthy of explanation. The deceleration at a point in time
is proportional to the atmospheric density whereas the heating rate is, in the
transitional and continuum regimes, proportional to the atmospheric density
to a power less than one, and probably close to 0.5. The velocity loss is
dependent on the time integral of the deceleration. The duration of the atmo-
spheric flight therefore increases more rapidly than the heating rate decreases
for the velocity loss to be invariant. Hence the increase in heat load with an
associated decrease in heating rate. For the same reasons the heat load in-
creases while the heating rate decreases for the multi-pass maneuvers.

These studies were confined, essentially to areas not studied previously,
namely, the effects of multi-pass maneuvers on heating rates, definition of
allowable mission altitudes for given materials and maneuver strategy and
a brief study of different methods of dissipating the heat load.

3.1 MISSION ALTITUDE LIMITS FOR RERADIATIVE AND INSULATIVE MATERIALS

The purpose of this task was to establish the mission altitude limits
for recyclable TPS materials using a particular maneuver strategy. To this
end nominal trajectories were generated for mission altitudes from 1500 km
(800 n.mi.) through geostationary altitude (35,833 km (19,348 n.mi.)). Up to
and including 20 pass maneuvers were considered within this mission altitude
range. Off nominal trajectories were not generated since previous studies
(Ref. 3) have shown that the AMOOS and AMRS guidance schemes were cap-
able of limiting the heating rate and heat load excursions to approximately
1% of the nominal value. Such an excursion will have neglible effect on the
TPS materials range capability when necessary safety factors are considered.

The heating rates and hence the temperature for an emissivity of 0.7 were computed for the AMOOS vehicle at a 45 deg angle of attack. Heating rates were computed using transitional flow equations since the AMOOS vehicle is within the transitional regime except at altitudes close to 120 km (400,000 ft). At these high altitudes, the flight is in the free molecular regime and the heating rate is negligible. The heating rates were computed for the stagnation point on the nose of the vehicle and on the stagnation line on the elliptical cone frustum sections of the vehicle. The heating rates for other locations were obtained using factors derived from heating rate tests reported within the literature. A wealth of information is available on windward side heating rates but data is sparce on leeward side heating rates at these high angles of attack. The tudy of lee side heating rates has been recommended as a technology study task in Section 8.

The stagnation point temperature on the nose of the vehicle is plotted in Fig. 18. As expected, the trend is reducing temperatures with decreasing mission altitude and increasing number of passes per maneuver. The decrease in temperature is very slow with decrease in mission altitude until approximately 10,000 km (5400 n.mi). At this altitude the rate of decrease in temperature is noticeably changing. All the temperature curves fair into zero temperature at ~ mission altitude of 720 km (388 n.mi.), the target apogee for the phasing orbit. The cause of this slow variation in temperature is due to many factors. Thes·. factors are in turn a consequence of the basic laws of physics and thermodynamics and not, in general, a function of the AMOOS vehicle nor its modus operandi.

The overriding effect so far as surface temperature is concerned is that the heat loss is largely by radiation which obeys a fourth power law in temperature difference. The temperature varies, then, as the fourth root of the heating rate. This basically accounts for the slow variation of temperature with mission altitude and other strategies that change either the maximum atmospheric density, the drag force or both.
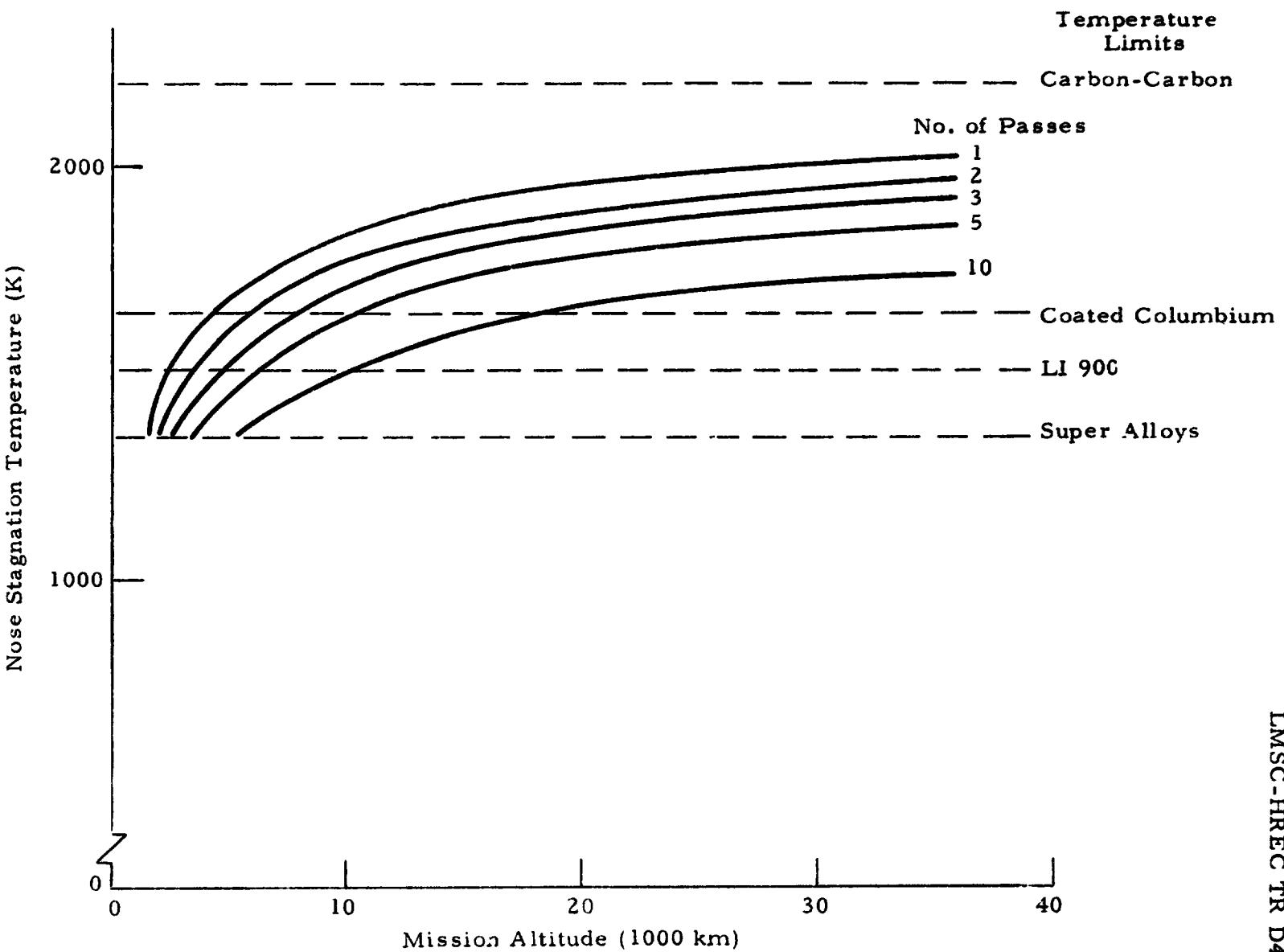
Fig. 18 - Maximum Temperature vs Mission Altitude

As noted previously, the tendency is for the heating rate to vary more slowly than the drag force with changes in number of passes per mission and atmospheric density. The temperature decreases more slowly than the mission altitude due to the slower decrease of the heating rate. The time of flight is, of course, dependent upon the deceleration and hence the drag force. Therefore, the net effect of an increase in the number of passes per maneuver or decrease in peak density is to increase the heat load.

The temperature limits of several recyclable TPS materials have been superimposed on Fig. 18. The first considered is carbon-carbon with a temperature limit of approximately 2200 K (3500 F). This material may be used for all mission altitudes up to geostationary and beyond. However, its drawbacks are weight and cost. Although these drawbacks have not been fully investigated, they are considered sufficient to eliminate it at this time. The masses required per unit area of various materials are given as a function of temperature in Fig. 19. Discussion of the method of generating this figure will appear later in this section.

The next material considered was coated columbium which may be used to a emperature of approximately 1650 K (2500 F). For general application on the nose of AMOOS, its use is limited to mission altitudes below 18,000 km (9700 n.mi.) approximately for a ten-pass maneuver by the temperatures at or near the stagnation point. This maximum mission altitude for general use decreases, of course, with decreasing number of passes per maneuver. For a single-pass maneuver, it has decreased to approximately 4200 km (2300 n.mi.). Although aeromaneuvering is still advantageous at such low altitudes, all propulsive maneuvering costs a relatively small percentage of the payload. In certain conditions, particularly if a large plane change is involved, the all propulsive strategy has the advantage. The mass per square meter is reduced from that of carbon-carbon but still exceeds that required by a light weight ablator, in particular, SLA 561.
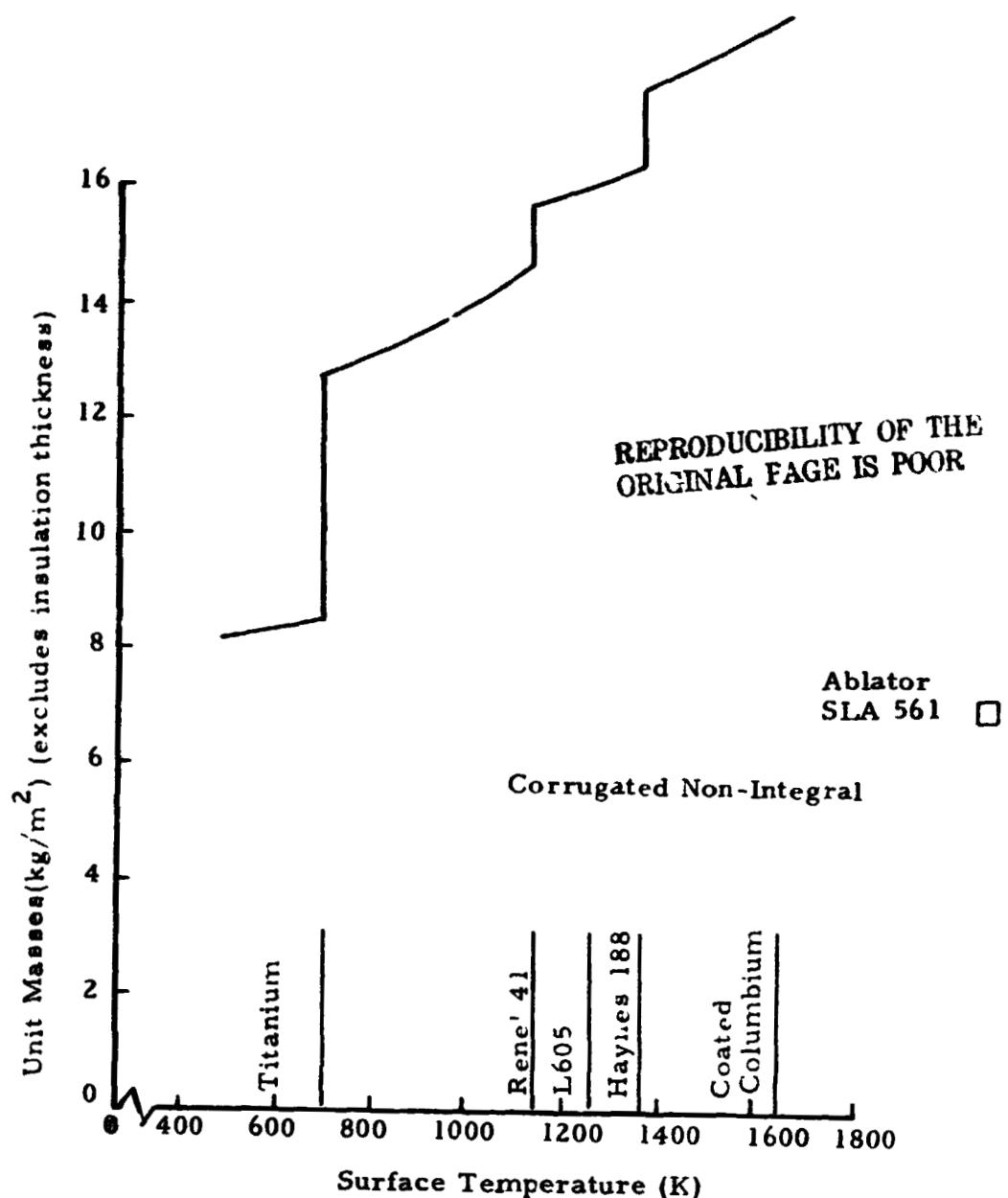
Fig. 19 - Mass per Unit Area for Various Radiative-Type TPS Materials

The two materials considered above are structural materials able to withstand very high temperatures. Equilibrium temperature is reached very quickly so that an insulator is required on the inside to protect the interior of the v hicle. The next material considered is LI 900 which is an insulator. It must be supported by an underlying structure, which it protects. Heat dissipation, as in the case of the structure materials, is by reradiation. This material can be used in areas where the temperature is as high as 1500 K (2300 F). The mission altitude for general use is reduced to approximately 11030 km (6000 n.mi.) for the ten-pass mission and to approximately 2300 km (1250 n.mi.) for the one-pass mission.

Finally, the temperature regime of the superalloys is reached. For practical purposes, its range of general applicability is below 5000 km (2700 n.mi.) mission altitude. The maximum allowable temperature is 1350 K (2000 F). Such materials are also structural and produce reradiative-type TPS and also require an insulator backing to protect the interior of the vehicle. In general, the superalloys are not competitive with LI 900 on a mass per unit area basis. Furthermore, they are not competitive on a temperature basis. They are, of course, less liable to accidental damage since the alloys are steel or nickel based. The unit mass for LI 900 is not only temperature dependent but also time dependent. As with all insulative materials, temperature equalization or stability is achieved, given sufficient time. Therefore the material thickness required is a function of the temperature-time history. This also applies to the reradiative, structural-type TPS since the use of such materials requires an insulative backing. However, in the case of the reradiative, structural type TPS, the mass of the insulation is small compared to the structural material itself for flight times of a few minutes in the high heating rate portions of atmospheric flight.

The task of comparing an ablative TPS to a reradiative, structural TPS on a unit mass basis is even more difficult since the ablator weight is a function of the heat load rather than the heating rate. In Fig. 20, unit masses of

40
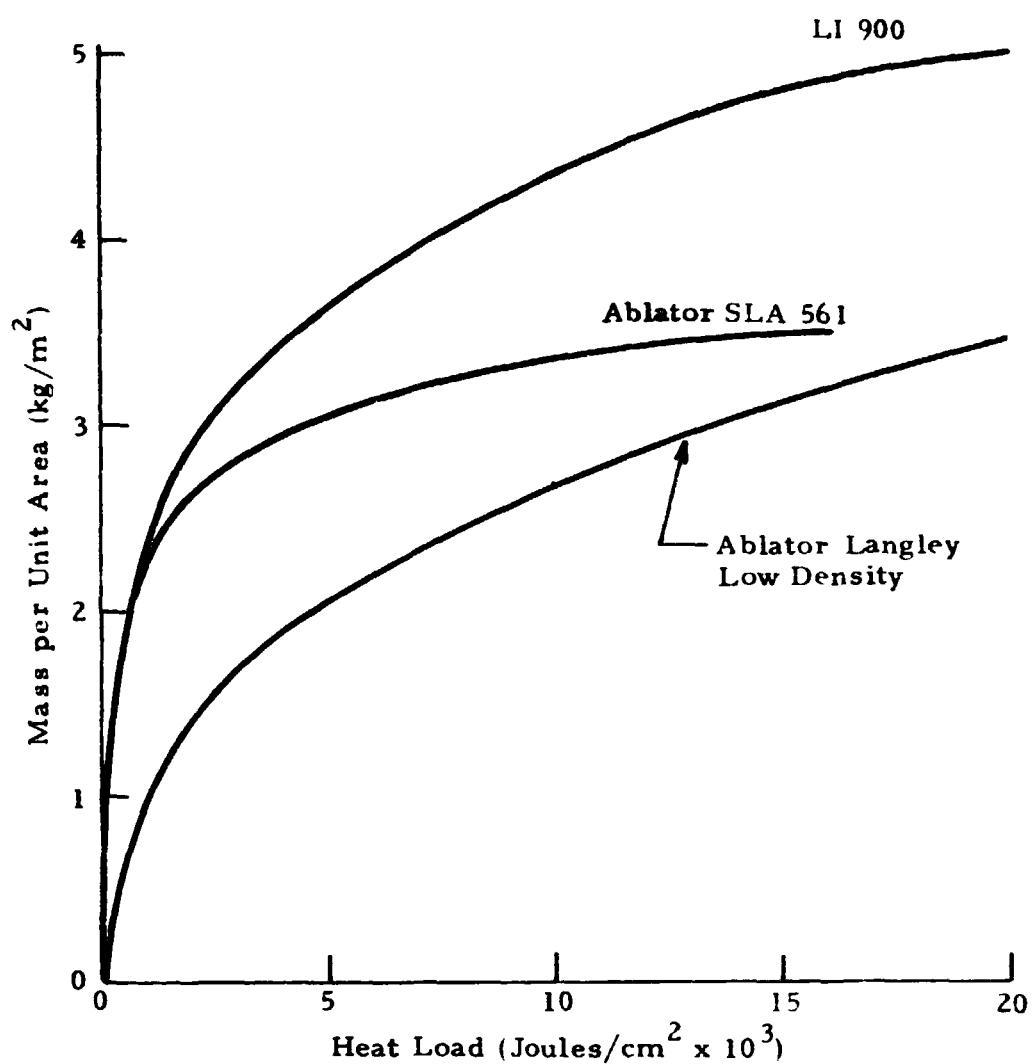
Fig. 20 - Unit Mass of Typical Insulative and Ablative TPS over a Range
of Heat Loads Representative of Various Locations on the
AMOOS Vehilce

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LI 900 are compared to those of the Langley low density ablator and the Martin Marietta ablator (SLA 561), each as a function of heat load. The ablators out perform the insulator over the heat load range considered, and, furthermore have a higher maximum heating rate capability. In each case a load bearing structure is required to support the TPS. Furthermore, the peak heating rates are assumed to be within the limitations of each material.

Provided the heat load, the peak heating rate, and hence temperature, and the duration of the flight within the sensible atmosphere is within limits, Figs. 19 and 20 may be compared. The primary structural mass must be added to the ablators and the LI 900. This lies in the range of 3 to 4 kg/m^2 (0.6 to 0.8 lb/ft^2). Adding in this unit mass range to Fig. 20 gives a range for LI 900 from 6 to 9 kg/m^2 (1.2 to 1.8 lb/ft^2) (Ref. 2) for comparison with Fig. 19. Over the peak temperature range of the super alloys, LI 900 will yield a lighter weight TPS for our applications. The ablators considered will yield an even lighter TPS. However, the Langley low density ablator was replaced with the SLA-561 because it is a more developed material, and is presently being used on various vehicles.

To this point, only a homogeneous TPS has been considered. A viable alternative is usually a combination of two or more materials, each matched to the local thermal environment on the vehicle. Previously an ablator on the hot spots and LI 900 from approximately the 70 deg radial from the stagnation line was considered (Ref. 2). Such a combination is expected to require considerable development at the junction of the ablator and LI 900, in particular since the ablator areas must be scraped and resprayed without damaging the LI 900.

In order to evaluate several TPS materials, plots of surface temperature versus mission altitude were generated for 1 through ten-pass missions at various locations on the body. Isolated points were also computed for a 20-pass mission. The plot of the body stagnation line temperatures versus

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mission altitude is given in Fig.21. The temperatures of Fig.21 are con-
siderably lower than those of Fig.18. Carbon-carbon, excepting ablators,
is still the only material of general applicability. However, in the ten-pass
case, coated columbium may be used for missions to geostationary altitude
and beyond. The range of missions of the remaining materials are also ex-
tended for use on the stagnation line when compared to use on the stagnation
point on the nose.

The temperature decreases steadily from the stagnation line around the
body toward the leeward side. Reliable data is available only on the wind-
ward side, therefore the temperatures were computed only to the 90 deg
radial (Fig.22). However, at this radial, the surface temperatures have
fallen to within the temperature range of all the TPS materials considered.
LI 900 probably yields the lightest weight TPS over its temperature range,
ablator excepted, of course. Since its temperature range is tc 1500 K (2300 F),
it may be used from the 90 deg radial outward for all missions up to geosta-
tionary regardless of maneuver strategy. However, not much is to be gained
by increasing the number of passes per maneuver up to five. Increasing the
number beyond five yields rapidly increasing benefits, e.g., for a ten pass
maneuver, LI 900 can be used from the 32 deg radial onward. Coated colum-
bium could be used over the remaining portions of the body, except the nose,
of course. Such combinations have been considered previously (Ref.2), in
particular from the cost viewpoint. Since these studies the modular AMOOS
concept has been developed, so that, although the findings, in general, are
applicable, the detailed findings are not. In order to put the reradiative and
insulative TPS systems in the correct perspective with respect to the ablative
TPS, several primary structural designs were generated for coated columbium
and Rene' 41. The primary structural weight for titanium was computed from
previous designs and that for Haynes 188 interpolated using its physical proper-
ties which lie, in general between those of Rene' 41 and coated columbium.
For structural design purposes, the vehicle is divided into several sections
such as: the nose, forward cylinder, center cylinder and aft cylinder. The
division is arbitrary and so was selected at axial points where a significant
change in the bending moment occurs. Over each of these sections, the

43
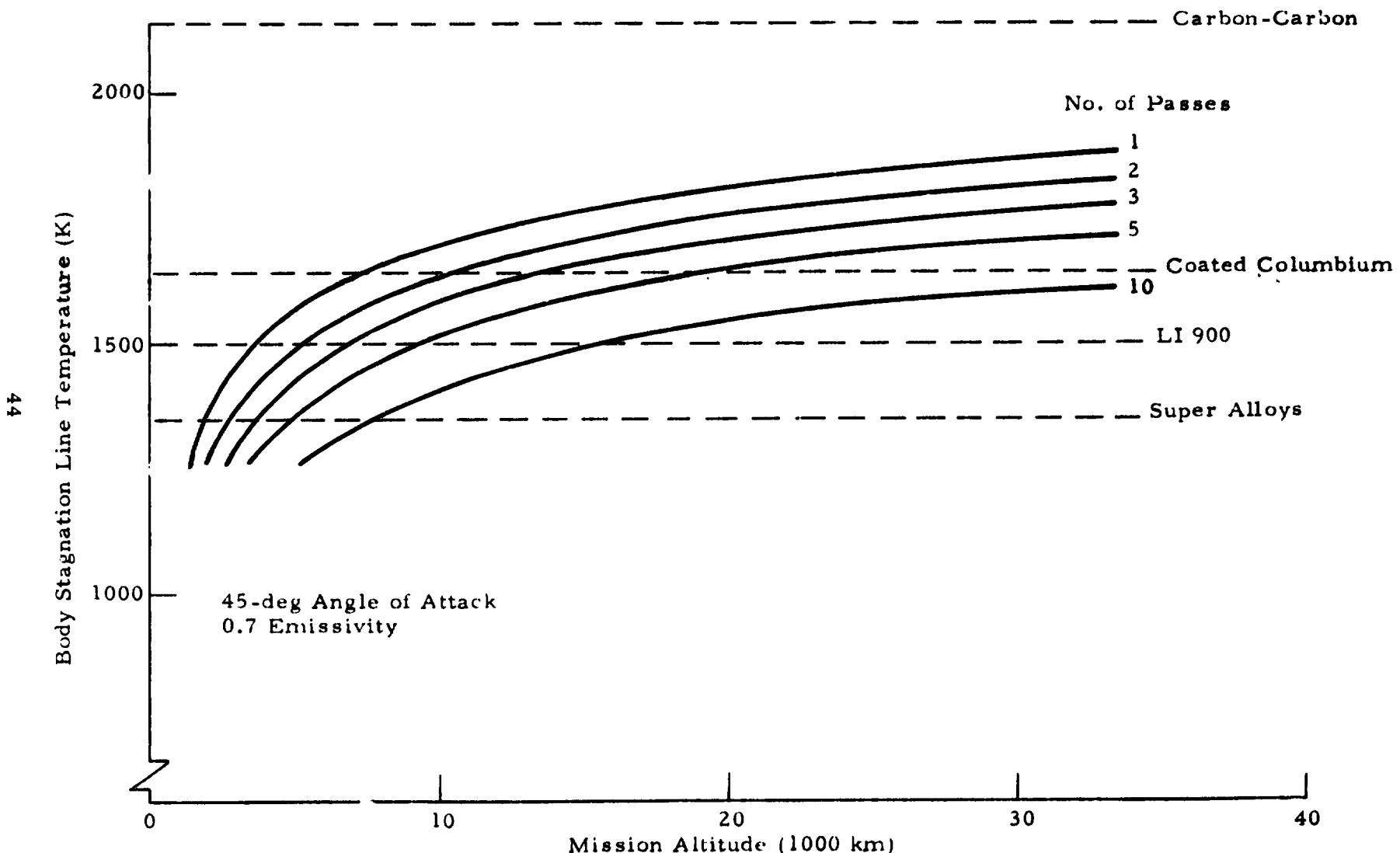
Fig. 21 - AMOOS Body Stagnation Line Temperature
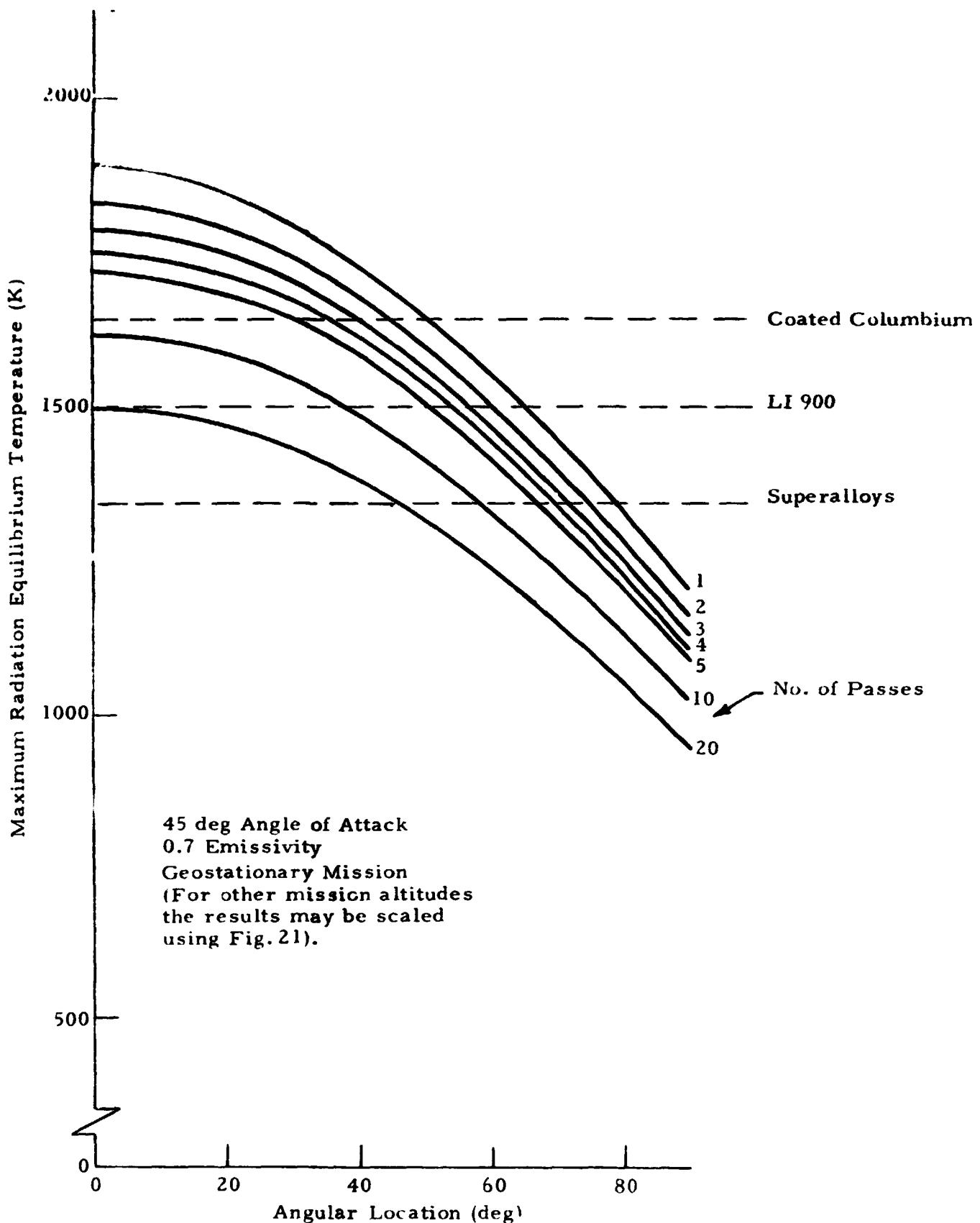
Fig. 22 - Surface Temperature as a Function of Location on AMOOS Body

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physical properties of the structural material must be held constant. This means that to optimize the structure to local thermal environment, several runs must be made and the radial matching of materials and thicknesses performed manually. Such a procedure was used to develop a near optimum AMOOS propulsion unit design using coated columbium on the hot spots and Rene' 41 within the allowable temperature range. The resulting primary structure weight, which is also the TPS less the internal insulation, is given in Table 5.

Table 5

COMBINED RERADIATIVE TPS AND PRIMARY STRUCTURAL WEIGHT FOR THE AMOOS PROPULSIVE UNIT

	Weight	
Components	(kg)	(lb)
Nose	721	(1590)
Forward Cylinder	495	(1092)
Intermediate Cylinder	555	(1223)
Rings	1038	(2289)
	2809	(6194)
+ 10% Contingency	281	(619)
Total	3090	(6813)

The resulting primary structure mass of 3090 kg (6813 lb) is considerably above the combined ablative TPS and magnesium (HM21A-T8) primary structure mass of 1022 kg (2254 lb). Figure 19 may be used to estimate the TPS and primary structure masses for other TPS material combinations. These masses do not include the mass of the internal insulator required to prevent the primary or sub-structure radiating heat to the internal components. This insulative mass increases with increasing temperature and is, therefore heavier for the reradiative TPS than for the ablator. The internal insulation for the titanium structure will be approximately the same as for the ablator.

Figure 19 was generated from the above analysis. The primary structure weight was computed using properties of the various materials corresponding to surface temperatures within the range of the respective material. The primary structure weights were then reduced to unit weights which may be used for preliminary design purposes. The unit weights for a given temperature change slightly from one section of the propulsion module to another since the structural loads vary with section. The unit weights of the ablative and a carbon-carbon TPS are given. The former is for an average over the entire body and so corresponds approximately to the middle of the temperature range, the latter for a single temperature.

3.2 HEAT SINK AND PHASE CHANGE TPS

An alternative to the conventional TPS considered previously is a system in which the heat is stored and dissipated slowly at some later time. Also included here is a TPS in which a change of state occurs, thus using the latent heat as a storage device. A study of physical properties of materials revealed that hydrogen has excellent properties for this purpose, except for its low density. However, hydrogen over a given temperature rise will yield the minimum mass of active material required for such a TPS. Furthermore, it will also yield a lower bound to the heat sink type TPS because its specific heat is high whether in the liquid or gaseous state. The potential, then, of each of these types of TPS can be determined by computing the mass of liquid hydrogen required to absorb the heat load. To make use of the temperature limits of currently available materials, not only is the LH_2 boiled but the gas is assumed raised to a moderate temperature, depending on the structural material. The temperatures chosen were well below the capabilities of aluminum and titanuim, the selected materials for this analysis. However, these are average gas temperatures and must be well below the wall temperatures for adequate convection. Furthermore, any practical design of the primary structure will involve varying material thicknesses, stringers and rings which may cause locally higher temperatures. For these reasons, the exhaust gas temperatures of 177 K (-124 F) and 334 K (140 F) are considered reasonable at this level of study.

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Heat loads were computed for return from geostationary altitude for
1, 3 and 10-pass maneuvers and over a range of altitudes up to geostationary
for the one pass maneuver. Three configurations were considered, namely,
the propulsion unit alone, the propulsion unit plus manned capsule and the
near maximum length vehicle. The corresponding lengths are: 10.4 m (34 ft),
13.9 m (45.6 ft) and 18 m (59 ft) respectively. The liquid hydrogen require-
ments were computed using the properties given in Table 6. No temperature
rise was considered in the liquid state. A hydrogen slush was not considered
since the hydrogen must be piped through narrow tubes near the surface to
obtain an even coverage over the entire body, in particular over the nose and
near the body stagnation line where maximum heating rates occur. Further-
more, the tubing must be kept thin so that the internal volume used is small.
Recall that, with AMOOS or an all propulsive tug, since each is transported
in the Shuttle cargo bay, internal volume is at a premium.

Table 6
PHYSICAL PROPERTIES OF HYDROGEN

Latent Heat of Vaporization

    at 20.5 K   454 joules/gram
       (36.1 R)   (194.3 Btu/lb)

Thermal Capacitance

   Liquid

    15 through 21 K   7.35 through 9.79 joules/gm/K
    (27 through 38 R)   (1.75 through 2.33 Btu/lb/F)
  (Use 2.33 for minimum mass of hydrogen)

   Gas

    14.45 joules/gram/K
    (3.44 Btu/lb/F)

Autoignition   859 K
    (4-75% by volume H_2-O_2 ratio)

The resulting weights of liquid hydrogen are given in Figs.23 through 26. As would be expected from earlier discussion, the mass of LH_2 required decreases slowly with decreasing mission altitude. On the other hand the mass of LH_2 required increases with increasing number of passes for the maneuver. This is due to the heat load increasing with number of passes per maneuver even though the heating rate decreases. Again, this aspect has been discussed previously.

A realistically low figure for the LH_2 required is approximately 1000 kg (2200 lb) for return from geostationary altitude and allowing a (333 K) 600 F temperature rise in the gas. This figure is to be compared to the ablator alone since the primary structure is required for both systems. It must be recalled that the LH_2 system requires extra tankage, plumbing, pump, manifold and an intricate heat exchanger in contact with the outer skin. This heat exchanger will consist of an inner skin at least thereby adding a minimum of 300 kg (660 lb) to the dry mass. In order for the LH_2 to be potentially competitive to an ablator, then a temperature rise of about 660 K (1200 F) is required or to a temperature of approximately 700 K (800 F). This temperature is approaching the autoignition temperature of 700 K (800 F). This temperature such temperatures require a primary structure of titanium alloy or one of the super alloys. This change in structural material adds a mass well in excess of the ablator mass and so destroys all possibility of finding an area in which the $LH_2 - GH_2$ TPS becomes potentially competitive.

Certain conclusions may be drawn from the above results for heat sink type TPS. Common materials have specific heats between 0.1 and 1 with many metals in the vicinity of 0.1 through 0.3. The specific heat of water is, by definition, 1 (taken strictly at 277 K (40 F)). Recall that the specific heat of GH_2 used for the preceeding calculations was 3.44. The primary structure, therefore, has negligible capacity compared to the GH_2. A heat sink TPS will weigh many times more than the $LH_2 - GH_2$ TPS unless some material with a specific heat superior to GH_2 can be found. Furthermore, this

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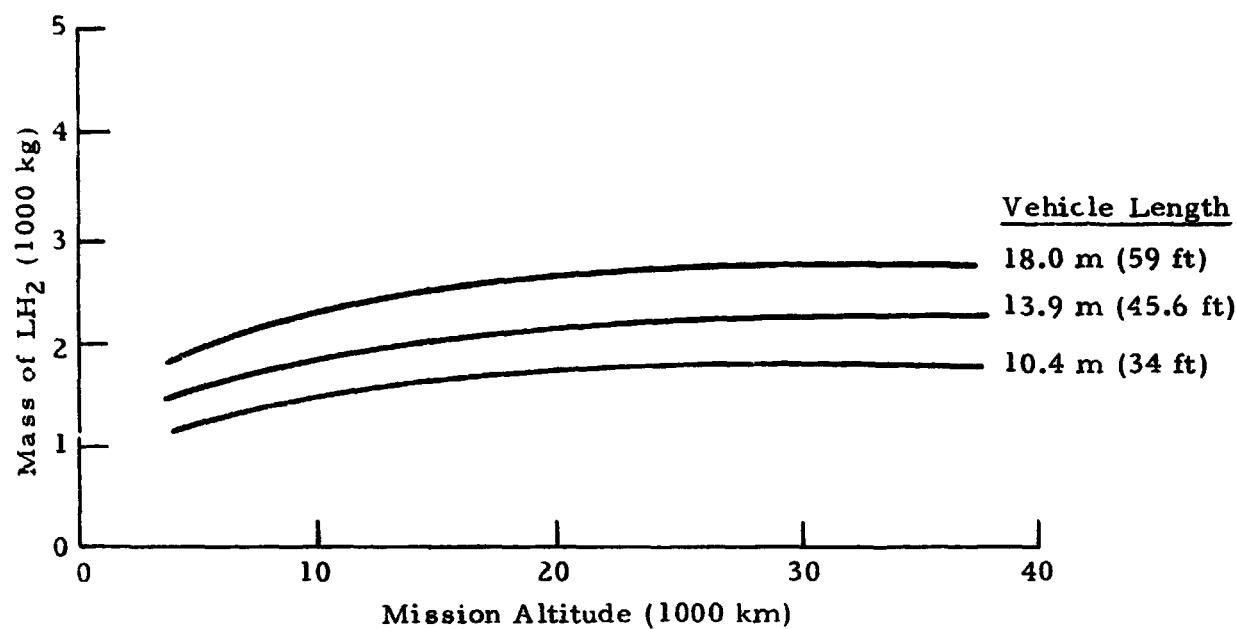
Fig. 23 - Mass of LH$_2$ Required to Cool AMOOS vs Mission Altitude
(167 K (300 $^{\circ}$F) Temperature Rise of the GH$_2$)
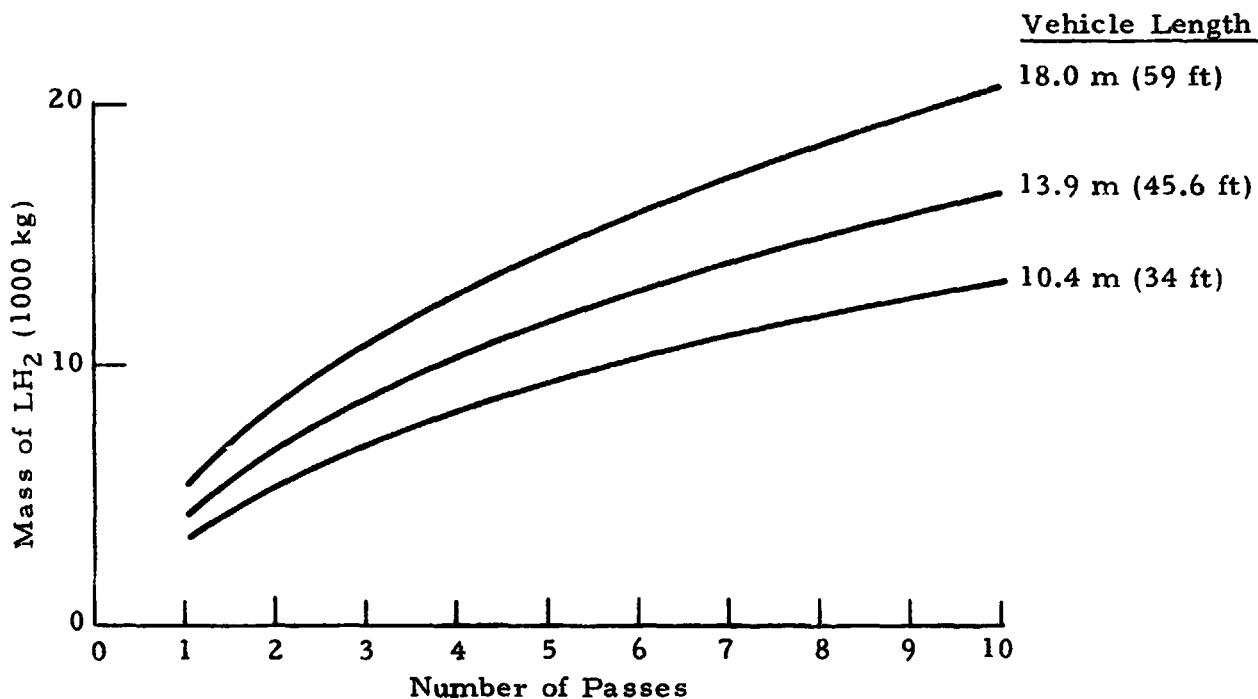
Fig. 24 - Mass of LH$_2$ Required to Cool AMOOS vs Number
of Passes (167 K (300 F) Temperature Rise of GH$_2$)

50
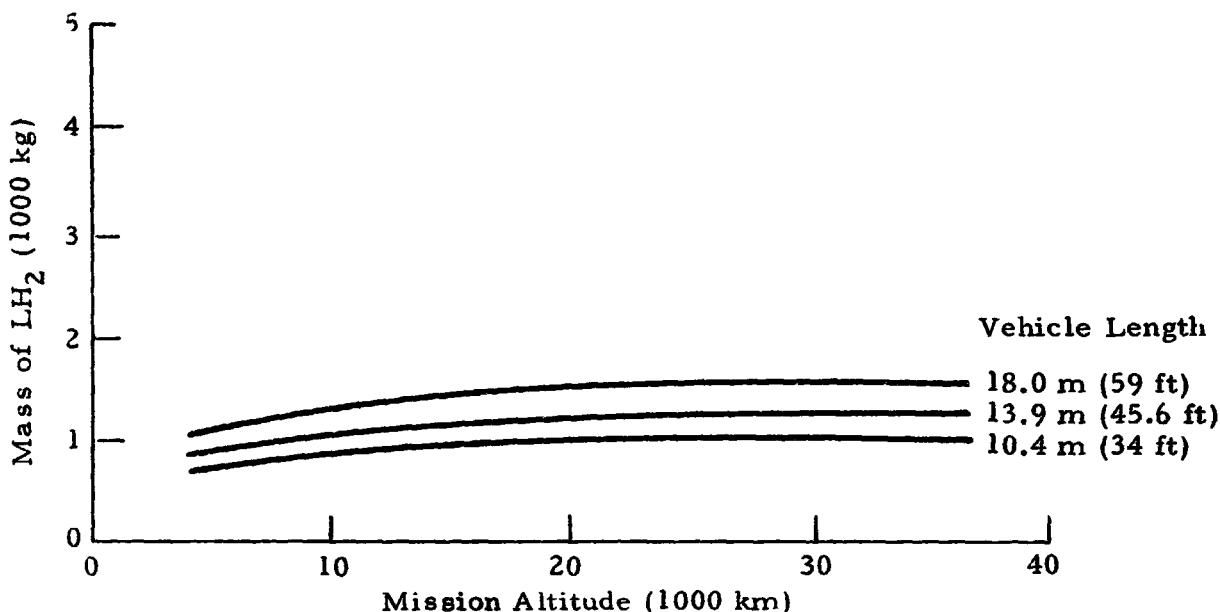
Fig. 25 - Mass of LH$_2$ Required to Cool AMOOS vs Mission Altitude
(334 K (600 F) Temperature Rise of GH$_2$)
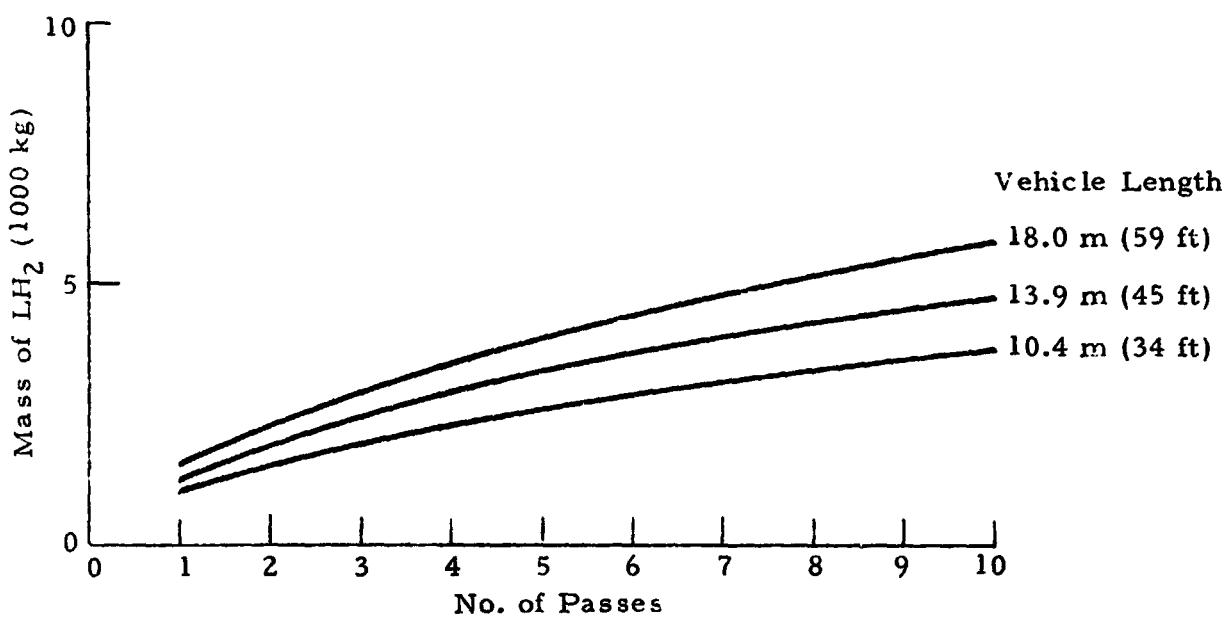
Fig. 26 - Mass of LH$_2$ Required to Cool AMOOS vs Mission Altitude
(334 K (600 K) Temperature Rise of GH$_2$)

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material must be capable of quickly dissipating heat evenly throughout itself. Either high thermal conductivity is required if a solid or otherwise it must be liquid. Gases, of course, are insufficiently dense to fit into the allowable volume.

3.3 HEAT PIPE TPS

The heat pipe TPS transports heat from the higher heating rate zones to the lower, thereby approximately equalizing the heating rates and hence temperature. Over the areas where there is a net influx of heat from the heat pipe, heat dissipation is by conduction into the atmosphere as well as by radiation. However, an estimate of the maximum temperature may be obtained by ignoring the conduction and considering all the heat loss to be by radiation. A typical average heating rate is 22.7 W/cm^2 (20 Btu/ft^2 sec). This heating rate would yield a temperature of 1500 K (2200 F) over the surface. From previous estimates (Ref. 3) the lee side temperatures are in the neighborhood of 700 K (800 F). At temperatures above 700 K (800 F) titanium alloys can no longer be used for the primary structure. Nor, of course, can it be used for the heat pipes. Referring to Fig. 19. shows that probably Rene' 41 or Haynes 188 must be used resulting in a 60 to 100% increase in the structural weight over that for titanium. Such a structure weight is some four times that for HM21A-T8 magnesium or beryllium-aluminum, or twice the weight of the ablative TPS plus the light weight alloy primary structure. A further weight increase will occur since the active substance must be contained in a tube. This could increase the structural weight at least 50% on allowing for thinner skins and load bearing tubes. Recall that the skin thickness currently is in the neighborhood of .6 mm (.025 in.) through 1 mm (0.04 in.). Halving these thicknesses, allowing for the reduction in rings and stringers and the threefold increase in inner and outer surface areas to form the tubes are taken into consideration in estimating the weight increase. Without increasing the dry weight for the active substance, a weight increase of approximately 1590 kg (3500 lb) to 1810 kg (4000 lb) is expected. These dry weight increases directly reduce the round trip payload capability. Furthermore, the payload itself must be protected

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with similar weight increases to the structure, namely some 680 kg (1500 lb). The resulting payloads of 900 kg (2000 lb) through 1130 kg (2500 lb) reduces the round trip payload capability to approximately that of the all propulsive cryogenic vehicle. From these brief considerations heat pipes can be eliminated as a viable TPS for AMOOS.

3.4 TPS SUMMARY

To date, no system is competitive with the lightweight ablator on a weight basis. In order to use reradiative materials, the peak heating rates and hence temperatures must be considerably reduced. This can be accomplished only by increasing the number of passes per maneuver or reducing the mission altitude range for AMOOS. Neither alternative is acceptable, the former because of the increase in orbit transfer time and the latter because of the importance of the geostationary mission. Furthermore, if the number of passes per maneuver is increased then the number of passes through, and the dwell time in, the Van Allen radiation belts is considerably increased. This latter condition is most undesirable for manned flight.

The TPS is, of course, one of the crucial areas for AMOOS. In fact, the total payload advantage can be attributed to the light weight ablator sprayed directly on the primary structure skin. In turn, the ablative TPS is made possible by trajectory control during the atmospheric flight. These two areas hold the key to AMOOS since each is unique to aeromaneuvering vehicles. Other areas important to AMOOS are also important to the all propulsive vehicle.

The above analysis has been based upon AMOOS. A similar analysis could be performed for the AMRS vehicle. The primary structure weights would suffer similar increases if insulative or reradiative TPS were used. Since the AMRS vehicle experiences higher heating rates and higher heat loads than AMOOS, the primary structure weight would increase accordingly. Furthermore, it would require more passes per maneuver, or a greater

reduction in mission altitude to be able to use a given material. Typically
AMRS weights may increase some 680 kg (1500 lb), and so taking it well
beyond the delivery capability of AMOOS to geostationary orbit. A passive
coolant, such as LH_2, would be catastrophic in the AMRS application since
it would require approximately one-half the LH_2 that AMOOS requires. The
LH_2 required may be roughly estimated by ratioing the ablator masses. This
may be used since the temperatures and heating rates are comparable and
both ablator and LH_2 mass required are heat load dependent. The method
slightly underestimates the AMRS requirements.

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Section 4
NAVIGATION HARDWARE REQUIREMENTS

Ideally, both AMOOS and AMRS should have autonomous navigation. This is, of course, true for any orbit transfer vehicle. In order to achieve automony, hardware somewhat more accurate than current off-the-shelf instrumentation is required. However, such hardware is under development. Two or three systems may be available within the next decade, which is approximately the OTV development period. These are: (1) the space sextant; (2) the interferometer tracker; and (3) the landmark tracker. Of these, the space sextant is probably the most versatile and universally applicable. Its action is similar to the regular sextant, but more versatile in that it can be used, in space. to determine the angle between charted stars and any detectable planet or the earth's moon. Its range is, for practical purposes, unlimited. The other two systems are for relatively near-earth applications. The interferometer tracker relies on the interference between airport radars. Since these radars are operated independently of the spacecraft and continuously, the system may be considered autonomous. Apparently such a system can be used from low-earth orbit through geostationary. Its applicability is limited only by direct line of sight from the spacecraft to two or more airport radars. The third system is the landmark tracker. This is currently available as a short range instrument. However, the current range of approximately 10,000 km (5000 n.mi.) curtails its usefulness with OTVs which typically operate up to geostationary altitudes. Furthermore, the all weather capability of the landmark tracker is open to question.

Other autonomous systems exist, but are insufficiently accurate. Such systems, such as the inertia measuring unit (IMU), have sufficient accuracy for short times after an update and other desirable features. The IMU gives essentially continuous data and so is invaluable during most phases of the mission. It must be updated frequently by more accurate hardware. In

general, the navigation system will consist of several independent subsystems such as a star tracker, horizon sensor, IMU, space sextant, etc., all tied together by the onboard digital computer. Such an onboard computer must be capable of filtering the data from several instruments simultaneously and obtain a best estimate trajectory for use in exoatmospheric targeting and guidance. During atmospheric flight, probably only the IMU can be used since the thermal environment will interfere with the operation of any system requiring signals external to the spacecraft. The accuracy of typical navigation systems will now be discussed in relationship with the corridor requirements of AMOOS and AMRS.

4.1 NAVIGATION ACCURACY REQUIREMENTS

The critical navigational accuracies for AMOOS and AMRS are during the return transfer phase of the mission. The guidance studies (Ref.3) showed that a corridor width of ± 6 km (± 3.25 n.mi.) would be acceptable with a little fine tuning of the guidance system gains. The nominal corridor was ± 3.5 km (± 2 n.mi.), 3σ.

The accuracies with which the vehicle's position and velocity are known immediately prior to the burn into return transfer orbit are probably highly mission dependent even for identical nominal mission altitudes. For example, if AMOOS is leaving a Space Station, then the position and velocity of the Space Station is probably known accurately, to the point where the errors are negligible. On the other hand, if AMOOS delivers a payload without reference to a Space Station or spacecraft, then the errors prior to burn may be significant, perhaps as high as 50 km (27 n.mi.) and 10 m/sec (33 ft/sec), 3σ (Ref. 4). For the purpose of these studies, a 1σ position uncertainty of 10 km (5 n.mi. approximately) and a 1σ speed uncertainty of 0.6 m/s (2 ft/sec) (Ref.4) is used. These errors correspond to the minima that can be obtained with a horizon sensor at geostationary altitude. However, to obtain this accuracy requires several hours of sensor data. The space sextant may provide angular data

about an order of magnitude more precise than the horizon sensor so that
realizing these accuracies in a short on-orbit stay is realistic.

4.2 EVALUATION OF THE SPACE SEXTANT

In'tial studies (Ref. 7) of the space sextant have shown that 1σ position
accuracies of the order of 1.85 km (1 n.mi.) can be obtained. These accu-
racies have been verified to a certain extent by onboard experiments in past
space flights. Gemini, Apollo and Skylab flights have been used for the asto-
nauts to take hand-held space sextant sightings of various stars. The short-
comings of this system have been in the areas of ease of operation and the
difficulties associated with rather restricted fields of view from spacecraft
windows. On the technical side, some difficulty occurs when using the illum-
inated limb of the moon or other bright object due to an apparent increase in
diameter due to the brightness.

The Space Sextant can probably meet a 3σ corridor of ± 6 km (± 3.25 n.mi.)
which is about the maximum capability of the guidance scheme as currently
developed.

4.3 LANDMARK TRACKER AND INTERFEROMETER TRACKER EVALUATION

The evaluation of the landmark tracker relies heavily on Ref. 8. In
that document considerable attention is given the navigation task, including
hardware and filtering.

The landmark tracker has a useful range of approximately 10,000 km
(≈ 5000 n.mi.) and thus cannot be used until the final 2000 sec before perigee.
This has several undesirable implications. The first is that by this time the
1σ position uncertainty will have exceeded 50 km (≈ 25 n.mi.). With uncer-
tainties of this magnitude, the Kalman filter is unlikely to converge (Ref. 8).
Another instrument, such as a horizon sensor, is required to reduce the un-
certainty before the earth is within the range of the landmark tracker.

The second disadvantage is that 2000 sec (~30 min.) is a short time in which to collect data, filter and make a trajectory correction. Such a short time is probably insufficient to recycle the midcourse correction sequence. Furthermore, performing the midcourse correction this close to the earth is relatively inefficient. A Δv of 120 to 140 m/sec (400 to 460 ft/sec) must be budgeted instead of 15 m/sec (50 ft/sec) if performed in the first 12,000 sec (200 min.) of the return transfer flight. The Δv requirement as a function of time from return orbit transfer insertion is given in Fig. 27. Also given is the radial perigee position uncertainty after horizon sensor and landmark tracker updates. Finally, the all-weather capability of the landmark tracker is open to question. The literature contains some discussion of tracking cloud formations. This is feasible since clouds move relatively slowly and the location of the landmark is not a prerequisit. If it is known, then the convergence of the filter is quicker and potentially more accurate.

The potential capability of a horizon sensor-landmark tracker system, in combination with a strapdown IMU and a star tracker is given in Fig. 28, extracted from Ref. 8. The 1σ uncertainty in position at atmospheric entry is 0.65 km (0.35 n.mi.). In practice, these theoretically possible accuracies frequently cannot be achieved. A realizable 1σ accuracy is probably of the order of twice this value. This inability to achieve the theoretical accuracy is due probably to several factors which includes the inability to model satellite motion exactly.

The interferometer tracker was considered, but sufficient data were not available to evaluate the method as applied to AMOOS. This method has the potential for application to AMOOS and AMRS and should be evaluated.

4.4 NAVIGATION HARDWARE SUMMARY AND DISCUSSION

The most promising navigation system considered to date consists of the space sextant, with, of course, an IMU, and star tracker. With this system, a horizon sensor is probably not necessary. Such a system with
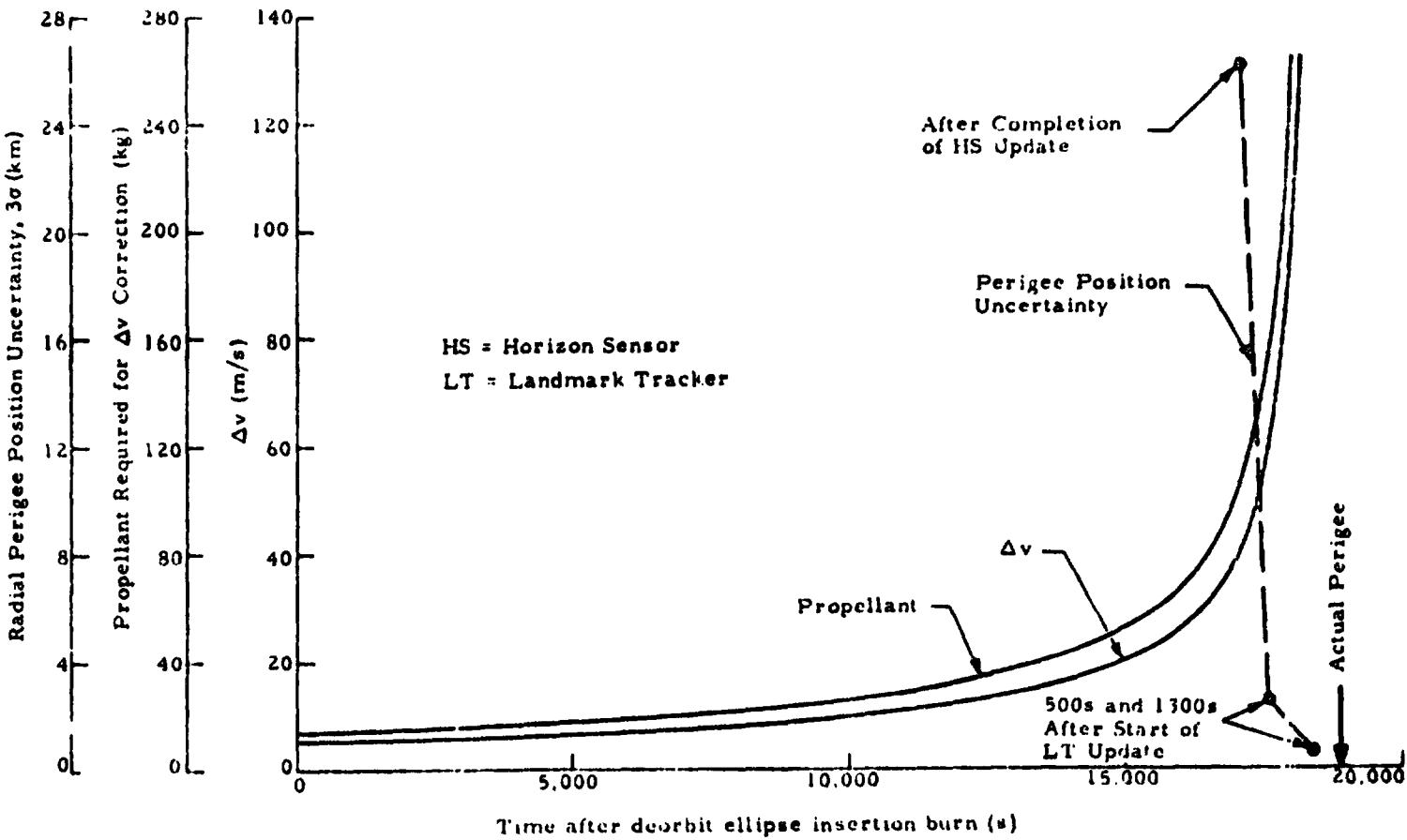
1 g.27 - Δv, Propellant Consumption for Midcourse Correction and Radial Position Uncertainty for Return from Geosynchronous Orbit
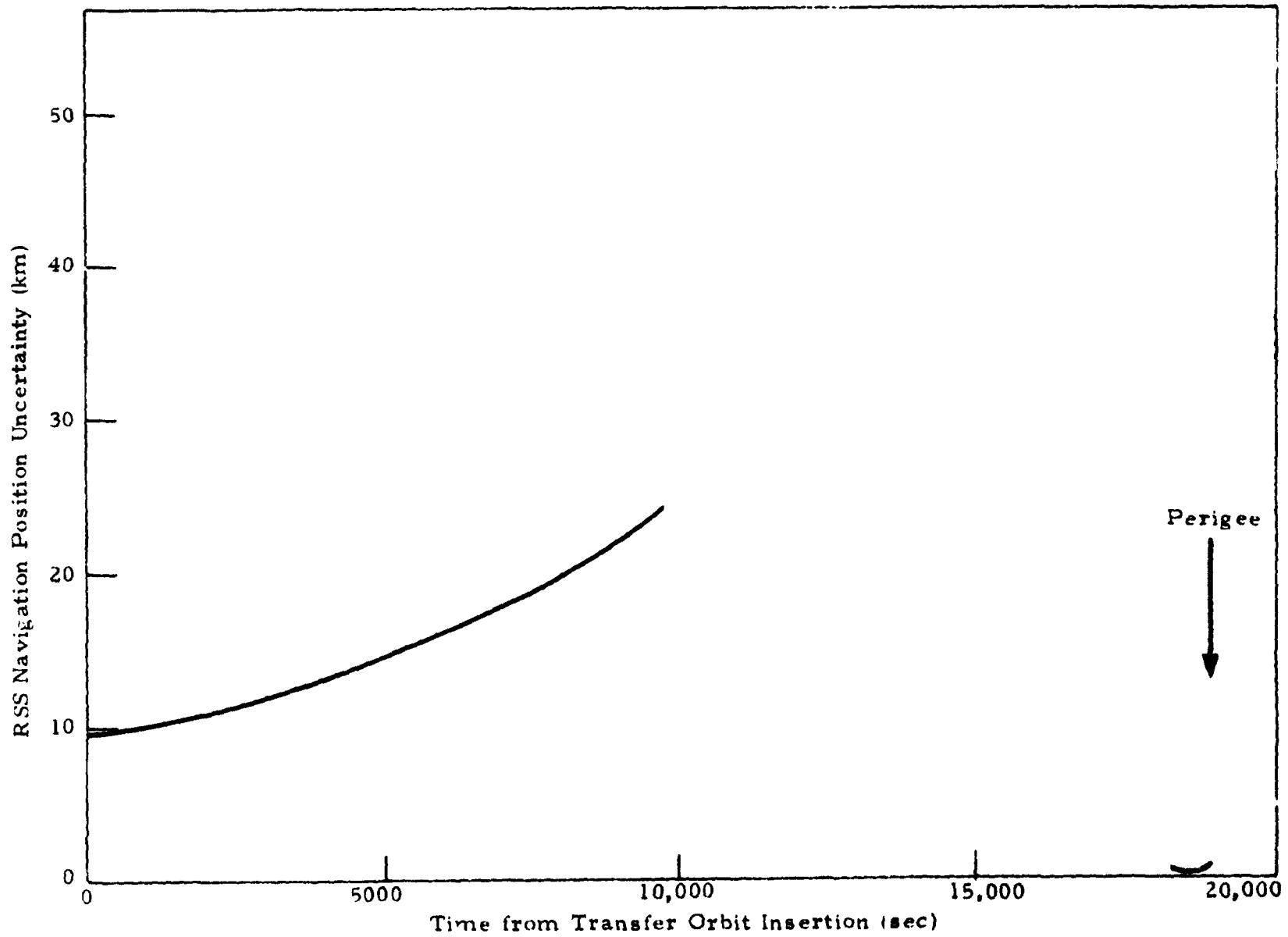
Fig. 28 - One-Pass Navigation Uncertainties (Ref. 7) Using Filtered Horizon Sensor and Landmark Tracker Data

an onboard computer and filtering software can be expected to require, at worse, a ± 6 km (± 3.5 n.mi.) entry corridor. Such a corridor is within the capability of the AMOOS guidance after the gains are appropriately adjusted.

The interferometer tracker and the landmark tracker offer alternatives that apparently are sufficiently accurate. Other alternatives may be feasible but are not autonomous. One method is a series of ground based beacons with which the spacecraft can communicate. Other methods that hold promise are the use of navigation satelites, which may be operation by the middle 1980s, and communication with the Shuttle.

Section 5

AMOOS/AMRS

5.1 GROUND RECOVERY OPTIONS FOR AMRS

5.1.1 Recovery Techniques

In its current configuration, AMRS is not capable of a horizontal airplane-type landing. Therefore, only uncoupled recovery techniques may be used for the ground recovery mode. These include parachutes, rotors, Regallo wings, etc. Of these devices, only parachutes have been used in the recovery of not only spacecraft but also of aircraft dropped stores, etc. The other methods have been studied extensively but pose difficult problems in packaging and deployment. For these reasons, and frequently a weight penalty, parachutes have been recommended and used for recovery of spacecraft. The reasoning applied to the selection of parachutes over other devices is considered to apply also to the uncoupled recovery of AMRS. The AMRS ground recovery trajectory consists, therefore of a guided reentry to approximately 7000 m (23,000 ft) at which altitude parachutes are deployed. The parachutes slow the vehicle to a terminal vertical speed of approximately 9 m/sec (30 ft/sec).

An alternative to recovering the complete vehicle is to recover the manned unit only. This is particularly appropriate if a solid motor is used for the orbit transfer maneuver to leave mission altitude. Each of the design alternatives may, therefore, be for the recovery of the complete vehicle or the manned unit only.

The recovery options are:

1. Land Recovery
2. Water Recovery
3. Air (Snatch) Recovery.

Each will be discussed in the following subsections.

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5.1.2 Land Recovery

In this mode, AMRS will impact the land at some point. Since AMRS is an emergency vehicle, targeting to a precise area would be too restrictive on the system. However, targeting to a general area may be practical. In the studies so far, AMRS is inserted into a 28.5 deg inclination transfer orbit. This gives AMRS the inherent capability of impacting within the United States provided the phasing capability is sufficient. For the purposes of this report, such will be assumed. Since AMRS is an emergency use vehicle, taking the risk of landing on unsuitable terrain or under inclement conditions is acceptable. Since the AMRS has moderate maneuverability at supersonic and hypersonic flight, some targeting can be accomplished so that features up to several thousands of square miles in area can be avoided. On the other hand, features with an area of several hundreds of square miles can be impacted.

The landing forces on the crew must be attenuated in some manner. This may be accomplished either by slowing the vehicle to a speed well below 9 m/sec (30 ft/sec) or by inserting some cushioning or crushable material somewhere in the load path from the ground to the crew. Some work has been done on the use of retro rockets to reduce the impact speed. However, the considerations here will be of methods using cushioning or crushing material. The retro rocket approach has been studied for application to other projects. The rocket must be fired at a precise height above the ground, the firing height is, of course, speed dependent. Furthermore, a ground impact attenuation system may still be required to absorb the residual energy, or even the total energy should the rocket fail. The retro rocket approach is eliminated because of the above complexity and duplication. Also eliminated as undesirable is requiring the crew to make a parachute jump for final recovery. All methods have been eliminated except the deformable structure to attenuate the impact. The design of such a system will now be discussed.

3.1.3 Recovery Sequence

The AMRS vehicle would make a guided reentry to approximately 7000 m (23,000 ft) at which point the main chutes would be deployed reefed. Two disreefings of the main chutes would be used with the chutes being fully deployed at about 3000 m (10,000 ft). Three main chutes are recommended to provide adequate redundancy. The design impact speed is 11 m/sec (36 ft/sec) which corresponds to two chutes deployed, one failed. The stroke of the attenuation system is given by

$$S = \frac{1}{2} \frac{u^2}{ng}$$

where u is the vertical component of the vehicle's velocity, ng is the deceleration and g is the gravitational acceleration on the earth's surface. The duration of the deceleration is given by

$$t = \frac{u}{ng} .$$

As an example, consider the case of 10 g deceleration

$$S \approx \frac{1}{2} \frac{(11)^2}{98.1} \text{ m} \left(\frac{1}{2} \frac{(36)^2}{320} \text{ ft} \right)$$

$$\approx 0.6 \text{ m (2 ft)}$$

The duration of the deceleration is

$$t = \frac{11}{98.1} = 0.112 \text{ sec.}$$

The AMRS design of Ref. 3 can be readily modified to allow a 0.9 m (3 ft) shock attenuation stroke. The maximum vertical component of the AMRS velocity is 14.6 m/sec (48 ft/sec). If a peak deceleration of 35 g is stipulated then the required stroke is

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$$S \sim \frac{1}{2} \frac{(14.6)^2}{35 \times 9.81} \text{ m} \left(\frac{1}{2} \frac{(48)^2}{35 \times 32} \text{ ft} \right)$$

$$= 0.31 \text{ m} \quad (1.02 \text{ ft})$$

The duration of the deceleration would be

$$t = 0.043 \text{ sec.}$$

The terminal velocity with three chutes deployed will be approximately 9 m/sec (30 ft/sec), with two, approximately 11 m/sec (36 ft/sec) and with only one 14.6 m/sec (48 ft/sec). The aim is to design an attenuation system with a stroke of less than 0.9 m (3 ft) capable of limiting the deceleration to 35 g for the 14.6 m/s (48 ft/sec) case and to 15 g for the 9 and 11 m/sec (30 and 36 ft/sec) cases. The stroke required to attenuate the shock of the 11 m/sec (36 ft/sec) impact to 15 g is

$$S_{36} = \frac{1}{2} \frac{(11)^2}{15 \times 9.81} \text{ m} \left(\frac{1}{2} \frac{(36)^2}{15 \times 32} \right)$$

$$\approx 0.41 \text{ m} \qquad (1.35 \text{ ft})$$

The duration of the deceleration is

$$t_{36} = \frac{11}{15 \times 9.81}$$

$$\approx 0.075 \text{ sec}$$

When the initial velocity is 14.6 m/sec (48 ft/sec), the velocity at the end of the 15 g deceleration stroke of 0.41 m (1.35 ft) is given by

$$v = \sqrt{u^2 - 2 \, n g S}$$

which in this case is

$$V = \sqrt{(14.6)^2 - 2 \times 15 \times 9.81 \times 0.61} \text{ m/s} \left(\sqrt{(48)^2 - 2 \times 15 \cdot \times 1.35} \text{ ft/sec} \right)$$

$$= 9.68 \text{ m/sec} \ (31.75 \text{ ft/sec})$$

The stroke required to attenuate 9.68 m/sec (31.749 ft/sec) at 35g is

$$S = \frac{93.65}{2 \times 35 \times 9.81} \text{ m} \left(\frac{1008}{2 \times 35 \times 32} \text{ ft} \right)$$

$$= 0.14 \text{ m} \ (.45 \text{ ft})$$

The total stroke length is

$$S_{48} = 0.41 + 0.14 \ (1.35 + 0.45)$$

$$= 0.55 \text{ m} \ (1.80 \text{ ft}).$$

The duration is

$$t_{48} = \frac{14.6 - 9.68}{15 \times 9.81} + \frac{9.68}{15 \times 9.8} \quad 0.0622 \text{ sec}$$

comprising of 0.0339 sec at 15 g followed by 0.0283 sec at 35 g. If up to 0.9 m (3 ft) stroke is allowable, then considerable variation from this minimum stroke design is possible.

The design terminal descent rates are 9 (30), 11 (36) and 14.6 (48) ft/sec, respectively, with three, two and one main chute deployed. These chutes may be sized approximately on assuming a C_D = 1.26, which corresponds to approximately a 20% porosity. Lower porsity will yield a higher C_D but introduce a

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tendency to oscillate. The main chutes may be sized by equating the drag to the gravatational force, namely

$$3 \cdot \frac{1}{2} \rho V^2 A C_D = mg$$

For a weight of 3175 kg (7000 lb), 9 m/sec (30 ft/sec) descent rate at sea level, this yields

$$A = \frac{2 \times 3175}{3 \times 1.23 \times (11)^2 \times 1.26} \ m^2 \left(\frac{2 \times 7000}{3 \times 0.00238 \times 900 \times 1.26} \ ft^2 \right)$$

$$= 160.7 \ m^2 \ (1730 \ ft^2) \ \text{approximately}$$

or

$$d = 14.3 \ m \ (47 \ ft) \ \text{approximately}$$

for the inflated frontal area and diameter, respectively. The surface area of the main chutes will be approximately 50% larger than A or approximately 241.6 m^2 (2600 ft^2). These are areas and dimensions for each of the three main chutes.

A drogue chute is not required, since the main chutes may be deployed reefed at 4310 N/m^2 (90 lb/ft^2). However, a drogue chute was sized for the AMRS using the foregoing pressures. A drogue chute is usually a high porosity chute, probably between 20 and 40%, which reduces the drag coefficient to about one. Using the $C_D = 1$, q = 4310 N/m^2 (90 $lb/sec-ft^2$) and an AMRS weight of 3175 kg (7000 lb) gives a drogue chute inflated frontal area, A_D, of

$$A_D = \frac{3175 \times 9.81}{4310} = 7.23 \ m^2 \ (77.8 \ ft^2)$$

or corresponding inflated frontal area diameter

$$d_D = 3.03 \ m \ (9.95 \ ft)$$

if one drogue chute is used. However, if the AMRS vehicle is stable in the subsonic regime, then a terminal q of approximately $1440 \ N/m^2$ (30 lb/ft^2) can be achieved. This eliminates the requirement for a drogue chute. If the configuration proves unstable or untrimmable at higher angles of attack then a small drogue chute, 1.5 m (5 ft) diameter or less, may be required to stabilize and trim the vehicle at the lower end of its speed range.

The main chutes are deployed reefed in order to reduce the peak loads on the AMRS vehicle. Assuming a q of 1440 N/m^2 (30 lb/ft^2) at main chutes deployment the load is given by

$$1440 \times 1.26 \times F \times 3 \times 160.7 \ N$$
$$(30 \times 1.26 \times F \times 3 \times 1730 \ lb)$$

$$= 873,000 F \ N = (196,182 \ F \ lb)$$

Using an F of 0.14 gives a force of approximately 122,200 N (27,470 lb) or a load factor of 3.92 g; Γ is the reefed area ratio.

The terminal dynamic pressure (q_{r1}) in this reefed condition is

$$q_{r1} = \frac{3175 \times 9.81}{0.14 \times 1.26 \times 3 \times 160.7} \ N/m^2 \left(\frac{7000}{0.14 \times 1.26 \times 3 \times 1730} \right) lb/ft^2$$

$$= 370 \ N/m^2 \ (7.65 \ lb/ft^2)$$

Disreefing to a 0.4 condition at say 380 N/m^2 (8 psf) gives a load of

$$93,120 \ N \ (20,926 \ lb)$$

or a load factor of 2.99 g.

The terminal dynamic pressure, q_{r2}, in this second reefed condition is

$$q_{r2} = \frac{3175 \times 9.81}{.4 \times 1.26 \times 3 \times 160.7} \text{ N/m}^2 \left(\frac{7000}{.4 \times 1.26 \times 3 \times 1730} \text{ lb/ft}^2 \right)$$

$$= 130 \text{ N/m}^2 \ (2.7 \text{ lb/ft}^2)$$

Complete disreefing at $q = 140 \text{ N/m}^2$ (3 lb/ft^2) yields a load of

$$939,000 \text{ N} \ (19,600 \text{ lb})$$

or a load factor of 2.80 g.

The design load for the main chutes will be 4 g since this covers all the above cases.

5.1.3 Water Recovery

The foregoing discussion of the parachute deployment and descent applies equally well to a water recovery of the complete vehicle. In this case the water yields to provide the necessary attenuation of the impact forces. However, the AMRS vehicle will need be oriented correctly for the impact, a 45-deg angle, nose down altitude is currently recommended. The reasons for this recommendation is: (1) to orient the impact force vector in a favorable direction to the crew, and (2) to provide a reduction in the impact forces by reducing the impact area.

The density of the AMRS vehicle is low, of the order of 48 kg/m^3 (3 lb/ft^3), so that floatation is no problem provided the vehicle is watertight. Since the density is so low, the vehicle will float high, with only a few inches submerged. This will simplify the floatation problem since any interconnectors through the aft bulkhead may be above the waterline. This may be so even if the vehicle rotates about its longitudinal axis when floating.

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A more severe problem may be the roll stability of the vehicle when floating. The elliptical cross section will yield some roll stability. However, there may be two stable positions, each with the major axis horizontal but one upside down from the seated crew position. Since the center-of-gravity is expected to be near the centerline, one stable position will not be preferred to the other by the cross section. However, the nose shape will give a preference for the crew upright position.

Prior to and during the crew egress the vehicle must be rotated to and held with the crew hatch out of the water. This corresponds to the crew being in the right side up position when seated.

5.1.4 Air (Snatch) Recovery

In this mode, only the crew capsule will be recovered. This is to reduce the weight to be recovered to approximately one-half the total weight of the AMRS vehicle upon reentry, or namely some 1488 kg (3283 lb) including crew. The weight estimate for this configuration is given in Table 7.

Since the capsule weight is reduced, then the parachute area may also be reduced. However, it may be desirable to reduce the rate of descent to increase the time available for snagging. If the 14.3 m (47 ft) diameter chutes are used in this case, the descent rate will be reduced to approximately 6.7 m/sec (22 ft/sec) and so increase the time for snagging by approximately 40%.

To reduce the loads on the manned capsule, a friction winch or similar load limiting device should be used. This limited load accelerates the AMRS capsule to aircraft speed. The weight of the capsule is approximately 1500 kg (3300 lb). To limit the acceleration on the capsule to 4 g requires the load limited to 60000N (13,200 lb). The overshoot on initial snatch is usually of the order of twice the steady state load. The friction load should be set at approximately 31,000 N (7000 lb). The current air recovery technique using a pair of arms mounted on the nose practically ensures that a clean capture is made and that the parachutes are collapsed. The friction winch should be mounted

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Table 7

WEIGHT ESTIMATE FOR THE STRIPPED AMRS
CONFIGURATION FOR AIR RECOVERY

	m^3	(ft^3)	kg	(lb)
Crew, Four @ 56 ft^3	6.34	(224)	339	(748)
Food 8 lb/ft^3	0.057	(2)	7	(16)
Furnishings 2 lb/ft^3	1.22	(43)	39	(86)
Medical 10 lb/ft^3	0.057	(2)	9	(20)
Personnel Effects	0.283	(10)	25	(56)
EC/LSS			335	(738)
Atmosphere	0.057	(2)		
Water 62 lb/ft^3	0.028	(1)		
Wastes Management	0.113	(4)		
Hardware	0.283	(10)		
Electronics	0.113	(4)	59	(130)
Communications and Data System	0.283	(10)	148	(327)
Instrumentation	0.453	(16)	85	(188)
Miscellaneous Equipment	0.382	(10)	9	(20)
Expendables	0.085	(3)		
Crew Capsule			230	(507)
Shell Structure			70	(154)
Contingency 10%			133	(293)
Total*			1488	(3283)

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*Weight of parachutes not included; weights converted and rounded
individually so that totals may not convert exactly.

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between the capsule and the parachutes. The length of cable required to accelerate the capsule to an aircraft speed of 60 m/sec (200 ft/sec) is

$$S = Ut - 1/2 \ 0.2 \ g \cdot t^2$$

where
$$t = \frac{60}{2g} \left(\frac{200}{2g}\right)$$

$$\therefore S = \frac{60^2}{2g} - 1/2 \cdot 2g \cdot \frac{(60)^2}{(2g)^2} = \frac{(60)^2}{4g} = \frac{900}{g} \text{ m}$$

92 m (300 ft)

Perhaps the more difficult task is man-rating the system. A problem arises since the system could snag and damage the chutes without obtaining a positive hold on the system. This could cause a catastrophic failure.

Another problem may arise should the chutes not collapse. The force at 60 m/sec (200 ft/sec) will be approximately 133,500 N (30,000 lb). This is not an inconsiderable fraction of the total engine thrust on, say, a C-130 aircraft. Furthermore, the chutes would fail at well below this load unless an excessive weight penalty were accepted. In general, the above tasks and problems that may or will arise makes the system unattractive for manned systems.

5.1.5 Selection of a System

Currently, the land impact system appears most attractive. Furthermore, this system could, in an emergency impact in water. Since the AMRS is an emergency system, phasing in orbit for a land impact may be undesirable. Therefore a system with both capabilities is desirable. Alternatives to the land impact configuration discussed above are: (1) recovering the manned capsule only, and (2) having the crew jump, each with a parachute. Of these, only recovering the manned capsule appears worthy of further

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consideration. such parameters as stroke length of the impact attenuation system are the same for the capsule as for the complete vehicle. The total parachute area required will decrease in direct ratio to the recovered weight. Hence the impact attenuation system and parachute system will be lighter. However, offsetting this decrease in weight will be the separation mechanism. Since the total weight of the parachute and shock attenuation system for recovery of the complete vehicle is some 600 lb, the separation system must be lightweight to show a favorable weight trade. The desirability of the capsule only recovery is further decreased by the increase in complexity and a corresponding decrease in reliability.

5.1.6 AMRS Ground Recovery Conceptual Design

A load factor of 4.0 was applied to the AMRS static weight of 3175 kg (7000 lb) to determine the snatch load for parachute deployment. A safety factor of 2.0 was also used to account for uncertainties. This is larger than what is normally used for manned vehicles but is not unduly conservative for the load analysis in this case. Using these factors, the total load to be taken by the parachute cable is:

$$P = Wt(LF)(SF) = 3175 \times 4 \times 2 \times g = 249175 \text{ N}$$

$$= (7000(4)(2) = 56,000 \text{ lb})$$

A deployed cable length of twice the AMRS vehicle length was assumed

$$\ell_{cable} = (2)\ell_{AMRS} = 2 \times 7.62 = 15.24 \text{ m}$$

$$= (2 \times 300 = 600 \text{ in.})$$

A cable size adequate to carry the above load was chosen from a table of Aircraft Stainless Steel Cable (Ref. McMartin & Carr catalog).

The allowable load for a 2.22 cm (7/8 in.) diameter flexible (7 x 19 construction) cable is P_{allow} = 295925 N (66,500 lb)

$$\therefore MS = \left(\frac{295925}{249175} - 1 \right) = +0.19$$

The cable weight was determined to be 49 kg (108 lb) on assuming the cable solid and its density to be 0.083 kg/cm^3 (0.3 lb/in^3). The weight for the storage box and deployment mechanism was assumed to be 45 kg (100 lb).

The parachute and deployment mechanism was located at the top of the vehicle above the nozzle bell as shown in Fig.29. A truss structure on either side of the deployment box and intercostals along side of the box were used to take the concentrated parachute load into the vehicle shell. Longitudinal intercostals were also added between the ring at the oxidizer tank and the aft ring that was added for this installation.

The forward parachute cable attachment is at the interface ring between the manned portion of the vehicle and the aft propulsion section. Intercostals were added between the aft attachment ring for the manned capsule and the fuel tank attachment ring. Eight longerons are equally spaced around the vehicle.

The truss structure at the aft end of the vehicle was designed to take the full load on either side. Therefore, the load in each of the three members was

$$P_{member} = 249,175/3 = 83,058 \text{ N}$$
$$(56,000/3 = 18,666 \text{ lb})$$

A 4.45 cm (1.75) o.d. x 0.64 cm (0.25 in.) wall thickness 4130 steel tube was selected for the truss members. The stress level in the member is

$$f = P/A = 83,058/7.60 = 10,929 \text{ N/cm}^2$$
$$= (18,666/1.178 = 15,845 \text{ lb/in.}^2)$$

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**Primary Structure from
Optimized Liquid AMRS**

Cable, Longerons
and Intercostals
Stressed for 4g
plus a Safety
Factor of 2.
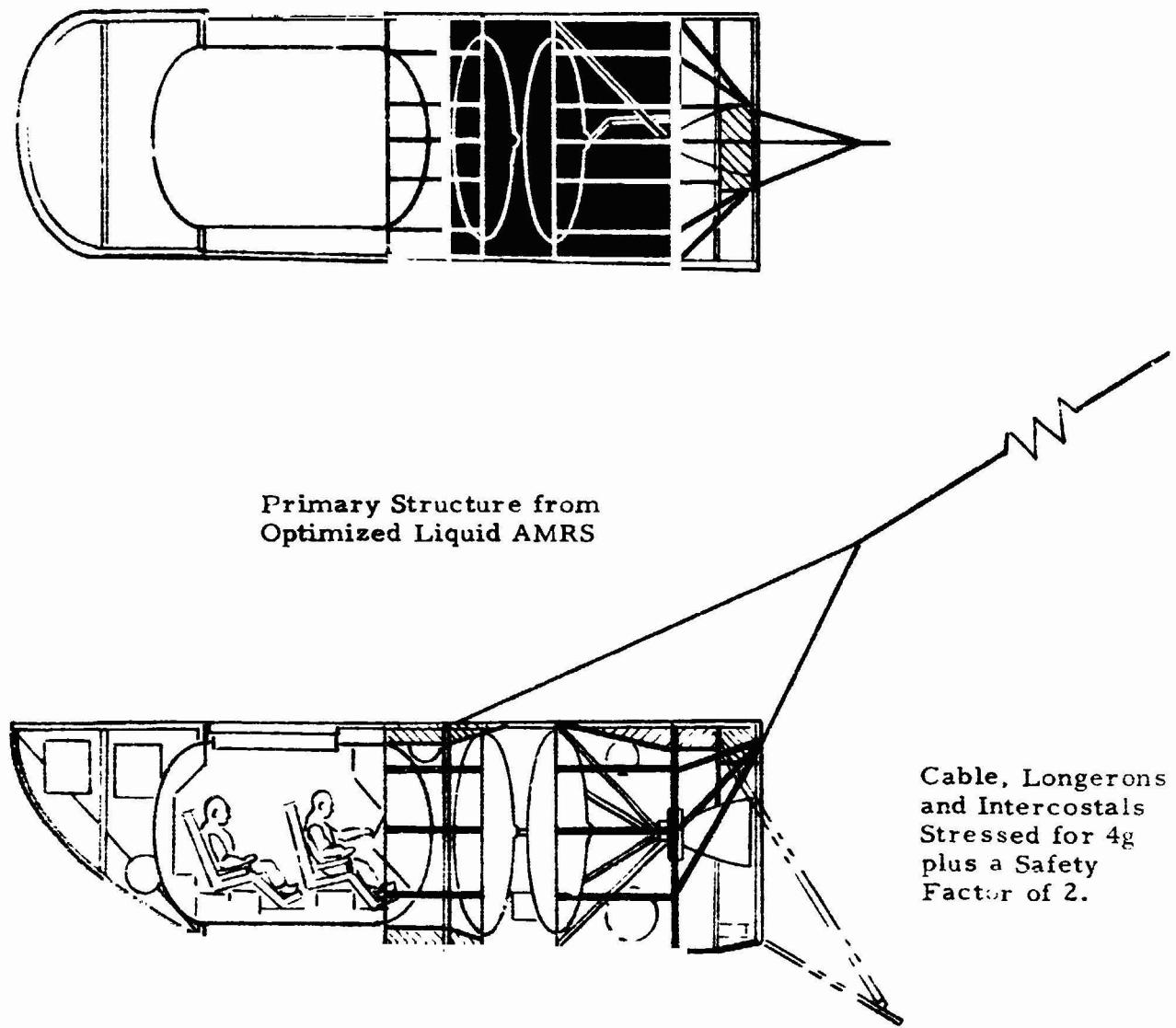
Fig.29 - Conceptual Design of Ground Recoverable AMRS
(Liquid Main Engine)

The critical column load for the longest truss member is

$$Pa = \frac{\pi^2 EI}{\ell^2} = 116,680 \text{ N} \quad (26,220 \text{ lb})$$

$$MS = \frac{116,680}{83,058} - 1 = +0.40$$

The tube weight per unit length is 0.0248 kg/cm (0.353 lb/in). The weight of each set to truss members is:

Tube Length	Weight
1.57 m (62 in.)	9.93 kg (21.9 lb)
1.27 m (50 in.)	8.03 kg (17.7 lb)
0.94 m (37 in.)	5.94 kg (13.1 lb)
	23.90 kg (52.7 lb/set)

Total Weight, Two sets = 47.8 kg (105.4 lb).

The weight of the ring added to the magnesium vehicle structure was assumed to be the same as the other rings in the vehicle with a weight of 17.1 kg (37.6 lb). The longerons were sized to carry the load of a single truss member. This gave a web of 0.127 cm (0.050 in.) thickness and a 3.8 x 3.8 x 0.8 cm (1.5 x 1.5 x .32 in.) tee for the inner cap. The integral stiffeners of the vehicle wall were used for the outer cap. This gave a weight of 1.17 kg (2.58 lb) for each longeron at the aft end for a total of 9.3 kg (20.5 lb) for the eight members. The longerons at the forward attachment point had a total weight of 7.8 kg (17.3 lb).

Total weight breakdown for the parachute, cables, and all structural modifications are given in Table 8.

5.2 CREW SIZE OPTIONS

Conceptual designs for crew modules have been studied for 6, 12, 18 and 24 man crews. In general, two stations have been considered for the vehicle

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Table 8

INCREMENTAL WEIGHT ESTIMATE FOR GROUND
RECOVERY AMRS

Item	Weight	
	(kg)	(lb)
Parachute	91	(200)
Cable	49	(108)
Storage and Deployment Mechanism	45	(100)
Truss	48	(105.4)
Ring	17	(37.6)
Aft Longerons	9	(20.5)
Forward Longerons	8	(17.3)
	267	(588.8)
10% Contingency	27	(58.9)
Total Weight	294	(647.7)

commander and copilot. These stations have been allowed a larger free space volume, the remaining stations have been allowed regular airline seating volumes. The life support systems were sized for a five-day vehicle occupancy with 100% reserves. One space suit per module was included for emergency use. The weights and volumes of the life support system and related hardware were taken from Ref. 9. The volumes, lengths and weights of each module are given in Table 9. A seating arrangement for each module is given in Fig. 30.

5.2.1 The Six-Man Module

The size of the six-man module is approximately the same as the four-man, 30-day module of Ref. 3. Furthermore, its weight is comparable and hence compatible with a single-stage, one Shuttle launch, cryogenic AMOOS. The crew seating arrangement was selected to yield as even a distribution of of weight as possible. The arrangement of the seats is not considered critical

4.47 m
(14.67')

4.05 m
(13.3')

Alternate Seating with Exercise Area

3.23 m
(10.6')

6-Man

4.33 m
(14.2')

12-Man

5.43 m
(17.8')

18-Man

6.55 m
(21.5')

24-Man
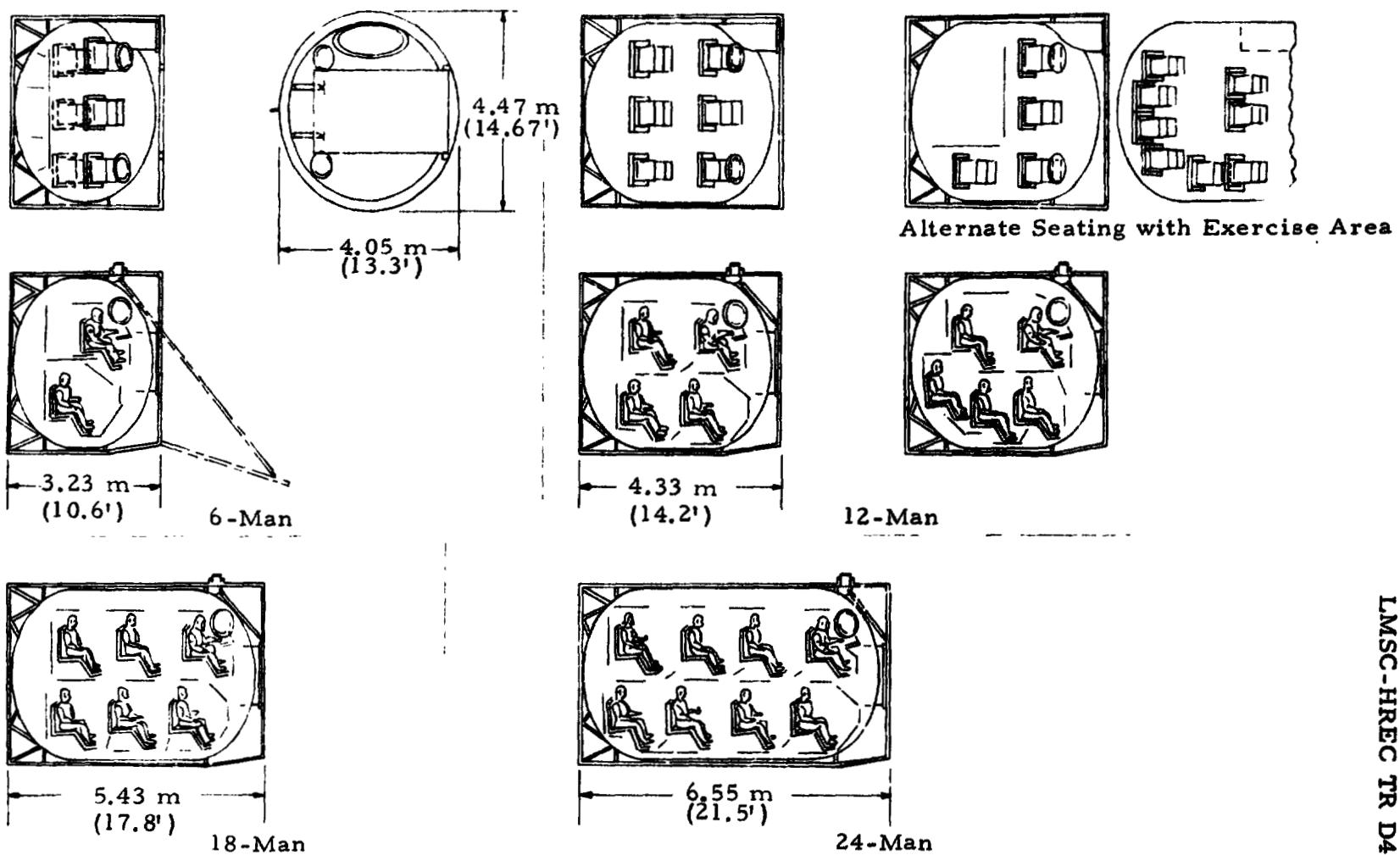
Fig. 30 - Manned Capsule Options

Table 9

MANNED MODULE VOLUMES, LENGTHS AND WEIGHTS

Crew Size	Volume Required m^3 (ft^3)	Length* m (ft)	Mass kg (lb)
6	16.74 (591)	3.23 (10.6)	3092 (6818)
12	29.25 (1033)	4.33 (14.2)	4656 (10,265)
18	41.80 (1476)	5.43 (17.8)	6221 (13,716)
24	54.40 (1921)	6.55 (21.5)	7792 (17,179)

*The cross sections are approximately 3.66 m (12 ft) by 4 m (13 ft) ellipes for the inner pressure vessels. The outer primary structures are eliptical cone frustums that mate to the AMOOS propulsion module.

so that some rearrangement could be made to yield an exercise area. The absolute need for an exercise area has not been established but its desirability is beyond doubt. The occupancy of the module will probably be twelve hours or more to a geostationary orbit on allowing for phasing, rendezvous and docking. This takes into consideration one revolution of phasing in low earth orbit (1.5 hours), 5 hours approximately, transfer to geostationary altitude and 5.5 hours (about 3 deg) of phasing, rendezvous and docking in geostationary. In consideration of volumes allowed, exercise areas etc., allowance must be made for the fact that the use is to transport crews to and from a space industrialization pilot plant. Furthermore, the crew members may not be so highly trained as the astronauts have been to date since their prime activity may be in their earthly skills in, say, construction, plant maintenance and operation, etc.

5.2.2   The 12-Man Module

The length of this module, 4.33 m (14.2 ft), allows it to be carried comfortably in the baseline Shuttle cargo bay. However, its weight of 4656 kg

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(10,265 lb) places it beyond the round trip capability of the single-stage, one Shuttle launch AMOOS. A Growth Shuttle with a payload capability of approximately 40,000 kg (88,000 lb) is required. The AMOOS propulsion module tankage would need to be increased appropriately resulting in a vehicle some 18 m (56 ft) long and so filling the entirety of the Shuttle cargo bay. The 12-man module as envisioned herein represents the limits of the payload growth potential of the Shuttle without modification to the orbiter.

In the case of the 12-man module, an alternative layout to evenly spaced seats is given. The seats are so arranged to yield an open exercise area sufficient for two crew members to exercise simultaneously. This will allow each crew member to exercise for some ten minutes of each hour.

5.2.3 18-Man and 24-Man Modules

These modules are stretched versions of the preceding modules. Either may be carried to low earth orbit in the baseline Shuttle cargo bay with a standard AMOOS propulsion module. In the case of the 18-man module a Growth Shuttle with a stretched payload bay capability of 45,360 kg (100,000 lb) and a growth AMOOS may be used for round trip geostationary missions. The 24-man module is well within the capability of a two baseline Shuttle launch, two stage AMOOS. A standard AMOOS and the 24-man module will fit in the baseline Shuttle cargo bay. However, AMOOS propellant must be off-loaded.

5.3 DUAL FUELED AMOOS (HYBRID ENGINE AMOOS)

The two viable alternatives for a dual fueled AMOOS engine appear to be $LOX-LH_2/LOX-RP-1$ and $LOX-LH_2/LOX-Methane$. These engines have been analyzed for NASA-MSFC in Ref. 10. The optimum engine thrust for AMOOS probably lies within the range from 44,500 to 89,000 N (10,000 to 20,000 lb) vacuum thrust. In this range the decrease in gravity losses are approximately equal to the increase in start and stop losses. The basic trad is therefore an engine weight increase versus propellant saved with

a higher I_{sp} within the above thrust range. A decrease in I_{sp} from 470 to
456.5 sec requires a propellant increase of 280 kg (620 lb), or approximately
21 kg (46 lb) of propellant per second of I_{sp}. Applying this to the engine data
of Ref. 10. shows that an approximate increase of 25 sec in I_{sp} from the base-
line expansion ratio of 40.1 to 200:1 results in a saving of 522 kg (1150 lb) of
propellant. This is obtained by a nozzle extension which weighs some 9 to
23 kg (20 to 50 lb) depending on engine thrust. A further increase in expan-
sion ratio of 400:1 increases the I_{sp} by a further 7 sec saving approximately
136 kg (300 lb) of propellant for a similar, but further, increase in nozzle
extension weight. Increasing the thrust from approximately 31,150 N (7000 lb)
vacuum to 129,000 N (29,000 lb) vacuum increases the I_{sp} by less than 1 sec. The
engine weight, however, increases by approximately 81.7 kg (180 lb), or a net
penalty of over 59 kg (130 lb). From these considerations and the engine data
(Refs. 10 and 11) the optimum vacuum thrust for AMOOS applications appears
to be between 44,500 N (10,000 lb) and 66,750 N (15,000 lb). The 66,750 N
(15,000 lb) vacuum thrust engine will be used for the dual mode engine AMOOS
design. The 200:1 expansion ratio nozzle data will be used since the gain by
going to 400:1 appears small for the size of nozzle involved and the small
weight advantage gained. The approiate data from Ref. 10 is reproduced herein
as Table 10.

5.3.1 Dual Fueled AMOOS Weights Analysis and Conceptual Design

The total weight savings for the AMOOS propulsion unit from the use
of a dual fueled engine has been estimated. The mixture ratios used were
6:1 for LOX/LH_2 and 3.81/1 for LOX/CH_4.

The total weights for the LOX/LH_2 and LOX/CH_4 are (17,366 kg) 38,285
lb and 4633 kg (10,215 lb), respectively. These values result in required vol-
umes of 17.58 m^3 (621 ft^3) of LOX, 2.41 m^3 (85 ft^3) of CH_4 and 37.94 m^3 (1340
ft^3) of LH_2. Tanks were sized for these volumes with the methane tank located
inside the LOX tank (Fig. 31). Based on these volumes the changes in tank size
and vehicle length of the original AMOOS propulsion unit was determined. The
engine swap resulted in zero net weight change. The dual fuel configuration

Table 10

DUAL MODE ENGINE DATA (FROM REF. 10)

| Expansion Ratio	Vacuum Thrust (lb)	Vac. I_{sp} (sec)	Engine Weight (lb)	Length Noz. Ext. (in.)	Diam. (in.)	Oxidizer	Fuel
60:1	15,000	439.18	369.97	38.90	15.56	LOX	LH_2
200:1	15,841	463.81	408.23	67.60	34.78	LOX	LH_2
400:1		471.04	456.94	89.11	49.18	LOX	LH_2
60:1	9,954	341.95	367.46	38.90	15.56	LOX	RP1
200:1	10,730	368.58	405.72	67.60	34.78	LOX	RP1
400:1	10,978	377.12	456.94	89.11	49.18	LOX	RP1
60:1	9,961	351.15	369.97	38.90	15.56	LOX	CH_4
200:1	10,743	379.24	408.23	67.60	34.78	LOX	CH_4
400:1	10,995	388.15	456.94	89.11	49.18	LOX	CH_4
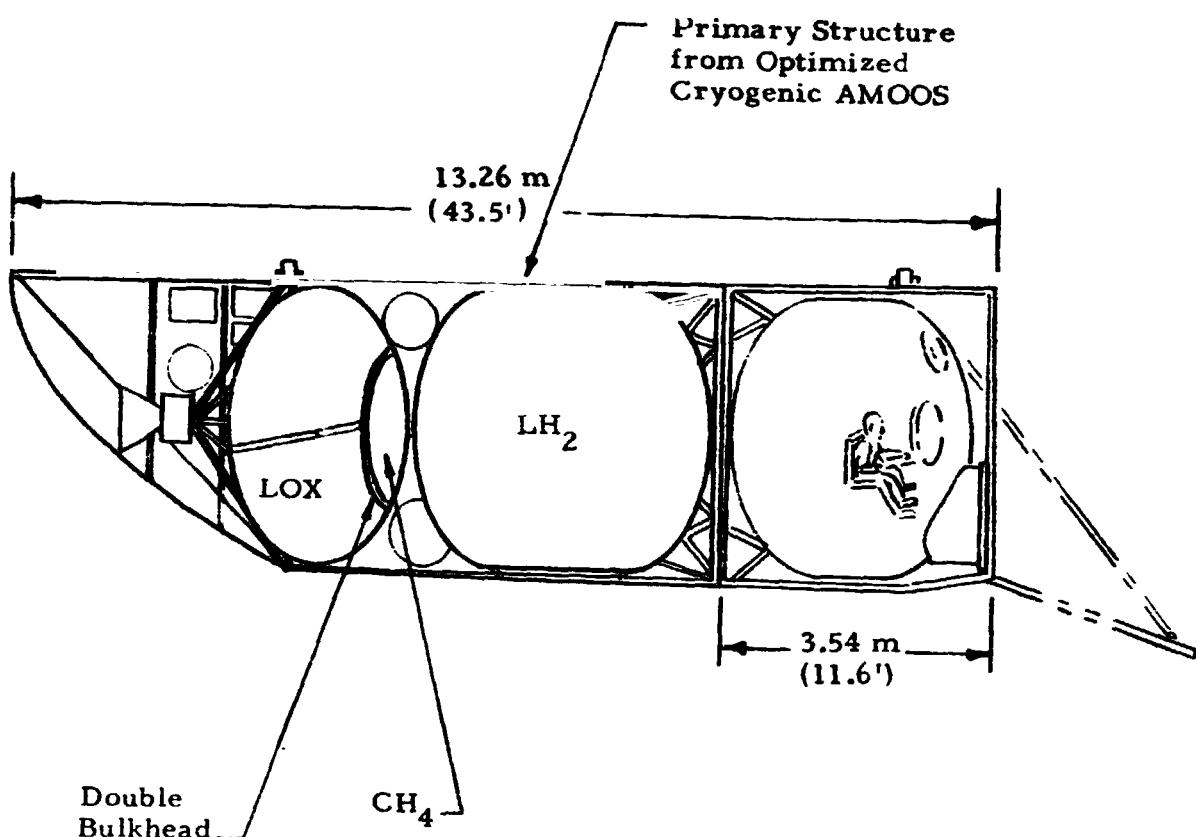
Fig. 31 - Dual Fueled AMOOS Conceptual Design

propulsion unit is 9.72 m (31.9 ft) in length, or some 0.64 m (2.1 ft) shorter than the cryogenic propulsion module and 51.3 kg (113.1 lb) lighter. The weight changes are summarized in Table 11. The 10% contingency was included to account for the contingency that was added to the original configuration.

Table 11
DUAL FUELED AMOOS DRY WEIGHT CHANGE

	kg	(lb)
Structure	-13.6	(-30.0)
Engine	0	0
LH_2 Tank	-23.9	(-52.8)
LOX/CH_4 Tank	+20.7	(+45.6)
TPS	-29.0	(-64.0)
10% Contingency	- 4.6	(-10.1)
	-51.3	(-113.1)

5.3.2 Dual Fueled AMOOS Performance

The payload performance of the dual fueled AMOOS was computed for a geostationary mission. A dry weight of 3039 kg (6700 lb) was used. Since the weights savings is 51.3 kg (113.1 lb), the payload estimates using this dry weight was considered sufficiently accurate for these purposes. An I_{sp} of 463 sec was used for the $LOX-LH_2$ burns and an I_{sp} of 377 sec was used for the LOX-methane mode. These values are slightly smaller than those given in Table 10. The performance is given in Fig. 32. The performance of the dual fueled all-propulsive vehicle was also computed. The results are also given in Fig. 32. For comparative purposes, the all cryogenic modes are given for both AMOOS and the baseline cryogenic tug.

In the performance analysis it was assumed that the tank sizes were such that a certain amount of trading between LH_2 and methane could be made.
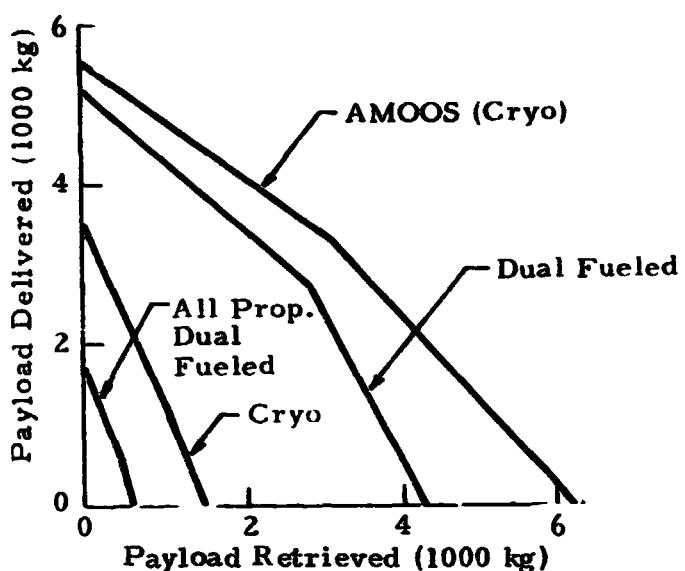
Fig. 32 - Payload Performance of Single-Stage OTV Using a Dual-Fueled
Engine to Obtain a Long On-Orbit Lifetime

Note: The dual fueled engine is used in these studies to obtain a long
on-orbit lifetime. In this application the cryogenic propellant is
burned to achieve mission orbit and the high density, space storable
to return to low earth orbit. This is in the reverse order to maxi-
mize performance. The performance of the long on-orbit lifetime
vehicle may possibly be enhanced by the following mode of opera-
tion. In this mode, an initial high density fuel burn is followed by
a cryogenic burn to achieve mission altitude. After the mission is
complete, the high density fuel is used to return. Such a mode of
operation was not incorporated into these studies but should be
performed before its effect can be assigned a quantitative value.

If the methane tank is designed for the round trip geostationary mission, then the round trip payload is the maximum payload that can be returned. This is so since the LH_2 can neither be stored for use on the return transfer orbit insertion burn nor traded for methane.

The results show a small payload penalty for AMOOS. This relatively small penalty is due to the relatively small Δv required from the LOX-methane mode. On the other hand, half the total Δv is required of the LOX-methane mode for the all-propulsive tug. This results in a much larger percentage payload decrease for the all-propulsive vehicle than for AMOOS.

The round trip payload decrease is approximately 450 kg (1000 lb), for the dual fueled AMOOS to approximately 2770 kg (6100 lb), which is insufficient for the four-man, 30-day crew module. The crew module was redesigned to a four-man, five-day module and reduced in volume. This new volume is consistent with airline standards for passengers. The resulting module weight was reduced by approximately 900 kg (2000 lb). This is considered a minimum capsule for use in the late 1980s through the 1990s. The resulting weight of 2184 kg (4814 lb) as shown in Table 12 is well below the round trip payload capability of 2770 kg (6100 lb). This difference in module weight and payload capability may be used in many ways, e.g., as an unassigned contingency to increase confidence in the design, to increase the life support capability of the capsule or, possibly, increase the crew size.

In conclusion then, the dual fueled AMOOS can, using one Shuttle launch, perform the crew rotation mission to geostationary orbit and remain on station for some three to six months. During this time it is available for crew return at any time. Converting the all propulsive OTV to dual fueled operation virtually eliminates its payload capability to geostationary, it being only one-sixth that of the corresponding AMOOS. It should be noted, however, that the dual-fueled mode is used to extend on-orbit lifetime, so that the cryogenic propellant is burned first.

Table 12

DUAL FUELED AMOOS MANNED MODULE
WEIGHTS BREAKDOWN

Item	Weight[*]	
	kg	(lb)
Shell Structure	115	(253)
TPS	118	(260)
Flap	102	(225)
Docking Mechanism	54	(120)
Capsule (Inner Pressure Vessel)	532	(1173)
Crew, Life Support System, etc.	1263	(2783)
Total	2184	(4814)

[*]Includes 10% contingency.

5.4 RESCUE – ABORT MODES

Currently, both rendezvous with the Shuttle and ground recovery modes
have been considered for AMRS. However, the mode to be used was assumed
to be known prior to deployment so that during the entire AMRS mission the
operation was toward one recovery method or the other. If both systems
were incorporated into the vehicle, namely, that it always carried a recovery
parachute and the propellant required to rendezvous, the decision of the mode
to be used could be delayed at least to the midcourse correction and possibly
until after atmospheric entry. The location of the decision point was investi-
gated using the AMOOS three-dimensional guidance and trajectory computer
program.

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The AMRS vehicle was targeted to an AMOOS skip type trajectory. The target perigee for such a trajectory is only some 2 km (1 n.mi.) higher than the AMRS reentry-type trajectory target perigee. The AMOOS-type trajectory was chosen since the pull into the atmosphere onto a reentry trajectory was considered to have certain advantages. These were: (1) if the skip maneuver were selected, then a better phasing orbit would be achieved; (2) if the transfer were initiated too late, then a safe orbit could be achieved; and (3) it probably delayed the decision point as long as possible after entry because the velocity loss is less rapid.

To determine the location of the decision point the nominal AMOOS type trajectory, with 90 deg bank angle, was flown to some time, say t_1. At this time a bank angle change was started to increase the bank angle to 180 deg. This would point the lift vector vertically downward and so yield the maximum trajectory control in changing from the skip maneuver to the reentry trajectory. The bank angle was changed at an average rate of 4.5 deg/sec which corresponds approximately to an average 1 deg/sec^2 bank angle acceleration capability. The results are shown in Fig. 33 as a plot of apogee altitude versus time of recovery mode transfer maneuver initiation.

An absolute maximum to the time of initiation is about 200 sec after atmospheric entry. However, a more practical maximum is some 170 or 180 sec after entry. These times are 40 to 50 sec, respectively, after perigee passage so that the flight path angle, γ, is positive, hence the quite well defined apogee altitude. The sharp turn up in apogee altitude is due to the AMRS flying into relatively low density air before γ can be reduced to zero. The lift forces at altitudes of 90 km and above are small compared to the vehicle weight.

The weight penalty for the dual recovery mode over the ground recovery mode is approximately 90 kg (200 lb) in recovery weight when liquid propulsion is used. This yields a penalty of approximately 270 kg (600 lb) penalty over the Shuttle recovery. Since the ratio of on station weight to recovered weight

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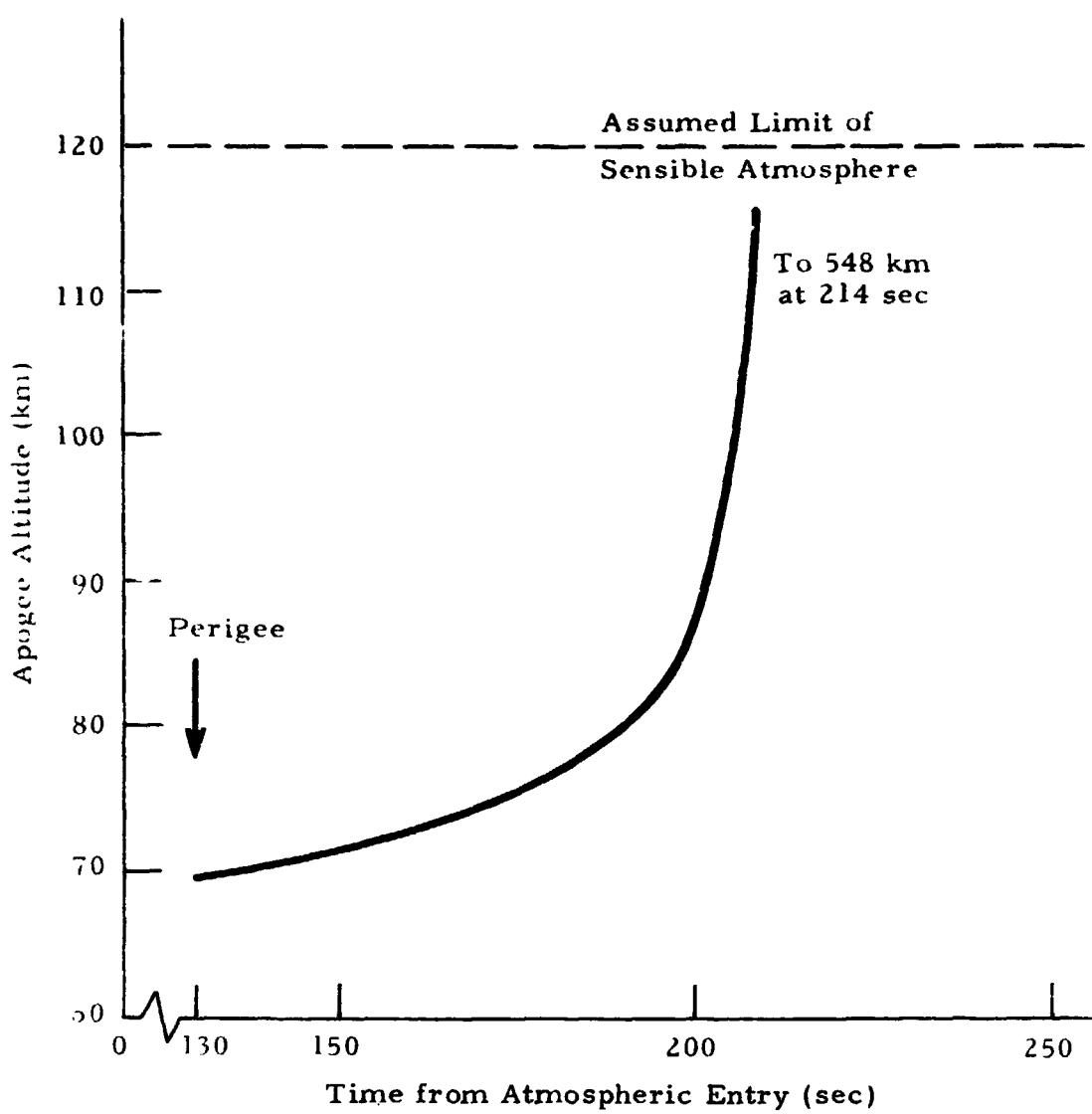
Fig. 33 - Apogee Altitude as a Function of Time from Atmospheric
Entry at Which Recovery Mode Changeover is Made. (Skip
mode to ground recovery mode).

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is approximately 2:1, the all-up weight penalties are approximately twice these, namely 180 kg (400 lb) and 540 kg (1200 lb), respectively. Using a solid rocket motor may reduce these penalties somewhat since not only is the all-up weight less but the recovered weight is considerably less. The solid motor AMRS is discussed in Section 5.5. The resulting parachute system and, hence, the basic penalty would be reduced. These reductions could be as much as 180 kg (400 lb) of the on-station weight. As mentioned in Section 5.1, further reductions may be possible after a detailed analysis of loads and load paths in the primary structure to the parachute riging attachment points. The current best estimate of the dual recovery mode AMRS, without crew, of approximately 5450 kg (12,000 lb) is within the delivery capability of the 470 sec I_{sp} cryogenic AMOOS.

5.5 SOLID KICK STAGE FOR AMRS

Under certain circumstances, solid rocket motors can be competitive with liquid motors. Since AMRS requires a relatively small motor, low thrust and low total impulse, and a space storable propellant, a solid motor may be used with the potential for a low or even negative weight penalty. In Ref.3, the main engine consumables are given versus recovered weight. Recovered weight includes the weight of the engine and tanks and must, therefore, include the weight of the solid motor casing whether or not it is recovered. The recovered weight of AMRS is approximately 2730 kg (6000 lb).[*] The on station weight, including a crew of four, is 5670 kg (12,500 lb) for an I_{sp} of 300 sec. These data are taken from Fig. A-7 of Ref. 3. The corresponding main engine consumables are approximately 2880 kg (6350 lb) from part (b) of the above figure. The engine parameters may now be obtained from the general design data presented in Ref. 12. From Fig. 1 of Ref. 12, a propellant

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[*] Sometimes the upper bound to the recovery weight is taken in order to obtain an upper bound to some design parameter. In this case, the lower bound is taken in order to obtain an upper bound on the accelerations experienced during the burn.

total weight of 3290 kg (7250 lb) is required for a solid propellant weight of 2880 kg (6350 lb). The length and diameter of such a stage are approximately 2.59 m (102 in.) and 2.03 m (80 in.), respectively. The average thrust for such a motor is approximately 98,000 N (22,000 lb). The weight at burnout will be somewhat higher than the recovery weight of 2720 kg (6000 lb). Hence the maximum acceleration will be approximately 3.7 g. Provided the thrust time-history of this motor follows a pattern typical for solid rocket motors, the maximum acceleration should be below 4 g even for a 3σ high thrust motor.

The preceeding discussion assumes an optimum motor design expressly for the AMRS. The second approach is to use an existing solid rocket motor or, if necessary, two or more of an existing motor. It is considered impractical to use more than three motors. In the case of several motors, they could be staged or burn concurrently. Multiple engines raise the problem of potentially large torques should an engine failure occur.

Reference 12 can be used to obtain general design data. Both optimum and worst-case data can be generated. The worst-case data are an upper bound to the engine weight and represent the case where the switch must be made to n + 1 motors from n motors. This assumes the use of an existing motor which is non-optimum for the application. The data presented here are for an I_{sp} = 300 sec which represents an achievable maximum with predictable advances in technology.

The homing to a design is an iterative process since the weights are a function of the rocket motor propellant, consumable inerts (insulation, etc.) and rocket motor case weights. To start the iterative process the propellant weight of 3000 kg (6600 lb) is used. This gives 1500 and 1000 kg (3300 and 2200 lb) of propellant per motor for the two-engine and three-engine case, respectively. The plot of kickstage total weight versus propellant weight in Fig. 1 of Ref. 12 is again used to obtain total weights of 1769 and 1211 kg (3900 and 2670 lb), respectively. This yields engine cluster weights of 3538 and 3633 kg (7800 and 8010 lb), respectively. Comparing these weights to

the 3290 kg (7250 lb) of the single motor yields excess burnout weights of 136 and 231 kg (300 and 510 lb), respectively. The on-station weights are, therefore 5920 and 6015 kg (13,050 and 13,260 lb), respectively, and recovered weights of 2860 and 2950 kg (6300 and 6510 lb), respectively. The calculations are repeated starting with Fig. A-7 of Ref. 3. On-station weights cf 5920 and 6015 kg (13,050 and 13,260 lb) can recover approximately 2860 and 2925 kg (6300 and 6450 lb), respectively.

The two-motor design has converged, but the three-motor design has not. The propellant for the three motor design is now raised to 3060 kg (6750 lb). The motor cluster weight increases to 3715 kg (8190 lb), the on-station weight to 6095 kg (13,440 lb) and a recovery weight of 2965 kg (6540 lb). These weights are consistent with the plots of Fig. A-7 of Ref. 3 to the degree of accuracy expressable here and so the three-motor design has converged.

The above calculations assume that the motors burn concurrently, but not necessary with multiple motors. However, if the cases are to be recovered, then there is no difference, at the current level of accuracy, in concurrent or consecutive burns. However, if staging is allowed, then discarding the spent motors allows an increase in recovered payload performance. The above weights represent the best possible two-motor and three-motor cluster if the cases are recovered since each motor is optimized. Non-optimum motors cause a small weight penalty, perhaps of the order of 100 to 150 kg (200 to 300 lb) to the on-station weight (the difference between two and three and one and two motors, respectively). However, the increased weight of both the two-motor and the three-motor cluster, concurrent burning, takes the on-station weight of AMRS beyond the current delivery capability of AMOOS to geostationary orbit.

The effects of staging can be estimated as follows. The case weights of the two motor design is approximately 270 kg (600 lb) each. During the burn of the second motor, the weight is reduced by 270 kg (600 lb). This represents some 6% of the weight of the vehicle at second motor ignition and 10% of the weight at second motor burnout. The second motor propellant requirement

can, therefore be reduced by some 8% to, say, 1380 kg (3040 lb). A smaller reduction may be made to the total motor propellant, to allow for the reduction in second motor weight, in all, about 2.5% of the propellant savings, or about 30 kg (66 lb) of propellant. The procedure should be repeated until it converges. However, this first iteration (150 kg (330 lb)) is sufficiently close for current purposes. The total weight savings are between 150 and 200 kg (330 and 440 lb) or just about the savings of a single motor over two motors. At best 50 kg (110 lb) is saved which must cover the weight increase for complications in thrust structure, motor ignition system, etc. Since the potential trade is so small, staging solid motors for AMRS appears to be impractical.

5.5.1 Weights Estimate and Conceptual Design Solid Motor AMRS

A truss structure was designed to attach the solid rocket motor to the AMRS manned module. The truss members are round magnesium tubes 5.1 (2.0) o.d. x 0.64 cm (.250 in.) wall thickness. A computer analysis was performed of the truss structure to determine the maximum load in any member. From the computer analysis the maximum truss member axial load is

$$P_{max} = 63,266 \text{ N } (14,218 \text{ lb})$$

The critical column load for the truss member is

$$P_{cr_{col}} = \pi^2 \text{ EI}/\ell^2 = 361,260 \text{ N } (81,181 \text{ lb})$$

Critical load based on material allowable is

$$P_y = F_{cy} \cdot A = 128,400 \text{ N } (29,854 \text{ lb})$$

$$\therefore MS = \frac{P_y}{P_{max}} -1 = +1.029$$

Based on this size the total weight for the truss members is 17.0 kg (37.4 lb).

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A 2 x 2 angle ring with a wall thickness of 0.64 cm (.250 in.) was used for the truss members attachment. One half the ring weight was assumed for the attachment point hardware and fittings. The respective weights of the required structure are listed in Table 13.

Table 13

ESTIMATE OF STRUCTURAL WEIGHT OF SOLID
MOTOR AMRS THRUST STRUCTURE

Item	Weight	
	kg	(lb)
Truss	17.0	(37.4)
Ring	8.3	(18.4)
Attach Points	4.2	(9.2)
+10% Contingency	2.9	(6.5)
	32.4	(71.5)

The resulting conceptual design is given in Fig. 34.

2.50 m
(8.2')

2.04 m (6.7')

7.07 m
(23.2')

Optimum Solid
Motor Dimensions

Computer Analyzed
Truss Design

Optimized Primary
Structure from Liquid
Motor AMRS

2.53 m
(8.3')

2.26 m (7.4')
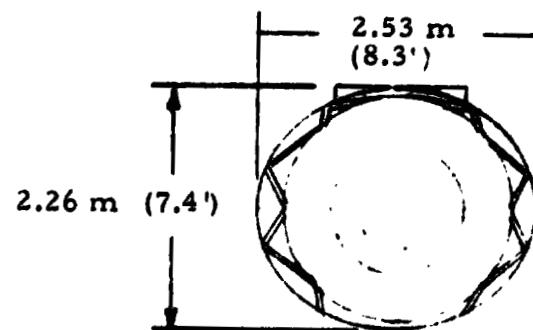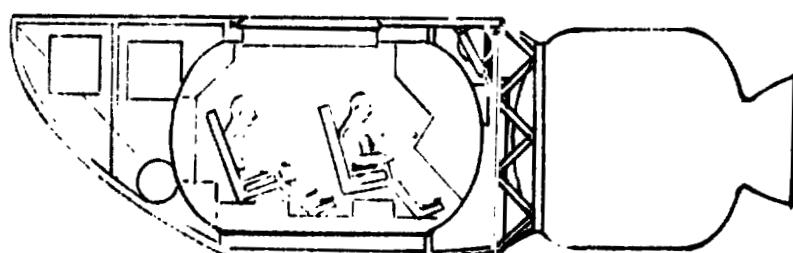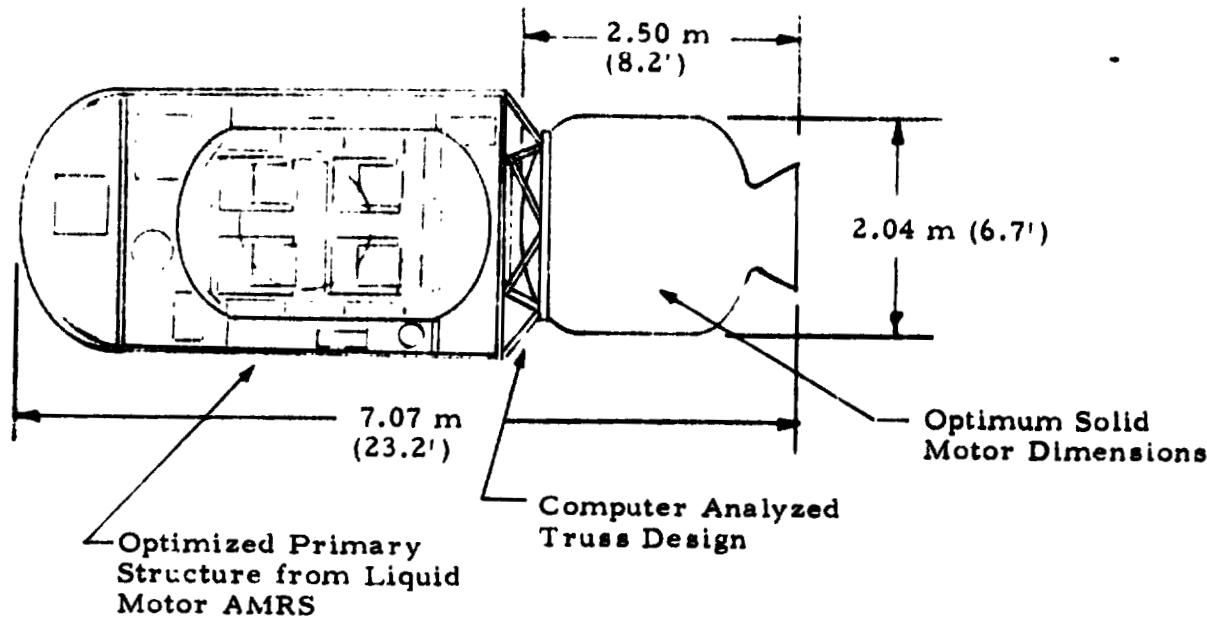
Fig. 34 - Conceptual Design for Solid Rocket Motor AMRS

Section 6
ADVANCED MISSION APPLICATIONS

Advanced missions for the Space Shuttle and hence the Orbit Transfer Vehicle (OTV) will be in support of Space Industrialization. Space Industrialization and Space Station Studies have been initiated by NASA in an effort to define the next logical space step. Eventually, both men and materials will be required in geostationary orbit since a spacecraft in geostationary orbit is the most satisfactory approach to providing a continuous service to a given area. Furthermore, a large area can be served by such a satellite. Whether or not activity in geostationary orbit is the next step, it is undoubtedly in the sequence of steps leading to Space Industrialization and possible colonization. Earlier steps may well be in low earth orbit, but at a sufficiently high orbit to give an on-orbit lifetime of several years. The OTV development must be toward a vehicle with good low earth orbit performance with the potential for development to perform geostationary missions. Furthermore, the vehicle must be suitable for performing the crew rotation mission which requires a relatively rapid transport to and from mission orbit.

Studies to date have shown that AMOOS fulfills these requirements to a greater degree than any other OTV candidate. However, crew sizes and payloads are expected to exceed the current baseline Shuttle's capability and require growth versions and Shuttle derived heavy lift launch vehicles (HLLVs). To match the growth of the launch vehicle, versions of the AMOOS have been studied. Concepts are suggested for use with both the Growth Shuttle and the Shuttle derived HLLVs. Two versions of the Growth Shuttle and two HLLV developments have been considered. For the purpose of these studies, each is defined by its payload capability to low earth orbit. Growth Shuttles are identified as the 36,300 kg (80,000 lb) and 45,400 kg (100,000 lb) payload versions and the HLLVs by 58,970 kg (130,000 lb) and 72,575 kg (160,000 lb) payloads. These payloads are to a 295 km (160 n.mi.) circular orbit with a 28.5 deg inclination. The launch would be from the Eastern Test Range.

Payload performances were computed for geostationary missions for both aeromaneuvering and all-propulsive cryogenic OTVs. Single stage OTVs were analyzed for the Growth Shuttle delivered vehicles, and two stage OTVs for the 58,970 kg (130,000 lb) and 72,575 kg (160,000 lb) HLLVs. The payloads were computed for an I_{sp} of 463 sec only. Payloads for other I_{sp} values may be estimated from Section 2 and Refs. 1 through 3. The results of the payload performance is given in Fig. 35 for missions to geostationary altitude. The aeromaneuvering vehicles outperform the all-propulsive for all missions which include the recovery of the OTV. The all-propulsive vehicles are more sensitive to number of stages than the aeromaneuvering vehicles. This is probably due to the larger total Δv value required for the mission.

The performance of AMOOS was also computed for high energy missions. These missions consist of a boost to a Δv greater than 3000 m/sec. The payload is then released and AMOOS decelerated to 3000 m/sec to ensure that it does not escape the earth's gravitational field. A small deceleration at apogee targets AMOOS to a perigee within the earth's atmosphere for recovery into low earth orbit. Only single stage aeromaneuvering OTV was analyzed. In all five vehicl s were considered, one each for the three Shuttle configurations and the two HLLVs. The results are given in Fig. 36 as a function of the payload Δv.

6.1 ADVANCED MISSION VEHICLE DESIGN

Conceptual esigns were developed and preliminary weights estimates were made for AMOOS configurations for advanced mission applications. Due consideration was given to transportation in the Shuttle cargo bay. The AMOOS vehicles for the 80K and 100K Growth Shuttles must, of course, fit in the Shuttle bay. The recommended vehicles for the 130K and 160K HLLVs are two-stage vehicles so that the AMOOS vehicles for the 65K and 80K Shuttles may be used. A conceptual design for the 80K AMOOS vehicle is shown in Fig. 37. The 100K AMOOS vehicle would be a stretched version of the 80K vehicle. The lengths of the vehicles are 12.25 m (40 ft) and 14.3 m (47 ft) for the 80K and 100K Shuttles, respectively. The dry weight of the 80K vehicle

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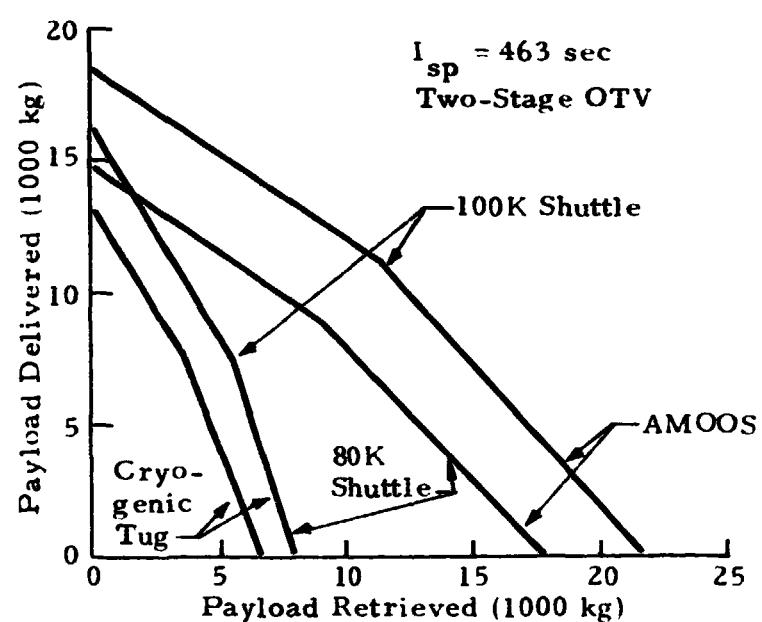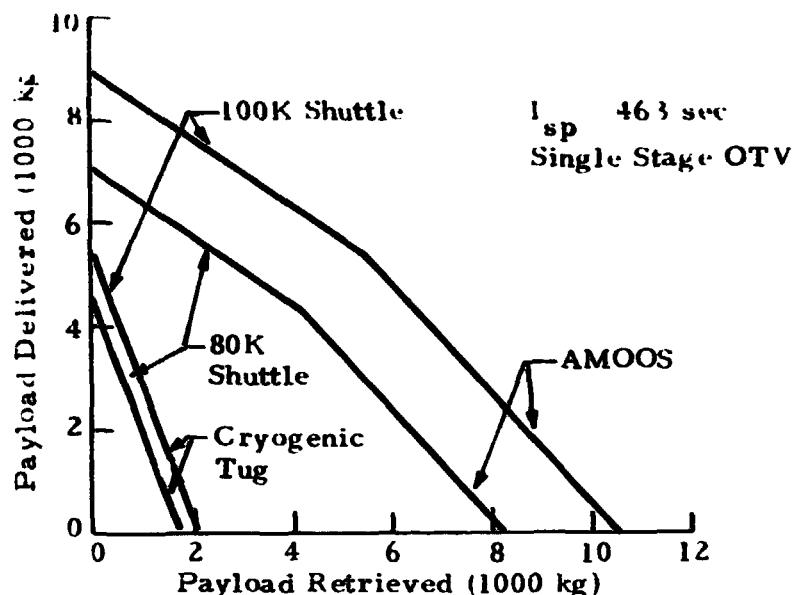
Fig. 35 - OTV Payloads to Geostationary Orbit Using the Growth Shuttles or Shuttle Derived HLLVs for Delivery to Low Earth Orbit
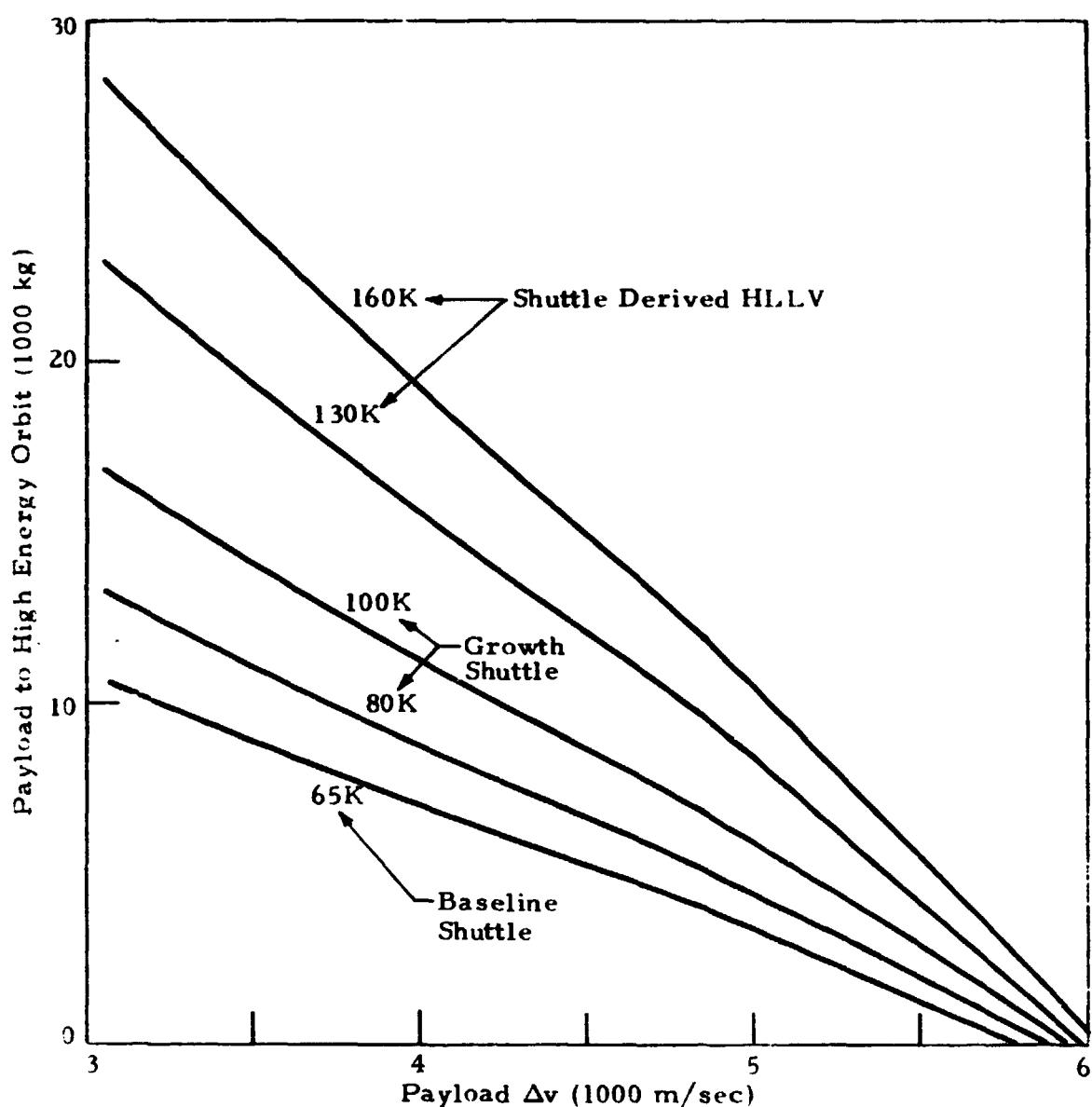
Fig. 36 - AMOOS Payloads to High Energy Missions. Recoverable Single Stage AMOOS, I_{sp} = 463 sec

99
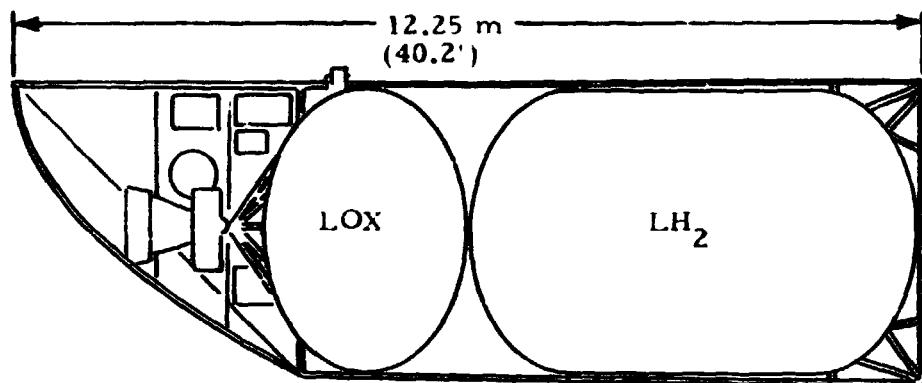
Fig. 37 - AMOOS Configuration for the 80K Growth Shuttle

was estimated from the dry weight of the AMOOS for the 65K Shuttle. The resulting dry weight was 3267 kg (7200 lb). Because of the method of obtaining this estimate, a dry weight of 3629 kg (8000 lb) was used in the performance calculations. Dry weights of 4536 kg (10,000 lb), 5897 kg (13,000 lb) and 7257 kg (16,000 lb) were used for the single stage AMOOS vehicles for the 100K Growth Shuttle and the 130K and 160K Shuttle derived HLLVs, respectively. Dry weights of 3039 kg (6700 lb) and 3629 kg (8000 lb) were used for each stage of the two-stage AMOOS vehicles for the HLLVs.

6.2 ADVANCED MANNED VEHICLE

The AMOOS applications have assumed that the Shuttle is available, at least, for transporting the manned AMOOS crews to and from low earth orbit. So far, only the AMRS has been considered in a ground recovery mode and this only as an option since it is an emergency vehicle. Because of its emergency nature, AMRS is not necessarily considered reusable after a ground recovery so that, in its design, no effort was made to protect the primary structure upon impact. Furthermore, the basic AMOOS and AMRS external geometry is not suited to a ground recovery technique that allows reuse. The reason for this is that the resulting subsonic aerodynamic characteristics are unsuited to a horizontal landing.

Introducing the Shuttle derived HLLVs into the AMOOS picture offers the opportunity for a mission which consists of the delivery of a large payload, a crew rotation and retrieval of a relatively small payload. To divorce the mission completely from the Shuttle, the recovery vehicle must be capable of an earth landing. If, further, it must be reusable, then a horizontal, airplane-type landing is preferred. Furthermore, even though the vehicle may operate independently of the Shuttle, it is desirable that it can be transported in the Shuttle cargo bay. Finally, it should be capable of transporting a crew of four to and from geostationary orbit. With these considerations in mind, the conceptual design of such a vehicle was prepared and is shown in Fig. 38. The design allows seating for five, and sufficient volume for an engine and some propellant. Since no performance estimates were made, the engine and p. opellant are not shown. However, the volume and areas available are sufficient to house an RL10-IIB engine. The main propellant would be carried in a slipper tank fitting underneath and around the nose of the vehicle. This tank would be expendable. Such tankage could provide sufficient volume for a round trip geostationary mission for the vehicle alone. Furthermore, the entire vehicle and tanks could be carried to low earth orbit in the Shuttle. For use with the Shuttle derived HLLV, further tankage would be required or possibly a stage or boosters. The configuration, propellant requirements, etc., for use with the Shuttle-derived HLLV requires further consideration.

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8.84 m
(29')

4.47 m
(14.67')

2.13 m
(7')

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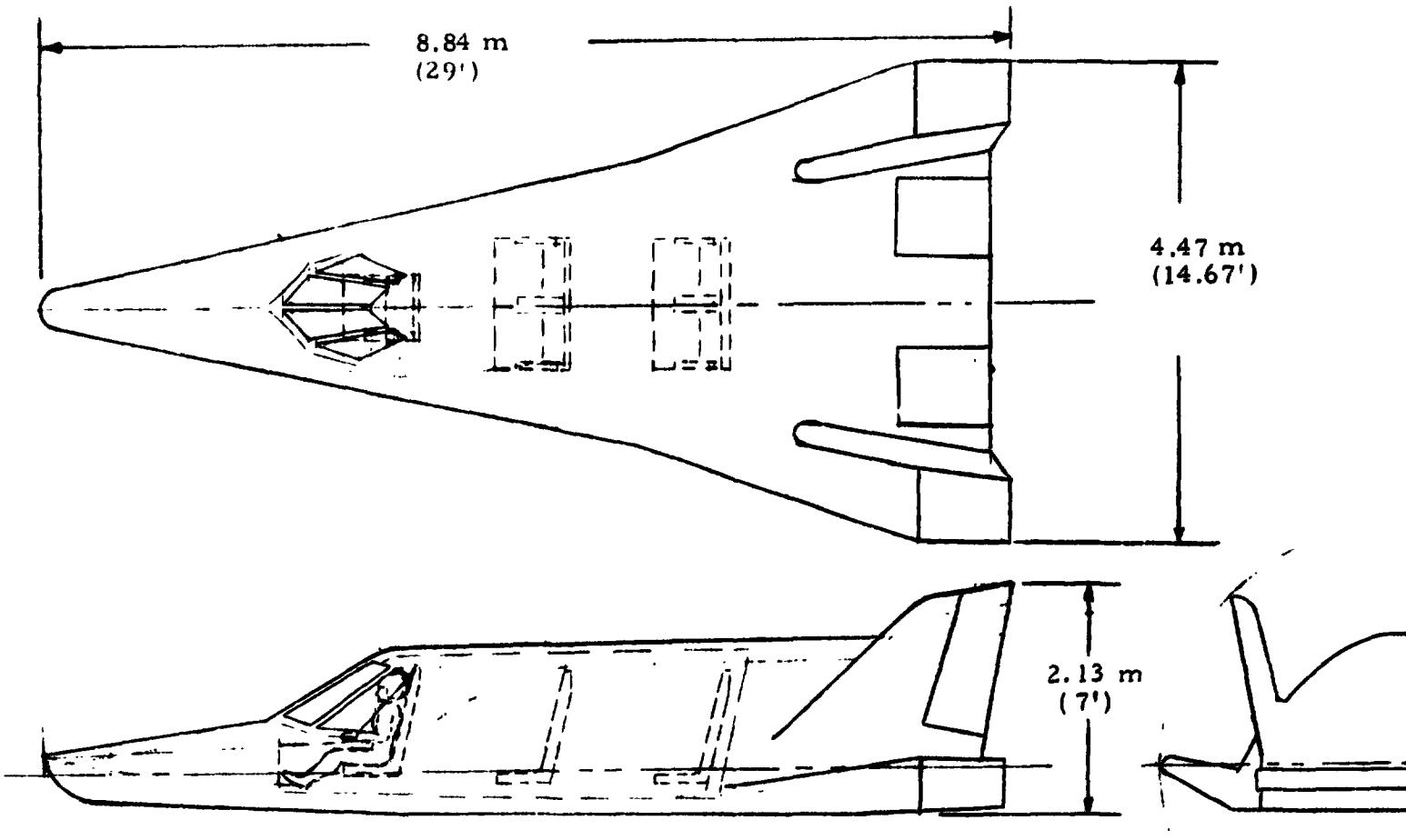
Fig. 38a - Horizontal Landing, Ground Recoverable AMOOS
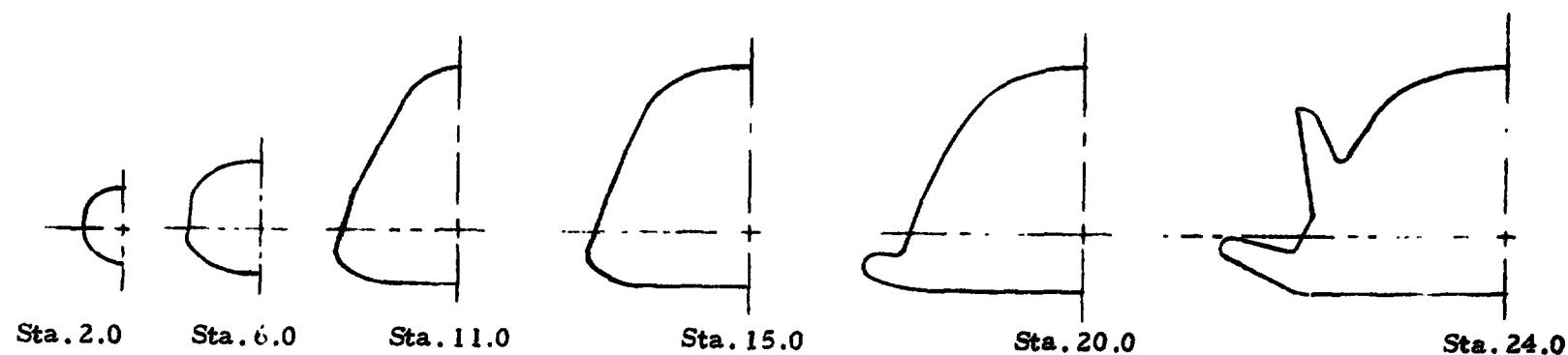
Sta. 2.0   Sta. 6.0   Sta. 11.0   Sta. 15.0   Sta. 20.0   Sta. 24.0

Fig. 38b - Cross Sections

Section 7
IMPLICATIONS OF SPACE BASED OPERATIONS

If large payloads are to be moved from low earth orbit to higher orbits up to geostationary, then the OTV may have to be space based. This may be necessary if the optimum vehicle is too large for transport to and from low earth orbit in the Shuttle cargo bay. Initially, to put an OTV in low earth orbit, it may be necessary to transport it, in some way dismantled, in the Shuttle cargo bay. After it is assembled in low earth orbit, it would be space based. There are both cost and weight trades available for space basing. Only the weight trades will be considered herein. Basically, on the one hand is the savings in not having to transport the vehicle, and on the other hand, all items for refurbishment must be transported to low earth orbit. These include items that may be carried on a scheduled or unscheduled basis. The first item that must be transported is the refurbishment base. This includes a hangar, tools and storage for the material expended during the refurbishment, storage tanks for propellant reserves and crew quarters. Certain test and vehicle checkout equipment will be required. Adequate power generating capability will have to be on orbit. For the purposes herein, these permanent structures and equipments will not be considered in the trades. The lifetime of these items is assumed sufficiently long that the weight cost of transporting them to low earth orbit will have negligible effect on the per flight weight analysis.

The remaining scheduled items cover the materials that are consumed in some way either during refurbishment or the OTV flight. These items include the life support consumables, hangar pressurization (if feasible) materials expended in refurbishment operations, refurbishment materials and OTV consumables. Also to be included in the trade with the weights of the foregoing items is the tare weight on the Shuttle from having to transport these items. This includes tanks, and other items such as vents, fill,

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drain and dump mechanisms, racks and containers. This general discussion
applies equally to an all-propulsive vehicle as to an aeromaneuvering vehicle.
The distinction is that for an aeromaneuvering vehicle the TPS must be re-
furbished.   On high energy missions, this TPS must currently be an ablator.
For low energy missions, an insulator of reradiative TPS may be used. At
this time, the insulator LI 900 is recommended since it gives the best temper-
ature range for the unit weight involved.

The baseline mission was originally specified as 550 km (300 n.mi.)
circular orbit with an inclination of 55 deg. This is considered somewhat
restrictive so that higher orbits will be considered.

Table 14 gives a weight breakdown of the utilization of the Shuttle pay-
load flying a refurbishment mission. The OTV payload is assumed to be in
orbit or may be a crew who may be carried in the orbiter without perform-
ance degradation. The weights analysis is based upon previous vehicle
weights analyses of AMOOS, the Cryogenic Tug and their impact on the
Shuttle. The tare on the Shuttle was reduced from (860 kg) 1900 lb to 410 kg
(900 lb). The weight of tanks to carry the propellant to low earth orbit was
estimated from the Cryogenic Tug tank weight. This weight is considered a
practical minimum for thin wall tank development by the mid 1980s. The
remaining weights were estimated on consideration of the structure involved
and weights to be carried.

The payload for the OTV is not included in the Shuttle payload. OTV
missions having propellant requirements of the order of the shuttle delivery
capability given in Table 14 are either to high energy orbits or correspondingly
increasing payloads to lower orbits. If the payload is to be transported with
the propellant, this will have to be at the expense of the propellant. The off-
loading can be estimated using Appendix A of Ref. 3 fo AMOOS on a geosta-
tionary mission. The all-up weight of AMOOS in low earth orbit will be
approximately 30,844 kg (68,000 lb) instead of 28,622 kg (63,100 lb). In this

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Table 14

SPACE BASED REFURBISHMENT WEIGHTS TRADE

	Weight	
Shuttle Tare and Miscellaneous Weight Costs	kg	(lb)
Tanks and Structure	450	(1000)
Racks and Containers	90	(200)
Crew Rotation (Avg. 1 man) Life Support for Maintenance Crews (Average per Flight)	140	(300)
Miscellaneous	410	(900)
Net Total	1090	(2400)
Contingency	110	(240)
Total	1200	(2640)

Shuttle Net Payload, Ablative TPS Option 1. Ablator Sprayed on in Space		
Propellant	26,762	(59,000)
Programmed Refurbishment	113	(250)
TPS: Ablator	590	(1300)
Unscheduled Replacement (Avg.)	23	(50)
Other Consumables	227	(500)
Allowance for Items Consumed During Refurbishment	227	(500)
Contingency	345	(760)
Total	28,287	(62,360)

Ablative TPS Option 2. Ablator Sprayed on Replacable Panels		
Propellant	26,308	(58,000)
TPS: Ablator on Panels	862	(1900)
Containers for Panels	181	(400)
Remaining Items	936	(2060)
Total	28,287	(62,360)

Recyclable TPS and All Propulsive		
Propellant	27,352	(60,300)
Remaining Items	936	(2060)
Total	28,287	(62,360)

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case, space basing increases the all-up weight by approximately 2222 kg
(4900 lb) or by 7.76%. To a geostationary orbit, the delivered payload is
increased by some 20% and the round trip payload by 21%, yielding approxi-
mately 6580 kg (14,500 lb) delivery capability and 4000 kg (8800 lb) round
trip capability. Vehicle dry weight increases to carry the extra propellant
could use up to one quarter of these increases in payload capability. If the
payload is transported separately, or is also space based, such as a manned
module for the round trip mission, then the increased payload would be 7545 kg
(16,600 lb) delivered and 4625 kg (10,200 lb) round trip. Again some 25% of
the increases could be used for increasing the tank capacity.

The payload increases to low earth orbit may be estimated from the
increase in propellant. The ratio of payload to propellant increases with
decreasing total propulsive Δv requirements. Below 900 km (490 n.mi.)
the Δv requirement for AMOOS is greater than the all propulsive Δv if the
mission involves no plane change for the OTV. In general, aeromaneuvering
will only be used above this altitude.

Table 15 can be used to determine the effects of increase in available
propellant. In general, the current AMOOS vehicle has sufficient tankage
to fly the missions of Table 15 except the 10,000 km (5400 n.mi.). If space
based refurbishment is available then, from Table 14 an extra 2222 kg (4900
lb) of propellant and payload can be carried. The ratio that this must be
split into between payload and propellant is given in Table 15. The resulting
increases in payloads are given in Table 16.

The basic assumption in Tables 15 and 16 is that two and four shuttle
launches are required into a 28.5 deg and polar orbit, respectively, to use
the full capability of the AMOOS vehicle. A 1000 km (540 n.mi.) mission
requires approximately five Shuttle launches to a 28.5 deg inclination orbit
and ten into a polar orbit (WTR). The resulting vehicle with payload would
weight some 145,000 kg (320,000 lb). The resulting payload increase would
be, then, some 2000 kg (4400 lb) on 200,000 kg (440,000 lb) or about 1%. For

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Table 15

AMOOS PAYLOADS TO LOW EARTH ORBIT (CIRCULAR)
50,000 KG (110,000 LB) ALL-UP WEIGHT

Altitude		Payload			
		Delivery		Round Trip	
km	(n.mi.)	kg	(lb)	kg	(lb)
1,000	(540)	42,200	(93,000)	38,900	(85,800)
2,000	(1080)	37,800	(83,300)	33,200	(73,200)
4,000	(2160)	31,800	(70,100)	26,000	(57,200)
10,000	(5400)	23,200	(51,100)	17,200	(37,900)

Propellant				Payload/Propellant Ratio	
Delivery		Round Trip		Delivery	Round Trip
kg	(lb)	kg	(lb)		
4,535	(10,000)	7,835	(17,300)	9.3	4.96
8,935	(19,700)	13,535	(29,800)	4.23	2.45
10,956	(24,200)	20,735	(45,712)	2.90	1.25
23,535	(51,900)	29,535	(65,100)	0.99	0.58

Table 16

INCREMENTAL PAYLOAD FROM SPACE BASING

Delivery

Altitude		Payload		Propellant	
km	n.mi.	kg	lb	kg	lb
1,000	(540)	2006	(4423)	216	(476)
2,000	(1080)	1797	(3962)	425	(937)
4,000	(2160)	1652	(3642)	570	(1256)
10,000	(5400)	1105	(2437)	1117	(2462)
Geostationary		453	(1000)	1769	(3900)

Round Trip

Payload		Propellant	
kg	lb	kg	lb
1849	(4077)	373	(822)
1578	(3479)	644	(1420)
1234	(2721)	988	(2177)
861	(1798)	1406	(3100)
635*	(1400)	4535	(10000)

*Manned mission, where manned module is refurbished in space.
(Single Shuttle launch)

very low earth orbits, then, space basing yields a 1% increase in payload for optimum use of the OTV to about 5% for least efficient use of the OTV. In each case, the most efficient use is made of the Shuttle. The percentages rise until for geostationary missions, a 20% increase in efficiency may be achieved.

A further possibility that has been discussed involves any unused propellant reserves that the Shuttle may have on achieving low earth orbit. The details involved in making any remaining reserves available for use are many and far reaching. Undoubtedly, they involve Shuttle operation and safety and the possibility of taking the ET into orbit. Most of the reserves will be in the ET. Furthermore, these reserves cannot be relied upon for any one particular flight. The best that can be done is to assign probabilities to the unused reserves being greater than a given value. Hopefully, the probability of them being greater than zero is 100% on each flight. The average or expected value is 2360 kg (5200 lb) much of which is in the ET, the remainder is in the SSME lines. The maximum that can be salvaged is obtained by taking the ET into orbit. The main engines would be burned longer, if there are any reserves, of course, to the point where the OMS usage to put the ET and the Orbiter into Shuttle parking orbit is the same as to put the Orbiter only into parking orbit from the current SSME shutdown orbit. The average unused propellant reserves would be some 1300 kg (3000 lb) to 1800 kg (4000 lb). The ET would be in the 295 km (160 n.mi.) parking orbit. Assuming that it requires a 150 m/s (500 ft/sec) Δv to deorbit yields a propellant requirement of 1190 kg (2618 lb) at I_{sp} = 450 sec. This reduces the reserves essentially to zero. Furthermore, a means of deorbiting has been assumed. What has in essence been shown is that the current mode of operation of the Shuttle is optimum if the ET is to be reentered.

The AMOOS configuration in the above applications is unchanged. The maximum propellant is assumed approximately 22,500 kg (50,000 lb) which, at most, is a minor modification to the current configurations. These configurations can move 100,000 kg (200,000 lb) or more from one low earth

orbit to another. However, to lift these large payloads to high energy orbits requires better than an order of magnitude increase in propellant. Transporting such vehicles to low earth orbit requires either in-orbit assen..ly or transportation by an HLLV. In particular, the propellant tanks pose a problem, since multiple tanks lead to complicated plumbing and weights and balance problems.

● Space Refurbishment Base

If the AMOOS vehicle is space based then a low earth orbit refurbishment base is required. Ideally, the base will provide a shirt sleeve environment for the maintenance crews. A conceptual design of such a base is given in Fig.39. The base can be folded for transportation in the Shuttle cargo bay.

The refurbishment operations that can be performed in the base are expected to differ from those that can be performed on earth. The undesirability of providing the amount of venting as on earth will preclude the use of some toxic materials, for example, berrylium-aluminum may not be worked in space. Certain operations which generate small particles are undesirable, such as grinding. The zero gravity environment will impose problems, for example, scraped off ablator will float, and furthermore ultrasonic water jets cannot be used to wash it off as is probably feasible on earth. Upon consideration of such impending problems, space based refurbishment should consist of removing and replacing parts. As an example, the ablative TPS should be made on metal panels. The complete panel would be removed and replaced. The application of TPS in space would then be limited to filling holes left for fastening the panels to the AMOOS primary structure.

Certain equipment will also be required in a space refurbishment base. No attempt has been made to identify this equipment since the space based refurbishment operations may bear little relation to the corresponding earth based operations.
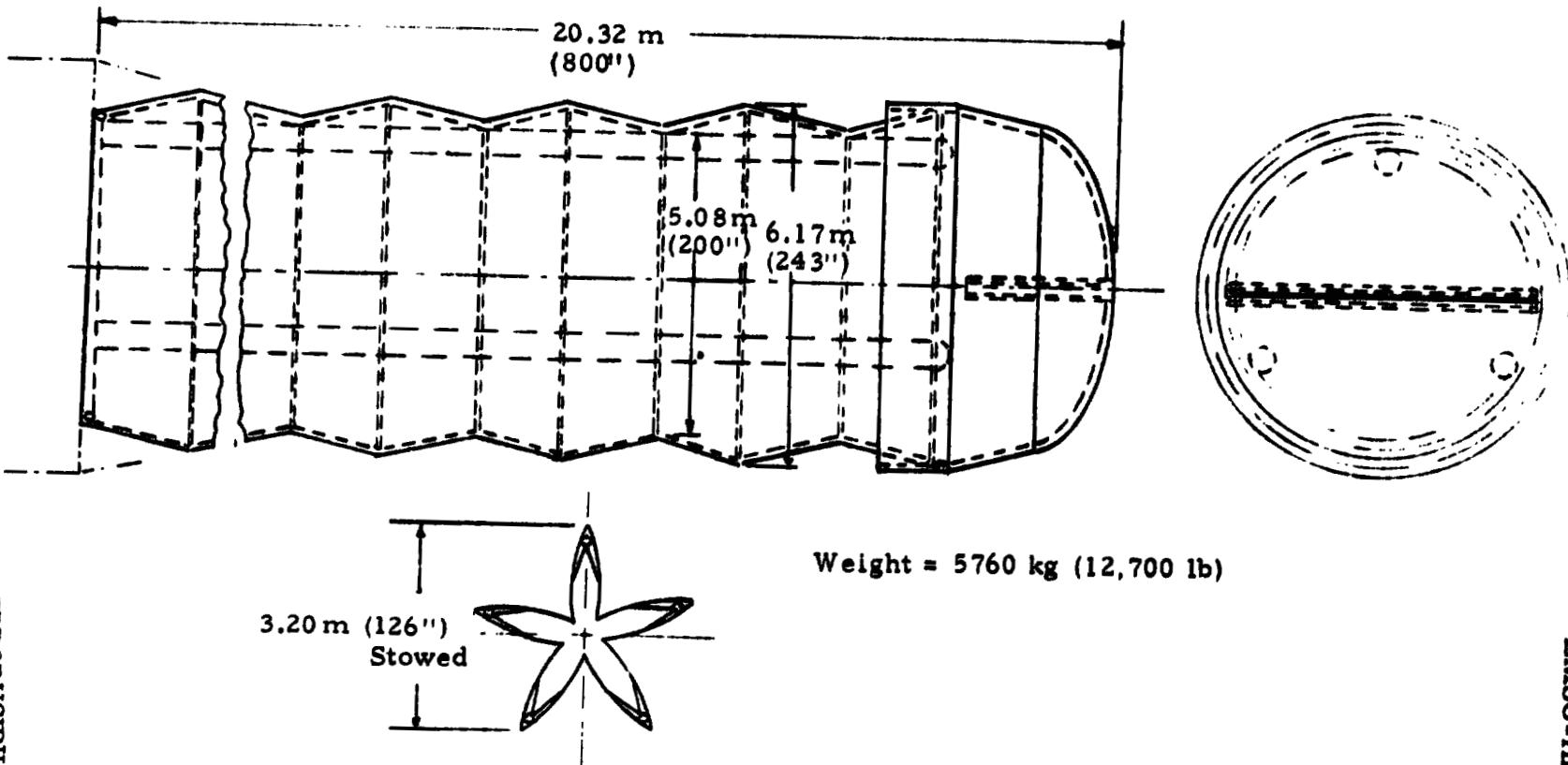
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Fig. 39 - Space Refurbishment Base

LMSC-HREC TR D497150

The refurbishment facility will, of course, be in the Space Base orbit. Such an orbit will be required to ensure an orbit lifetime of ten years or more. It will also minimize the commuting of crews to the refurbishment base from their habitat at the Space Base. This means that either the space based OTV or possibly a smaller vehicle is required to go down to the Shuttle delivery orbit to transport the propellants and refurbishment materials to the Space Base orbit. This change in the mode of operation will degrade the overall vehicle performance slightly. More important, however, are the increases in complexity of the mission and the propellant losses during the propellant transfer operations. Studies at MSFC have shown that under adverse circumstances as much as 25% of the propellants may be lost.

Section 8
CONCLUSIONS

These studies have continued to show the potential advantages of AMOOS over the all-propulsive OTV. In particular, the kit concept studies and the dual fueled AMOOS studies have shown its versatility and option potential over the all-propulsive vehicle. All of this potential of AMOOS and AMRS depends upon the ability to control the trajectory during atmospheric flight and so use an ablative TPS. In turn, this TPS must be lightweight which can be attained by spraying a lightweight ablator (e.g., Martin Marietta SLA 561) directly onto the load bearing skin. The significant findings of each subtask are given below.

AMOOS proved more readily adaptable to the kit concept than the Cryogenic Tug. In general, AMOOS outperformed the all-propulsive kit AMOOS and the Cryogenic Tug. Furthermore, AMOOS may be readily adapted to the kit concept without payload penalty.

The ablative TPS is still preferred over those using reradiative and insulative materials. The ablator yields a lighter TPS with a higher temperature range. Other materials, except carbon-carbon, are restricted to low energy missions or multiple-pass maneuvers. The latter requires many passes through the Van Allen radiation belts.

The development of the space sextant will provide autonomous navigation capability. Other systems, such as the interferometer tracker and the landmark tracker, may also provide autonomous operation. Development is required of both systems.

The preferred AMRS configuration uses an expendable solid rocket motor. The dual recovery mode of operation is feasible and carries a penalty of approximately 10% of its dry weight. The dual modes considered are

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Shuttle rendezvous and ground recovery. Land impact is preferred for the ground recovery mode.

The dual fueled AMOOS has a sufficient round trip payload capability to rotate a four-man crew to geostationary orbit. Its round trip payload capability is approximately six times that of the all-propulsive, dual fueled vehicle.

Six-man through 18-man crew modules may be round tripped using a Growth AMOOS vehicle and Growth Shuttle. The 24-man module requires either a single stage Growth AMOOS or staged baseline AMOOS vehicles delivered to low earth orbit by the 130K Shuttle derived HLLV.

The payload performance of AMOOS with the Growth Shuttle and Shuttle derived HLLVs is greatly enhanced. The OTV performance using the Shuttle derived HLLV is further enhanced by using two AMOOS stages. AMOOS requires considerable modification for ground recovery since, in its baseline configuration, it cannot perform a horizontal landing.

Space basing may yield small payload advantages; however, the potential increases are of the order of 15 to 20% in round trip payload capability. The use of Shuttle FPRs to further augment the OTV propellant is fraught with uncertainties and potential difficulties. More study is required; however, the current mode of operation of the Shuttle appears optimum so that making the FPRs available in low earth orbit would probably decrease the Shuttle payload performance.

Section 9
RECOMMENDATIONS

To date, studies have shown the great potential of AMOOS over the all-propulsive OTV. However, further work is required to increase the confidence in AMOOS and credibility in the results since AMOOS is operating on the edges of current technology. The one thing that would dispel all doubt and provide a wealth of design data is, of course, the model flight test. The recommendations are basically to develop a model flight test plan, with costs, and perform several supporting technology tasks.

9.1 MODEL FLIGHT TEST PLAN DEVELOPMENT AND EVALUATION

Under this task the flight test plan will be developed in detail and thoroughly evaluated. The overall task will be divided into subtasks as discussed below.

● Identify Data Required to Meet Objectives

Detailed data requirements will be established such as loss of TPS due to ablation, navigation data, aerodynamic loads, etc.

● Identify Hardware Requirement

Hardware necessary to measure and record and/or transmit the data will be identified. Sensors by type and model will be identified, if possible, together with supporting hardware and power requirements.

● Determine Flight Test Trajectory

Flight test trajectories will be determined which yield the environment, spatial position and vehicle attitude necessary to gather realistic data.

● Model Conceptual

From the above requirements, the flight test model conceptual design will be developed. The objective of this design will be to develop the system in sufficient detail for a meaningful cost analysis.

● Shuttle and IUS Requirements

The use of the Shuttle and, if necessary, IUS propulsion stages will be determined. The impact on the Shuttle flight will be evaluated.

● Identification of Alternatives

'dentify several model configurations and modes of operation. Identify data that can be obtained from each configuration and mode of operation. In particular, identify small models that may be used in Shuttle tether tests.

● Cost Evaluation of the Model Flight Test

The data generated above will be used to estimate the cost of the flight test program. Effort will be made to identify the most cost effective method.

● Estimate Cost Effectiveness of AMOOS

The payload performance of AMOOS will be evaluated against AMOOS costs including model flight test costs. The resulting cost estimates may be used in evaluating AMOOS as an OTV. The cost estimates for AMOOS will be generated under various assumptions; for example, use of an existing RL10 or modified RL10 engine or a new engine such as the ASE.

The output of this study will be a detailed model flight test plan, the cost of the proposed model flight tests and a cost effectiveness of AMOOS in the form of dollars per pound of payload. The Shuttle usage will be included in these studies since AMOOS can frequently do, in one Shuttle launch, tasks which require two Shuttle launches for the all-propulsive system.

9.2 SUPPORTING TECHNOLOGY

● Split Flap Studies

The AMOOS and AMRS body flap has been designed for longitudinal trim only. It could be used for pitch control and, if split, yield the possibility for aerodynamic roll control. The specific tasks would be similar to those of side flap studies for lateral control.

● Guidance

Guidance Development: The linear regulator guidance method can be refined by two modifications to the performance index. The first is to replace the bank angle term by the bank angle acceleration. This will result in a reduced attitude thruster fuel consumption. The second modification is to include a combination of position and velocity at atmospheric exit in the performance index. The purpose of this modification is to minimize the variation of the phasing time with the Shuttle.

Manual Guidance: The possibility of pilot interaction with the guidance system is a requirement during normal operation as well as a fail safe mode in case of a massive failure. For the normal operating system several levels of interaction between pilot and guidance system will be identified and analyzed. These levels of interaction will include a supervisory mode, an active interaction mode and a manual mode. A fail safe manual mode will be developed to be used in case of a massive failure. A guidance technique developed earlier and entitled "Velocity Lost Approach" may be suitably modified for that purpose.

● Navigation Studies

The initial phase of the navigation studies consists of determining the effects of navigational accuracy on atmospheric flight guidance and phasing with the Shuttle. The results thus obtained will be evaluated against acceptable phasing orbit variations to yield acceptable navigation errors. The

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study will include exoatmospheric navigation (midcourse correction) as well as navigation during the atmospheric flight. The established navigation error budget will then be used to evaluate existing hardware, define required or desirable technology and compare to that required for the Baseline Space Tug. The end result will be a practical set of navigational accuracies, navigational hardware and desired or required technology.

● AMOOS Structure — First Year

Test panels of typical integral stiffened structure designed for the AMOOS shell will be fabricated from candidate metallic and/or non-metallic materials. The panels will be approximately 50 x 50 cm (20 x 20 in.). Various TPS materials will be applied to the panels and thermocouples attached. Panels will be cycled through a typical mission environment and thermal distribution recorded. After testing, specimens will be examined for bond line and TPS failure and possible damage to the stiffened panel. Refurbishment of the TPS on the panel will be performed and tests repeated.

● AMOOS Structure — Second Year

The most promising configuration from the preceding test series will again be fabricated and tested with both the thermal and mechanical load applied simultaneously. Strains, deformations and temperature distributions will be recorded. The panels will be cycled through a typical mission environment with the corresponding thermal and mechanical loads applied. Panels will be examined after each test for TPS and structural failure.

● Wind Tunnel Testing

This task is divided into aerodynamic heating and aerodynamic force and moment tests. These are discussed separately as follows:

Aerodynamic Heating Tests: The objectives of these tests are: (1) to determine lee side heating rates for a range of angles of attack; and (2) check

the predictive method for estimating heating rates and temperatures. An
AMOOS Stycast model will be used for these tests. The heating rate will be
determined using temperature sensitive (Tempilaq) paint. Side and bottom
view movies at speeds of 16 frames/sec will be taken of the model. Shadow-
graphs will be taken at 10 deg angle-of-attack increments for every run.

The task will include an evaluation of test facilities against test require-
ments. A test facility will be chosen on the basis of meeting test objectives,
cost effectiveness and availability.

Aerodynamic Force and Moment: These tests will be performed using
existing and modified models to determine: (1) the effect of Reynolds number
on the aero forces and moments; (2) the effects of various flap configurations;
and (3) forces and moments at Mach numbers closer to flight values than those
of tests conducted under a previous contract.

● Lateral Control Using Aerodynamic Surfaces

This study would be spread over two years. In the first year the use
of side flaps would be studied for lateral control of the AMOOS and AMRS
vehicles, whereas in the second year the trades among the various flap options
and RCS lateral control, or a combination of each, would be studied.

First Year: (1) Establish flap planform options and locations on ve-
hicle; (2) size flaps and compute forces and moments due to flaps; (3) per-
form a preliminary design of flap structure, attachment, actuating mechanism,
and TPS; and (4) perform weights analysis of flaps and related subsystems.

Second Year: (1) Perform trade studies between weight added for flap
and related subsystems and weight removed through the reduced RCS require-
ment; (2) determine the differences in the guidance and control requirements
for the flaps and RCS, and establish trades; (3) determine the effect of flaps
on the vehicle structure and establish trades; and (4) evaluate flaps in compar-
ison with RCS for lateral control during atmospheric flight.

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- External Geometry Optimization

The weight and lateral maneuvering of the AMOOS and AMRS designs
are dependent upon the external geometry. The current design was selected
from a family of shapes yielding a high drag coefficient with little regard for
the attendant lift coefficient. The vehicles were selected to fly at a 45 deg
angle of attack. Limited trade studies and previous results have shown that
the TPS weight is both external geometry and angle-of-attack dependent.
Furthermore, a higher L/D ratio would yield more lateral maneuverability
as well as more trajectory control. In this task it is proposed to investigate
the effects of higher L/D and external geometry on vehicle performance and
so optimize the external geometry, flight attitude and mode of operation during
atmospheric flight.

- Optimum Dual-Fueled Operation

In this study, two modes of operation would be considered. One mode
would be applicable to the short on-orbit lifetime vehicle and one to the long-
lifetime vehicle. In each case an initial h density fuel burn is followed by
a cryogenic fuel burn to achieve mission orbit. After completion of its mis-
sion the short-lifetime vehicle would use cryogenic propellants to return,
whereas the long-lifetime vehicle would use space storable propellants.
The optimum Δv values for each propellant would be determined using dry
weight minimization and payload maximization as criteria.

9.3 DESIRABLE NEW TECHNOLOGY

- Lightweight, Recyclable TPS

The requirements for a lightweight, recyclable TPS material can be
established from the AMOOS and AMRS thermal environments. The develop-
ment of a material with a temperature range equal to that of Carbon-Carbon,
recyclable at least 20 times and with a density of not more than that of LI-900
would greatly enhance the operation of an aeromaneuvering OTV. Such a
material would have applications to a wide range of vehicles, including the
Growth Shuttle, SPS launch vehicles and planetary probes.

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