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PLANETARY/DoD ENTRY TECHNOLOGY FLIGHT EXPERIMENTS

DoD Entry Flight Experiments

By H.E. Christensen, R.J. Krieger, W.R. McNeilly and H.C. Vetter

Prepared by

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for Ames Research Center

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DoD Entry Flight Experiments

By H.E. Christensen, R.J.Krieger, W.R. McNeilly and H.C. Vetter

Prepared Under Contract No. NAS 2-8678 by

MCDONNELL DOUGLAS ASTRONAUTICS COMPANY – EAST

Saint Louis, Missouri

for

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION

FOREWORD

This final report was prepared by McDonnell Douglas Astronautics Company-East (MDAC-E) for NASA Ames Research Center Contract NAS2-8678, Planetary/DoD Entry Technology Flight Experiments. It covers the period 1 April 1975 to 29 February 1976. This effort was performed for the National Aeronautics and Space Administration, Ames Research Center, under the direction of the Thermal Protection Branch with Dr. Phillip R. Nachtsheim as Contract Technical Monitor and with the cooperation of Capt. R. J. Callahan of SAMSO and R. C. Loesch of Aerospace Corporation as advisors for the DoD portion of the study.

Special acknowledgements are in order to Duane Dugan of Ames Research Center and Ray Holland of the Massachusetts Institute of Technology/Lincoln Labs for their assistance in providing data for this study. Major contributors from MDAC-E included Messrs. H. E. Hemmes, L. J. Mockapetris, C. D. Poore, and J. L. Sedwick.

The report consists of four volumes:

- Volume I - Executive Summary
- Volume II - Planetary Entry Flight Experiments
- Volume III - Planetary Entry Flight Experiments Handbook
- Volume IV - DoD Entry Flight Experiments

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ABSTRACT

The objective of the DoD Entry Technology Flight Experiments study was to assess the feasibility of using the Shuttle to deploy DoD Entry Technology Flight Experiments. The key questions address by this study were:

1) What are the DoD needs for the 1980's?
2) How can Shuttle be used to fulfill these DoD needs?
3) What modifications to DoD testing procedures are required to use Shuttle?
4) What modifications to Shuttle are required to accommodate DoD missions?
5) What unique capabilities does Shuttle provide for DoD testing?

The major conclusion of this study is that Shuttle can provide unique simulation capability for DoD reentry vehicle experiments. Shuttle launches from either KSC or VAFB into a nominal 160 NMI circular orbit can provide DoD payload impact at Kwajalein, Poker Flat or Hawaii. The KREMS radars at KMR can track and record reentries from Shuttle deployed experiments approaching from the southwest. As a consequence, data will be of the same type, amount, and quality as that currently acquired from a VAFB to KMR launch into the northeast corridor at KMR. Shuttle deployed payloads can provide simulation of 1500-7100 NMI trajectories, high velocity (25 kft/sec) - shallow flight path angle (-5 deg) reentries, multiple payloads for Site Defense Radars, and overland terminal trajectories into Poker Flat, Alaska. There are certain DoD missions for which Shuttle is not well suited. These are the low velocity (20 kft/sec) - steep flight path angle (-40 deg or greater) reentries and eastern approaches to KMR. A payload deployment system (PDS) is required to deorbit payloads from Shuttle orbit. The preferred PDS consists of a liquid propellant booster, spin separation and attachment system, and the RV's. Use of an existing booster and a proven spin separation system minimizes development cost and risk of the PDS. For most of the DoD missions less than half of the Shuttle payload bay is required by the PDS. Improvements in the payload to Shuttle RF link and new procedures to provide tracking during PDS burns are needed. Pre-deployment checkout of the PDS will require either a separate onboard checkout console or data transmittal to the ground via one of Shuttle's wide band data links. Shuttle costs over and above traditional ground launches will involve primarily the communications aspects of the missions, i.e., checkout console, satellite or ship coverage during PDS burns, and special PDS equipment to interface with Shuttle data links. In general, the Shuttle can provide the DoD with unique simulation capability to meet their needs of the 1980's.

GLOSSARY

ALCOR	ARPA-Lincoln, C-band observables radar (C-band radar KREMS)
ALTAIR	ARPA-Long Range Tracking and Instrumentation Radar (UHF/VHF radar at KREMS)
ARPA	Advanced Research Projects Agency
BOA	Broad ocean areas
Booster	Large upper stage that provides initial thrust after the PDS has been deployed from Shuttle, e.g., Transtage, Delta, Centaur, Agena, Burner II
Bus	The term bus is used in two ways: 1) Synonymous to the Deployment Bus in Reference 3, used as a second stage for spacing RV's 2) Minibus used as primary stage for low ΔV missions
ETR	Eastern test range
IUS	Interim Upper Stage
KMR	Kwajalein missile range
KREMS	Kiernan reentry measurements site
KSC	Kennedy Space Center
Minibus	A class of buses that fit compactly inside the Shuttle cargo bay
Minibus (OMS)	A minibus constructed using Shuttle orbital maneuvering system (OMS) components
Minibus (RCS)	A minibus constructed using Shuttle Reaction Control System (RCS) components
PDS	Payload Deployment System is the Shuttle deployed payload. The assembly of a booster and/or bus, plus spin separation system plus RV's
PENAID	Penetration aid
RV	Reentry Vehicle
SAFSCOM	Strategic Air Force Systems Command
SAMSO	Space and Missiles Study Organization
SDR	Site defense radar
Shuttle	Space Shuttle
SRB	Solid rocket motor IUS
TRADEX	Target Resolution and Discrimination Experiment (S-band radar at KREMS)
Transtage	Same operational characteristics as the Representative IUS described in Reference 3
VAFB	Vandenberg Air Force Base
WTR	Western test range

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1.0 INTRODUCTION

The objective of the DoD Entry Technology Flight Experiments study was to assess the feasibility of using the Shuttle to deploy DoD Entry Technology Flight Experiments. It is timely to evaluate Shuttle's capability to do the type of testing that our Department of Defense has been engaged in for many years. Such testing will continue to have high priority as the state of the art expands in nose tip technology, heat shields, maneuvering reentry vehicles, terminal guidance, environment survivability provisions, penetration aids, multiple reentry vehicle deployment, and discrimination technology. Examples of how Shuttle may be of benefit include: (a) providing unique capability not available in ground launch systems, (b) targeting flexibility for investigating reentry into adverse atmospheric environments (rain, ice, snow), and (c) the cost effectiveness potential of conducting numerous experiments with a single launch. In addition, the Yardley/Lebarge accord, Reference 1, on the use of Shuttle establishes the basis for the Shuttle replacing most of the ground launch systems in the late 1980's.

This study was task oriented and the study flow is shown in Figure 1. The identification of DoD needs was a data collection task which was then summarized in the form of simulation requirements. Simulation requirements defined the entry vehicles, test operations, and scenarios which were postulated for the 1980's DoD testing. In parallel to these tasks, information on the available boosters and their characteristics was collected. A delivery strategy analysis defined both the characteristics of booster burns required to deorbit payloads and the Shuttle orbital parameters. These data coupled with the booster capabilities served as inputs to a delivery system sizing and performance estimate. The delivery system sizings then were used to develop in detail the designs and interfaces. The analyses of the communications,accuracy, range safety, servicing, and cost aspects of the missions provided the detailed data to define the feasibility of doing specific missions.

The key questions address by this study were:

1) What are the DoD entry test needs for the 1980's?
2) How can Shuttle be used to fulfill these DoD needs?
3) What modifications to DoD testing procedures are required to use Shuttle?
4) What modifications to Shuttle are required to accommodate DoD missions?
5) What unique capabilities does Shuttle provide for DoD testing?

By completing the tasks identified in Figure 1, these questions were answered. The following summary provides a brief description of the direction and conclusions of this study.

DoD ENTRY TECHNOLOGY FLIGHT EXPERIMENTS STUDY FLOW

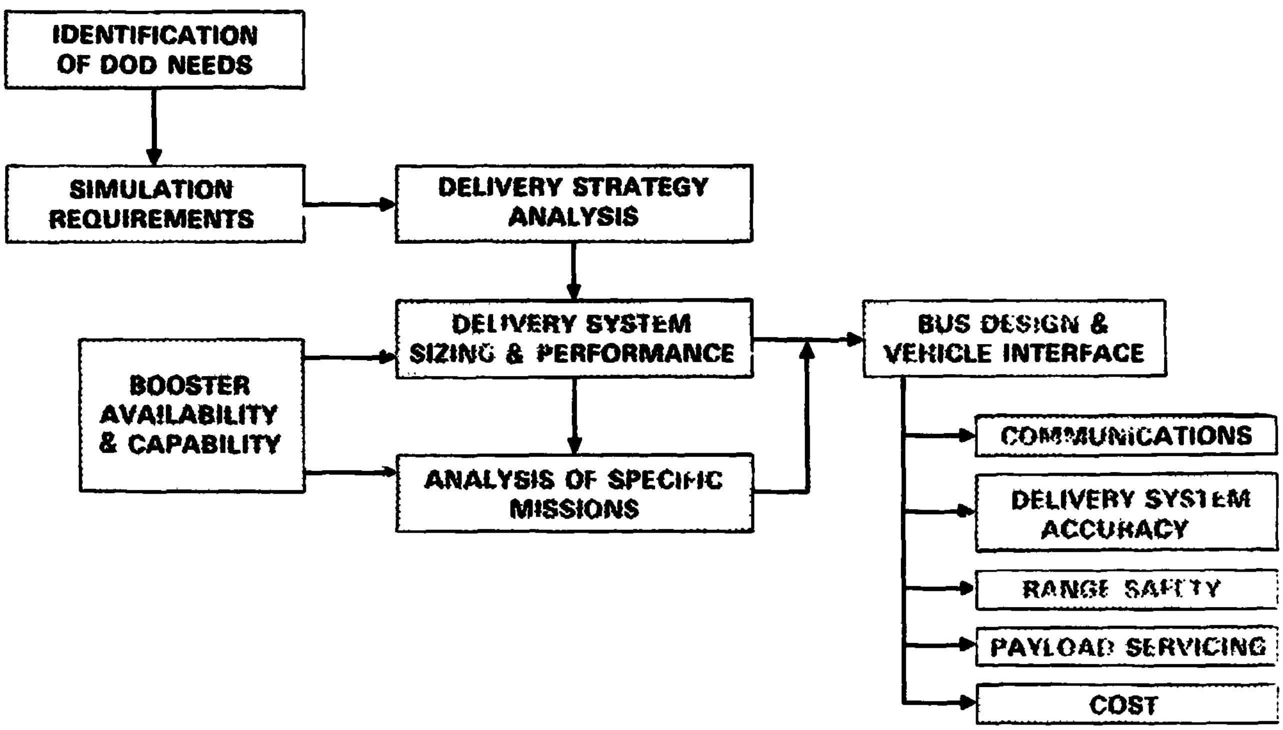


FIGURE 1

2.0 SUMMARY

The major conclusion of this study is that Shuttle can provide unique simulation capability for DoD reentry vehicle experiments. The areas of uniqueness are listed in Figure 2 for the typical DoD experiment mission shown in Figure 3. The Shuttle can be launched from either KSC or VAFB, is an orbiting launch platform, has a high weight payload capability, and can provide payload impact at Kwajalein Poker Flat and broad ocean areas. These conclusions were supported through the completion of the tasks identified in Figure 1.

The identification of DoD entry vehicle test needs for the 1980's was accomplished through interface meetings with SAMSO/Aerospace on several occasions. As a result, Reference (2) was issued to guide the initial assessment of the feasibility of using Shuttle to perform DoD missions. Further discussions led to a SAMSO letter, Reference (3), which described specific mission scenarios capitalizing on the unique capabilities of Shuttle.

Simulation requirements which resulted from this list of needs included the categories of:

<u>Reentry Vehicle</u>

1) Geometry
2) Service
3) Nuclear Safety (Radioactive sensors)
4) Test and Monitoring
5) Security

<u>Test Operation</u>

1) Shuttle Crew Activities
2) Communication Between Shuttle and DoD Test Director
3) Data Communications
4) Shuttle Data Acquisition of DoD Experiments
5) Abort Capability

<u>Deployment</u>

1) Range Safety and Debris
2) Weather and Ground Sensor

<u>Scenario</u>

1) Number of vehicles
2) Spacing of vehicles
3) Impact area

UNIQUE SHUTTLE REENTRY SIMULATION CAPABILITY

- SIMULATION OF 1500-7100 NMI TRAJECTORY
- SIMULATION OF HIGH VELOCITY (25 KFT/SEC) TRAJECTORIES
- SIMULATION OF FLAT REENTRIES (LOW ENTRY ANGLES)
- MULTIPLE PAYLOADS
  - RV'S
  - BUSES
  - SITE DEFENSE RADAR TARGETS
- IMPACT AREA SELECTION FLEXIBILITY
  - WEATHER EFFECTS
  - TERMINAL GUIDANCE OVER LAND
  - RANGE SAFETY PROBLEMS REDUCED

FIGURE 2

TYPICAL DOD EXPERIMENT MISSION

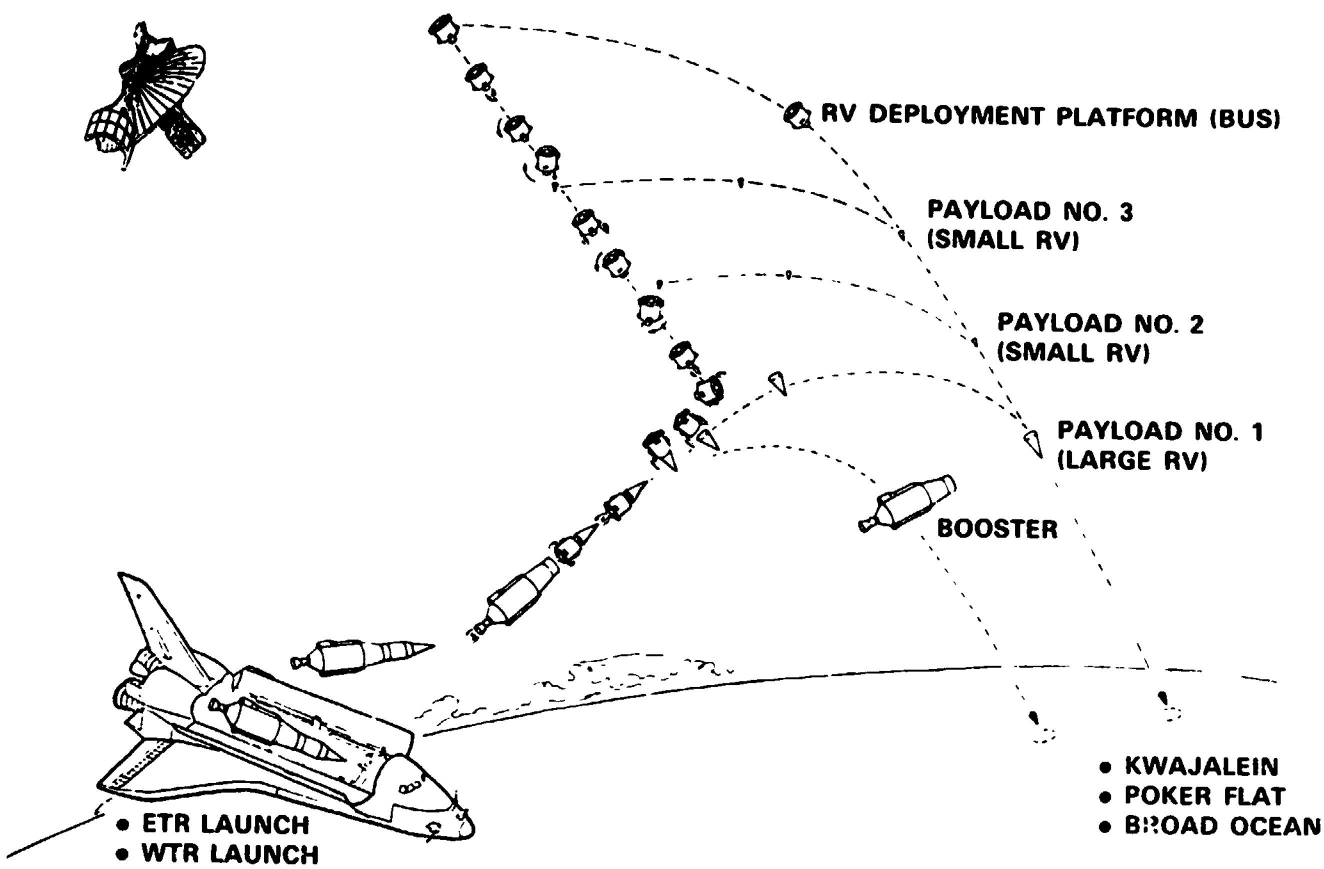


FIGURE 3

The next phase of the study established how well Shuttle could meet these requirements.

As a result of considering payload impact at KMR, a major study conclusion resulted, i.e., that the KREMS radars at KMR can be used to track and record reentries from Shuttle deployed experiments. As a consequence, the same type, amount, and quality of data can be acquired as is currently acquired from a VAFB to KMR launch into the northeast corridor at KMR.

Several delivery strategies were developed to investigate the trajectory shaping required to place multiple Shuttle payloads on a ballistic reentry trajectory to a precise impact point such as KMR. The types of booster burns, their locations on the Shuttle orbit and magnitudes were established. A preferred payload deorbit procedure was developed. This strategy, described in Figure 4, involves deploying the PDS from Shuttle in a 160 NMI circular orbit; the PDS then performs a plane change maneuver to target it for the impact point; a deorbit burn is then made to place the PDS on a ballistic reentry trajectory; spacing burns follow to provide the desired payload spacing at the pierce point.

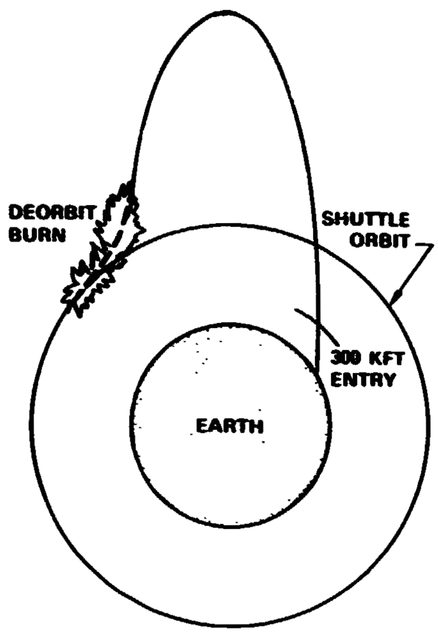
Using this strategy, the Shuttle has unique capability to provide high velocity shallow flight path angle reentries. Because the Shuttle is in a circular orbit with velocities near 25 kft/sec, it takes little booster energy to deorbit a payload from the Shuttle orbit at these velocities and shallow flight path angles. Accuracy of the pierce point location does suffer, however, because of tracking errors in Shuttle location and velocity. The uncertainty of downrange Shuttle velocity is the major contributor to range dispersions at the pierce point.

Because of the potential for a large number of burns and payloads, the PDS function is best served by a liquid propellant booster. The preferred PDS consists of a booster, spin separation and attachment system, and the RV's. Use of an existing booster and a proven spin separation system minimizes development cost and risk of the PDS. In special cases requiring only one large ΔV deorbit burn, a solid rocket motor booster suffices. Considerably smaller boosters made up of Shuttle RCS components more efficiently perform the function of the PDS booster for many of the low ΔV missions.

In parallel with establishing the mission strategies and PDS designs, communication coverage for telemetry, tracking, and commands was defined. Multiple telemetry system interfaces, including several options, were investigated. These interfaces included:

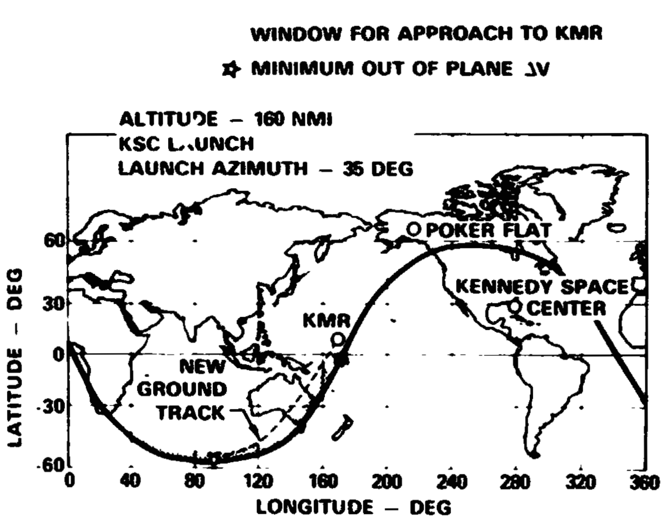
SHUTTLE/PDS MANEUVER STRATEGY STUDIES HAVE IDENTIFIED UNIQUE CAPABILITIES

DEORBIT



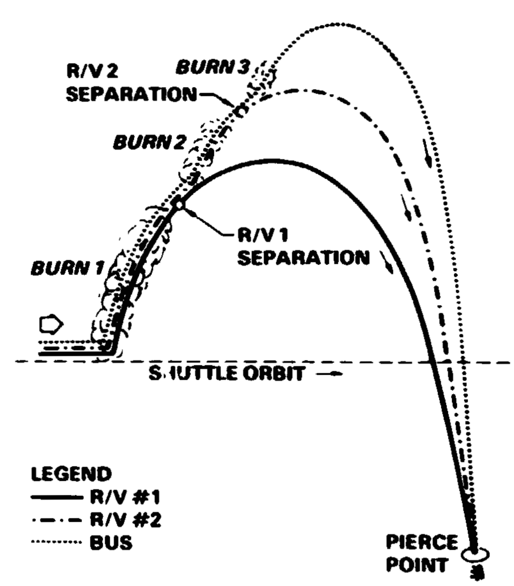
- ALLOWS TRAJECTORY RANGE SIMULATIONS OF 1500-7100 NMI

PLANE CHANGE



- PROVIDES FULL KREMS COVERAGE
- PROVIDES MULTIPLE PASS DEORBIT OPPORTUNITIES AT KWAJALEIN, HAWAII & POKER FLAT

PAYLOAD SPACING



- IMPROVES UPON MULTIPLE PAYLOAD NUMBER & WEIGHT OF GROUND LAUNCH SYSTEMS

FIGURE 4

a. Shuttle to ground

b. Payload to Shuttle

c. Payload to ground

d. Paylcad to satellite

Specific equipment and procedures requiring modification were identified as a result of this effort and are outlined in Figure 5. Improvements in the Payload to Shuttle RF link and new procedures to provide tracking during PDS burns were identified. If available Shuttle equipment is to be used for payload checkout, then the maximum data rate from each payload is limited. Where higher payload data rates are necessary, checkout will require either a separate onboard checkout console or the data must be transmitted to the ground via one of Shuttle's wide band data links.

Complete range safety studies are required before the range safety problems of Shuttle payload deployment can be fully assessed. In many situations, PDS burns are made with no ground coverage, and payloads are reentered over areas which have not been considered in previous range safety analyses. Specific mission analyses established firmly the capability of Shuttle to deliver DoD payloads at KMR for either the KREMS radars or Site Defense Radar and at Poker Flat. Complete range safety analyses are needed for each of these missions.

The cost analysis included cost estimates of Shuttle borne checkout and servicing equipment for DoD payloads and for any hardware or software item or service peculiar to Shuttle launched entry technology missions as compared to ballistically launched missions. Shuttle costs over and above traditional ground launches will involve primarily the communications aspects of the missions, i.e., checkout console, satellite or ship coverage during PDS burns, and special PDS equipment to interface with Shuttle data links.

This study demonstrated that Shuttle can perform the DoD missions of the 1980's and provide unique capabilities for high velocity-shallow flight path angle reentries at KMR and Poker Flat. The following text provides the detailed study approach and results which led to this conclusion.

SUMMARY OF COMMUNICATION SYSTEM CAPABILITY FOR DOD TECHNOLOGY

COMMUNICATION LINKS	PLANNED	ADDITIONAL NEEDS
SHUTTLE TO GROUND	S-BAND DIRECT S-BAND VIA SATELLITE Ku-BAND VIA SATELLITE	NONE
PAYLOAD(S) TO SHUTTLE PRE-SEPARATION	9 DATA INTERFACES 16 KBPS TO 50 MBPS 2 COMMAND INTERFACES	MULTIPLE PAYLOAD DATA LINKS OR HIGH DATA RATE PER PAYLOAD WILL REQUIRE OPERATION AND CHECKOUT CONSOLE ON SHUTTLE
POST-SEPARATION	1 DATA RF LINK 16 KBPS 1 COMMAND RF LINK 42 KM RANGE	NEED HIGHER DATA RATE FOR ADEQUATE MONITORING AND NEED GREATER RANGE. REQUIRES IMPROVED SHUTTLE RECEIVER AND ANTENNA.
PAYLOAD(S) TO GROUND	5 TO 8 WATT S-BAND TRANSMITTERS AND OMNI ANTENNAS PROVIDE 5000 KM RANGE ✓	NONE – SOME TARGET AREAS MAY NEED SUPPLEMENTING WITH SHIPS OR AIRCRAFT.
PAYLOAD(S) TO SATELLITE	20 TO 50 WATT TRANSMITTERS PLUS STEERABLE ANTENNA PROVIDE MINIMUM MARGIN ✓	USE OF TDRSS NEEDS MORE STUDY. POWER AND WEIGHT PENALTIES MAY EXCEED BENEFITS.
MISSION CONTROL	SCF GROUND STATIONS PROVIDE SGLS RANGING. C-BAND TRACKING FROM KMR, VAFB, HAWAII OR SHIPS. GLOBAL POSITIONING SATELLITE SYSTEM.	FOR SOME TRAJECTORIES AND TARGETS, GROUND TRACK IS INADEQUATE. SHUTTLE TRACKING IS RANGE LIMITED (560 KM). GLOBAL POSITIONING SYSTEM MAY BE ONLY SOURCE OF INFORMATION.

✓ RECOMMENDED FOR DOD USE

FIGURE 5

3.0 ESTIMATED DOD NEEDS

The identification of DoD entry vehicle test needs for the 1980's was accomplished through interface meetings with SAMSO/Aerospace on several occasions. As a result, Reference (2), describing estimated DoD needs for the 1980's was published by SAMSO. This document served to guide the initial assessment of the feasibility of using Shuttle to perform DoD missions. Once the initial feasibility assessment was completed, further discussions led to the publication of Reference (3), which described specific mission scenarios capitalizing on the unique capabilities of Shuttle. This letter served to guide the final study phase efforts and resulted in the analysis of specific missions described in Section 16.

In the following write-up, the contents of the first SAMSO document, Reference (2), are briefly described in Section 3.1. Because the document is classified, the reader is referred to the reference for specific information. The contents of the second SAMSO document are described in detail in Section 3.2.

3.1 Projected Missions - The following categories of DoD estimated needs are defined in Reference (2):

Reentry Vehicle

1) Geometry
2) Service
3) Nuclear Safety (Radioactive sensors)
4) Reentry Vehicle Test and Monitoring
5) Security

Test Operation

1) Shuttle Crew Activities
2) Communication Between Shuttle and DoD Test Director
3) Data Communications
4) Shuttle Data Acquisition of DoD Experiments
5) Abort Capability

Deployment

1) Range Safety and Debris
2) Weather and Ground Sensor

A list of eight reentry vehicle deployment scenarios was provided to serve as a baseline to determine compatibility with the Shuttle.

Several general DoD groundrules described in Reference (2) included: (1) payloads would be delivered to the pierce point (300 kft) with the DoD specified reentry conditions. Acceptable ranges for these parameters are given in Figure 6. Therefore, the atmospheric portion of reentry would be identical to that of a ground launch payload. This allows some flexibility in shaping the exoatmospheric portion of the trajectories. (2) The reentry vehicles and their onboard experiments would be identical to ground launch versions. This results in a wide range of instrumentation and experiments, described briefly in Figures 7 and 8. Shuttle capability to provide the telemetry channels and checkout facilities, therefore, is a primary consideration. Although specific reentry vehicle geometries are not identified, the geometries shown in Figure 9, were selected as typical and used throughout the study. (3) Three impact areas were desirable. These included Kwajalein, a broad ocean area, and an undisclosed land area. Later the broad ocean area was selected as Hawaii and the land area as Poker Flat, Alaska. Figure 10 summarizes the status, instrumentation and restrictions for these areas.

With these needs defined, Shuttle's capability to meet them was investigated. As Shuttle unique capabilities become better defined, a second set of DoD needs was developed to take advantage of these qualities.

3.2 Specific Mission Scenarios - Reference (3) documents the second generation ground rules which evolved from the initial feasibility assessment. A booster was selected for the purpose of evaluating a wide range of desirable missions and six specific missions. The booster was referred to as a representative IUS.

The booster corresponded to the Transtage and was assumed to have the following characteristics:

I_{SP}	301.3 sec
Usable propellant weight	23032 lb
Burnout weight	3751 lb
Length	178 in
Diameter	10 ft

Of particular interest during this phase of the study, was the high speed reentry at 25,000 ft/sec. To develop the total capability of Shuttle with Transtage a parametric was requested in which payload weight and number, and flight path angle were varied. The output of this parametric would be length, weight and specific impulse required for the payload delivery system as well as the sizing algorithms used for length and weight. These results are discussed in Section 9.

In addition specific missions described in Figure 11 were requested. Case 1 is a baseline for comparison with typical ground launch trajectories. Cases 2 through 5 take advantage of the high velocity-shallow flight path angle Shuttle

DoD PROJECTED REENTRY CONDITIONS FOR 1980's TESTING

INDIVIDUAL PAYLOAD WEIGHT:	30-3500 LB
REENTRY VELOCITY:	20,000-25,000 FT/SEC
REENTRY FLIGHT PATH ANGLE:	-5 - -60 DEG
RV LENGTH:	30 - 160 IN.
RV MAXIMUM DIAMETER:	9 - 50 IN.
NUMBER OF PAYLOADS:	1 - 8
SPACING AT PIERCE POINT:	0 - 90 SEC

FIGURE 6

TYPICAL DOD REENTRY VEHICLE INSTRUMENTATION

	TYPE	EXPERIMENTAL SUPPORT
1.	THERMOCOUPLES	HEAT SHIELD/NOSE TIP PERFORMANCE
2.	RADIOACTIVE ABLATION SENSORS	HEAT SHIELD/NOSE TIP RECESSION
3.	PRESSURE TRANSDUCERS	AERODYNAMIC PERFORMANCE
4.	VIBRATION AND ACOUSTIC SENSORS	BOUNDARY LAYER TRANSITION/VEHICLE PERFORMANCE
5.	RATE GYROS AND ACCELEROMETERS	VEHICLE DYNAMICS
6.	MAGNETOMETER	VEHICLE ROLL RATE
7.	RADIOMETERS/SPECTROMETERS	PLASMA AND WAKE OBSERVABLES
8.	ELECTROSTATIC PROBES	BOUNDARY LAYER PLASMA/OBSERVABLES
9.	RF REFLECTOMETERS	TELEMETRY PERFORMANCE/BOUNDARY LAYER PLASMA/OBSERVABLES
10.	MISCELLANEOUS SIGNALS	SUBSYSTEM PERFORMANCE/EVENT MONITORS
11.	S-BAND TELEMETRY	ALL
12.	C-BAND BEACON	RADAR ACQUISITION AND TRACKING
13.	HYDROMETEOR IMPACT PLUG	ADVERSE WEATHER EFFECTS

FIGURE 7

CURRENT AND PROJECTED DOD ENTRY TECHNOLOGY FLIGHT EXPERIMENTS

1. NOSE TIP TECHNOLOGY	o NOSE TIP PERFORMANCE o ABLATION EFFECTS ON VEHICLE STABILITY, PLASMA, WAKE OBSERVABLES, AND BOUNDARY LAYER TRANSITION o FINE WEAVE CARBON-CARBON (FWCC) NOSE TIP MATERIAL
2. HEAT SHIELDS	o HEAT SHIELD STRUCTURAL INTEGRITY o HEAT SHIELD MATERIALS EFFECTS ON PLASMA AND WAKE OBSERVABLES o EFFECTS OF MATERIAL DISCONTINUITIES AND FIBER ORIENTATION o BICONIC CONFIGURATION
3. PRECISION & MANEUVERING VEHICLES	o IMPROVED ACCURACY o EVASIVE MANEUVERS o CONTROL SYSTEMS DEVELOPMENT o ANTENNA PERFORMANCE
4. BOUNDARY LAYER EFFECT	o BOUNDARY LAYER TRIPPING o TRANSITION CRITERIA o TRANSITION EFFECTS ON VEHICLE DYNAMICS
5. TERMINAL GUIDANCE	o IMPROVED ACCURACY
6. ADVERSE ATMOSPHERIC CONDITIONS	o EFFECT OF RAIN OR SNOW ON PERFORMANCE
7. PENETRATION AIDS (PEN AIDS)	o ESTABLISH QUENCHANT REQUIREMENTS AND SIZE SYSTEMS o DEVELOP PRECISION DECOYS o FEASIBILITY OF TRAILING APPENDAGES
8. REENTRY VEHICLE DEPLOYMENT PLATFORMS	o DEVELOP MULTIPLE VEHICLE LAUNCH PLATFORMS
9. MIDCOURSE, DETECTION, TRACKING & SIGNATURE	o DEVELOP ADJUNCT SYSTEMS

FIGURE 8

REENTRY EXPERIMENT CANDIDATES

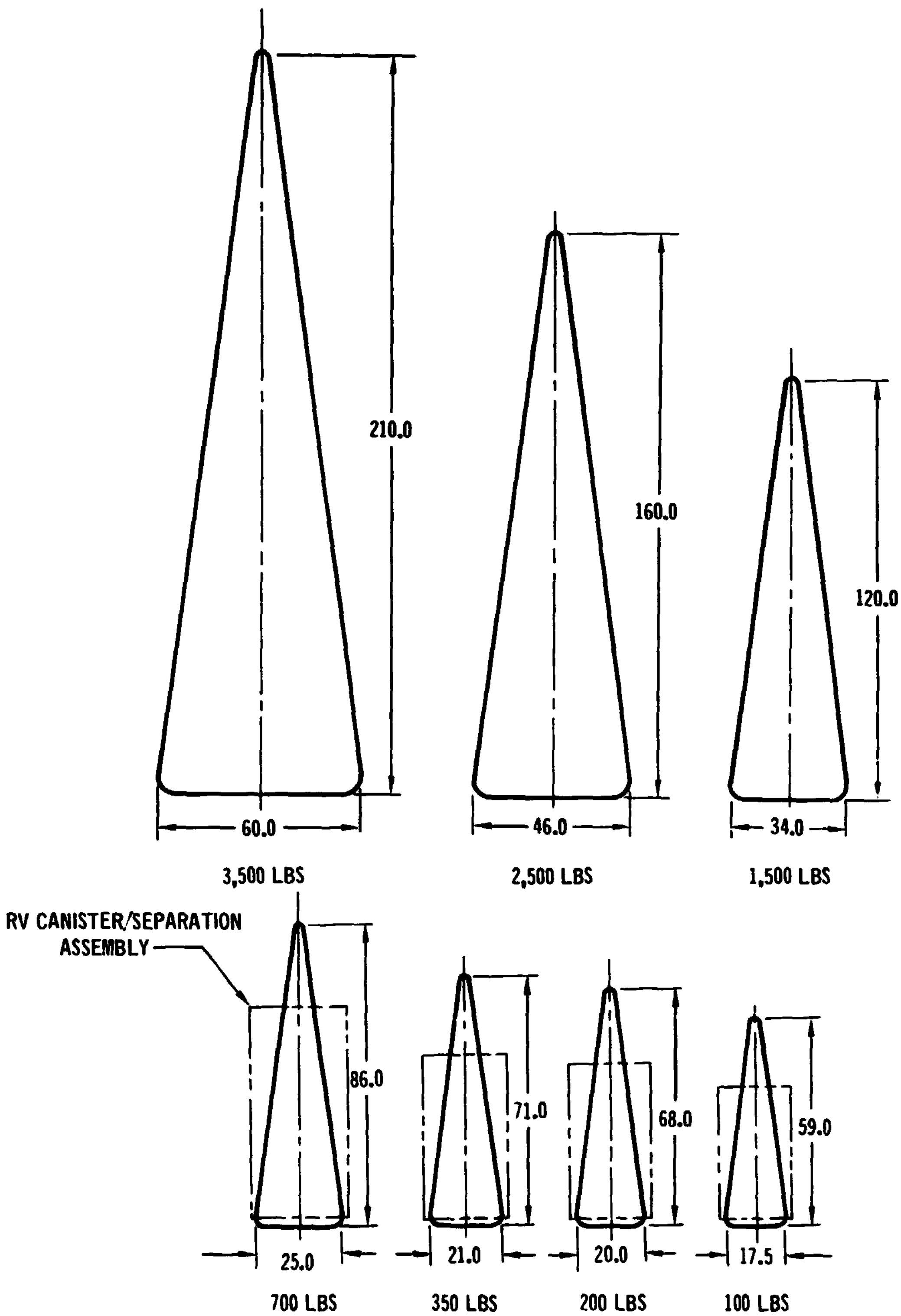


FIGURE 9

DOD REENTRY EXPERIMENT TEST SITES

IMPACT AREA	PRESENT STATUS	CURRENTLY AVAILABLE OFFBOARD INSTRUMENTATION	ENTRY RESTRICTIONS OR SPECIAL CONSIDERATIONS
1. KWAJALEIN (WTR)	OPERATIONAL	• TELEMETRY STATIONS • PRESS OBSERVABLES SENSORS (ALTAIR, ALCOR, TRADEX RADARS AND OPTICAL INSTRUMENTS) • TTR-4 BEACON TRACKING RADAR • TRAP AIRCRAFT (OPTICAL AND IR INSTRUMENTATION) • ARIA AIRCRAFT (TELEMETRY) • SAFEGUARD MSR RADAR	• ENTRY CORRIDOR CURRENTLY RESTRICTED TO AZIMUTHS FROM VANDENBERG AFB OR WAKE ISLAND • OTHER APPROACH AZIMUTHS ARE PROBABLY ACCEPTABLE PROVIDING NO OVERFLIGHT OF FOREIGN TERRITORY DURING ENTRY
2. BROAD OCEAN AREAS	NOT COMMONLY USED	• ARIS SHIPS (BEACON AND METRIC TRACKING RADARS AND TELEMETRY) • TRAP AND ARIA AIRCRAFT	• NO RESTRICTIONS EXCEPT THAT OVERFLIGHT OF FOREIGN TERRITORY REQUIRES LOCAL PERMISSION
3. ALASKA	NONEXISTENT	NONE	UNKNOWN

FIGURE 10

capability. Case 6 exploits its multiple payload capacity.

For each example case listed in Figure 11, the following items were requested;

1) Number of orbit opportunities available within Transtage's capability.
2) Range, azimuth, altitude and time of initiation relative to the respective test range of the first and subsequent burns of the Transtage.
3) Variations in velocity and flight path angle between multiple payloads.
4) Accuracy - Pierce point inaccuracies due to the estimated accuracy requirements of Transtage.
5) Communications - Identification of all potential communication augmentation needs in the Shuttle design and/or Transtage requirements relative to the estimated communication needs of the DoD reentry tests.
6) Range Safety - An identification of a feasible approach (e.g: using satellites) and of the equipment needed on the Shuttle and/or Transtage to implement: (a) real time knowledge at the impact range of the Transtage position, and (b) a command link to the Transtage from the impact range.
7) Payload Checkout and Servicing - Identification of all potential payload servicing augmentation needs in the Shuttle design relative to the estimated payload requirements including electrical and coolant fluid interfaces.
8) Costs - Planning type cost estimates of: (1) Shuttle borne checkout and servicing equipment for DoD payloads; (2) costs for any hardware or software item or service peculiar to Shuttle launched entry technology missions as compared to ballistically launched missions.

Items 1 through 3 are discussed in Section 16; item 4 in Section 10; item 5 in Section 12; item 6 in Section 15; item 7 in Section 12 and 13; and item 8 in Section 17.

SIX EXAMPLE CASES FOR SPECIFIC MISSION ANALYSIS

CASE NUMBER	REENTRY VEHICLE		PENETRATION AIDS		REENTRY CONDITIONS			
	NUMBER	WEIGHT-LB(EACH)	NUMBER	WEIGHT-LB(EACH)	γ DEGREES	V_E FT/SEC	SPACING (SEC)	IMPACT RANGE
1	1	600	1	30	28	22500	90*	KWAJALEIN
2	1	1000	-	-	5	25000	-	KWAJALEIN
3	2	1000	-	-	5	25000	90*	KWAJALEIN
4	1	1000	-	-	5	25000	-	POKER FLAT
5	2	1000	-	-	5	25000	90*	POKER FLAT
6**	2	350	4	30	28	22500	**	MECK ISLAND

*OR IDENTIFY MAXIMUM POSSIBLE SPACING IF 90 SEC CANNOT BE ACHIEVED BY TRANSTAGE

**SAME AS EXAMPLE H OF TABLE 1 OF REFERENCE (2).

FIGURE 11

4.0 SHUTTLE TRANSPORTATION CAPABILITIES

Definition o the launch location constraints and the resultant Shuttle orbital conditions was undertaken to determine the approach corridors at the impact points of interest. Both KSC and VAFB launches were considered. Figure 12 shows the Shuttle launch azimuth constraints at both launch sites. These data were obtained from Reference (4).

Orbit inclinations between 28.5 and 57 degrees are allowable for a KSC launch. The upper bound is established from range safety considerations and the lower bound from the latitude of the launch point. Because of the KSC launch constraints, the following generalizations can be made:

(1) Shuttle will be approaching Kwajalein, Hawaii, or Alaska from the west. Current ground launches from VAFB to Kwajalein approach from the east. As a result, the feasibility of approaching Kwajalein from non-traditional azimuths is addressed in Section 5.

(2) Because of the low orbit inclinations, no Shuttle overflight of Poker Flat impact areas at 67 deg latitude will be possible. However, plane change maneuvers by the payload deployment system can provide impact at these northern latitudes.

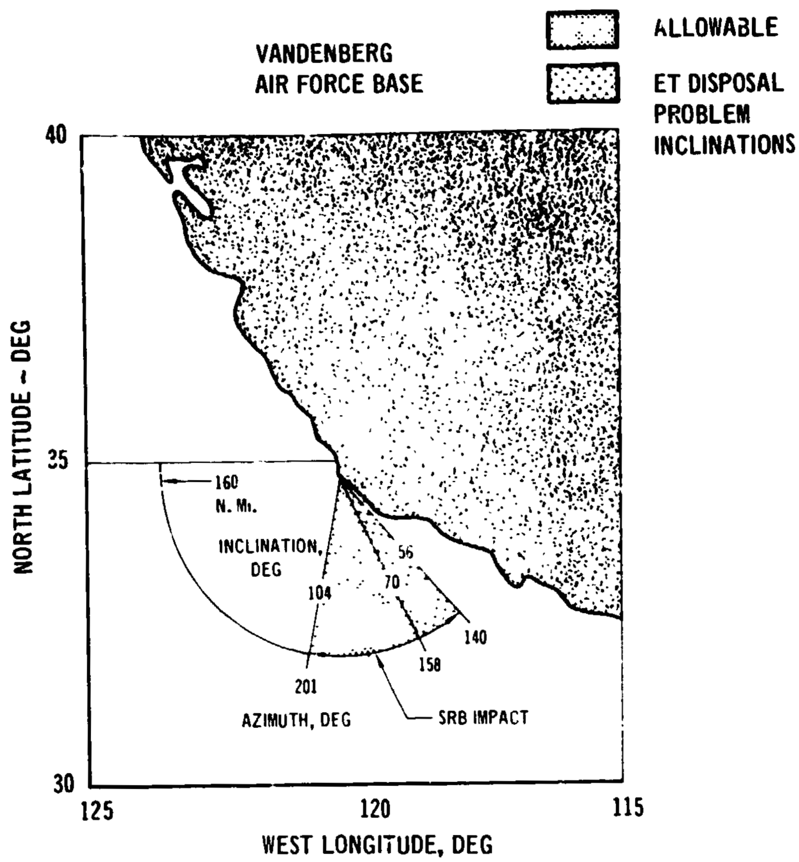
The VAFB launch constraints limit orbit inclinations between 56 and 104 degrees. The lower bound is established from range safety requirements, the upper bound by payload weight limitations due to retrograde launch. The following generalizations apply to the VAFB launch:

(1) Shuttle will be approaching the desired impact areas at Kwajalein, Hawaii, or Poker Flat from either the north or south depending on the orbit number. These are not the traditional approach corridors to Kwajalein and are also studied in Section 5.

(2) Because of the high orbit inclinations, overflight of Poker Flat will be achievable. This was not the case for the KSC launch.

The cargo weight as a function of orbit inclination for both KSC and VAFB launches is given in Figure 13. A nominal launch with one or no OMS kits for added total impulse achieves orbit altitudes between 100 to 300 NMI, and a wide range of payloads. Throughout this study, a 160 NMI circular orbit altitude was assumed as a baseline. (This is the nominal Shuttle orbit altitude defined by NASA.) This results in cargo weight capability as large as 70,000 lb for a due east launch from KSC (28.5 deg inclination) to as low as 37,000 lb for a due south launch from

LAUNCH AZIMUTH AND INCLINATION LIMITS FROM VAFB AND KSC



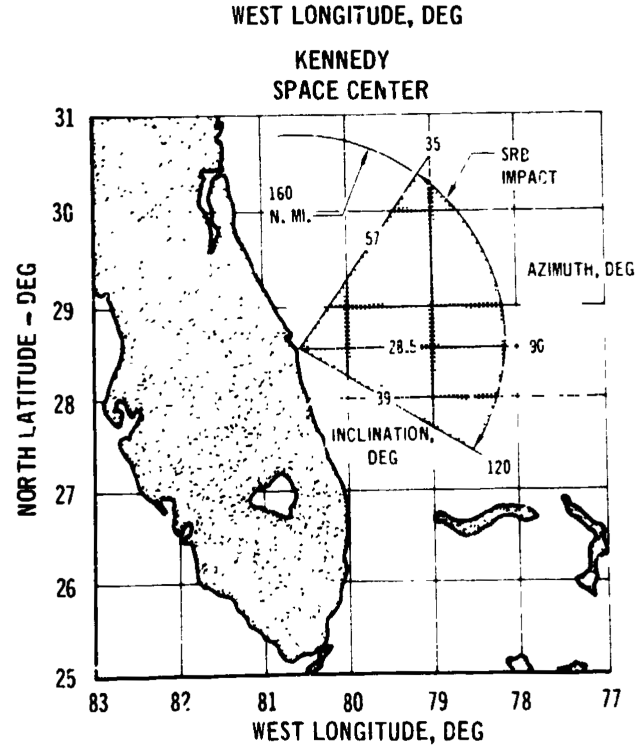


FIGURE 12

CARGO WEIGHT VERSUS INCLINATION FOR VARIOUS CIRCULAR ORBITAL ALTITUDE – DELIVERY ONLY

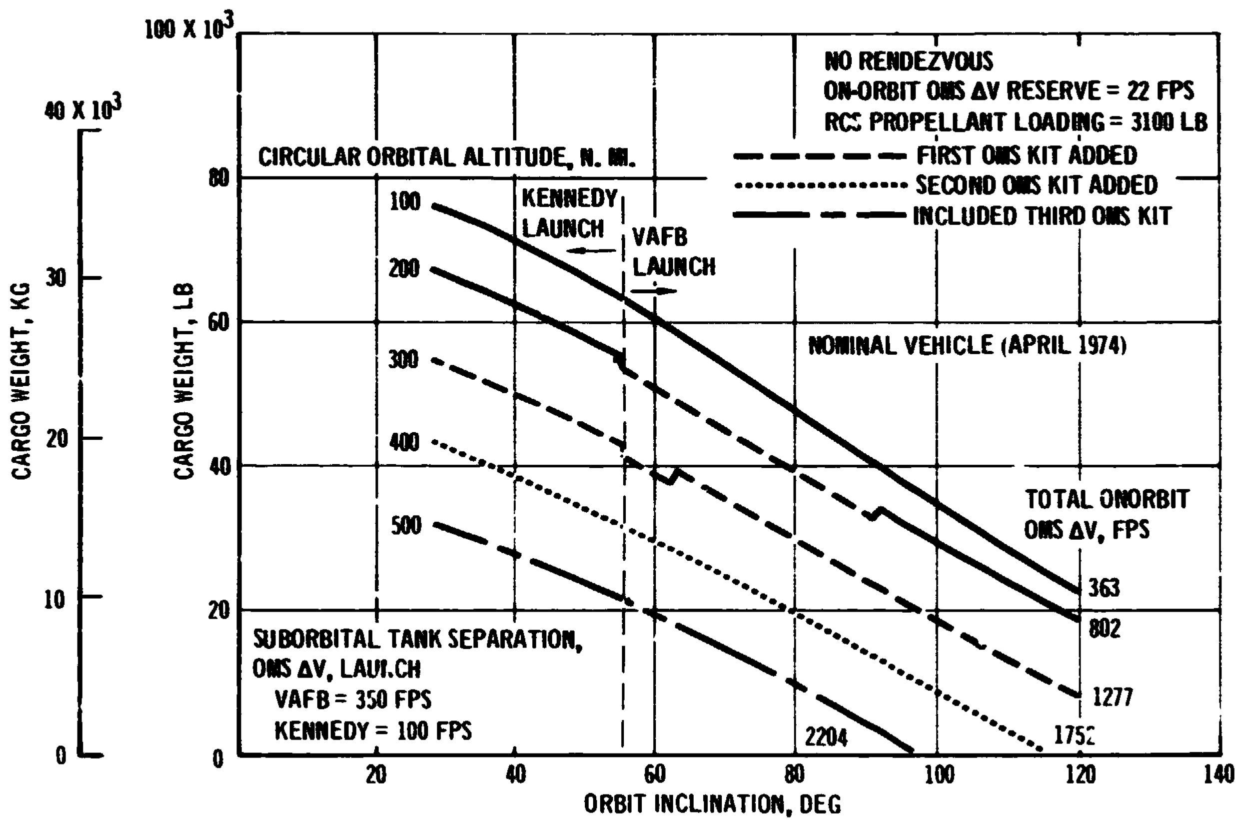


FIGURE 13

VAFB (90 deg inclination). The selection of the nominal Shuttle orbit altitude of 160 NMI as a baseline, allows this study to be compatible with the majority of other Shuttle mission plans and thus, enhances the possibility of sharing the payload bay with other payloads. Consideration of other Shuttle orbit altitudes may be warranted if payload deployment system plane change ΔV requirements become excessive for future specific missions.

5. EXPANDED CAPABILITIES OF KWAJALEIN MISSILE RANGE

For most DoD reentry measurement experiments, complete radar coverage from reentry to impact is required. The reentry measurements radar support required to KMR is principally provided by the major sensors of the KREMS (Kiernan REentry Measurements Site) complex located on Roi-Namur, an island within the Kwajalein Atoll - situated some 45 miles north of Kwajalein Island (See Figure 14). Specifically these radars are referred to as ALTAIR (ARPA-Long Range Tracking and Instrumentation Radar), ALCOR (ARPA-Lincoln C-band Observables Radar), and TRADEX (Target Resolution and Discrimination EXperiment). These radars and all other instrumentation at KMR are discussed in Reference (5).

Traditionally, payloads are launched from VAFB to impact near Roi-Namur as indicated in Figure 14. This approach azimuth provides full coverage by the KREMS radars. Because of the Shuttle launch constraints, described in the previous section, payload approach to KMR at this traditional azimuth is not feasible. In fact, the approach corridor is from the southwest or northwest instead of the northeast.

Reference (5) defines the azimuth zones for which one or more of the KREMS radars cannot operate at all elevations. These zones are referred to as inhibit zones. Generally, they result because the electromagnetic radiation from a radar is incident upon inhabited dwellings (or in some cases aircraft) at low elevations. The inhibit zones for the three KREMS radars are superimposed on a map of Roi-Namur on Figure 15. Four corridors remain feasible for complete radar coverage. Note that the northwest corridor is only open if aircraft operation is restricted. The northeast corridor provides for the traditional approach from VAFB. The southwest corridor is the most readily achieved with a Shuttle launched payload. An approach to KMR through the southwest corridor provides complete KREMS radar coverage of the payload to impact. A typical impact point location and the southwest corridor are shown on Figure 14.

To minimize payload deployment system plane change maneuvers to reenter through the southwest corridor, maximum inclination Shuttle launches from KSC are required. Therefore, further investigations continued to determine if the various inhibit zones, especially on the western approaches to Roi Namur could be bypassed. As a result of a thorough study initiated by Ray Holland of MIT/LL, a letter, Reference (6), was received which stated that all inhibit zones could be bypassed with appropriate pre-test preparation. However, two qualifications remained for target elevations less than 6 deg and ALCOR and TRADEX tracking to the west. The ALCOR

LOCATION OF IMPACT POINT FOR SOUTHWEST APPROACH TO KWAJALEIN

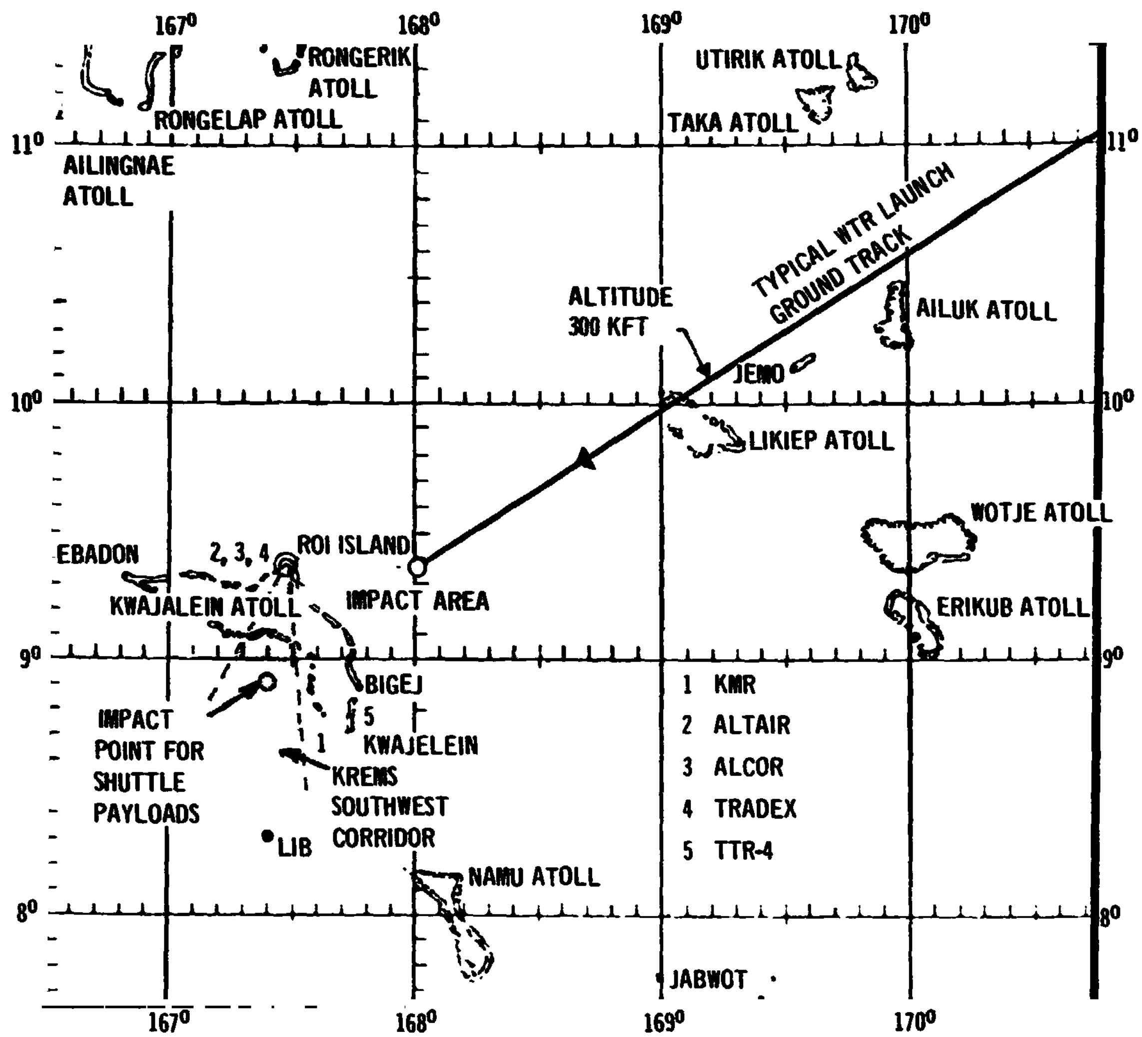
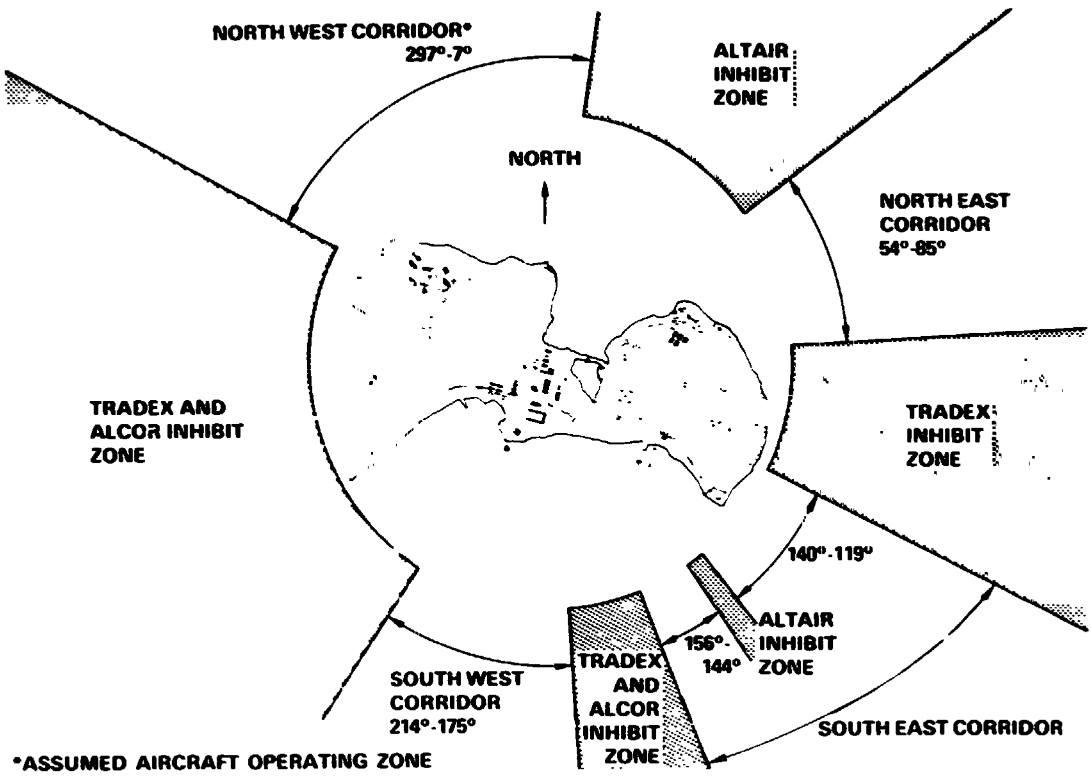


FIGURE 14

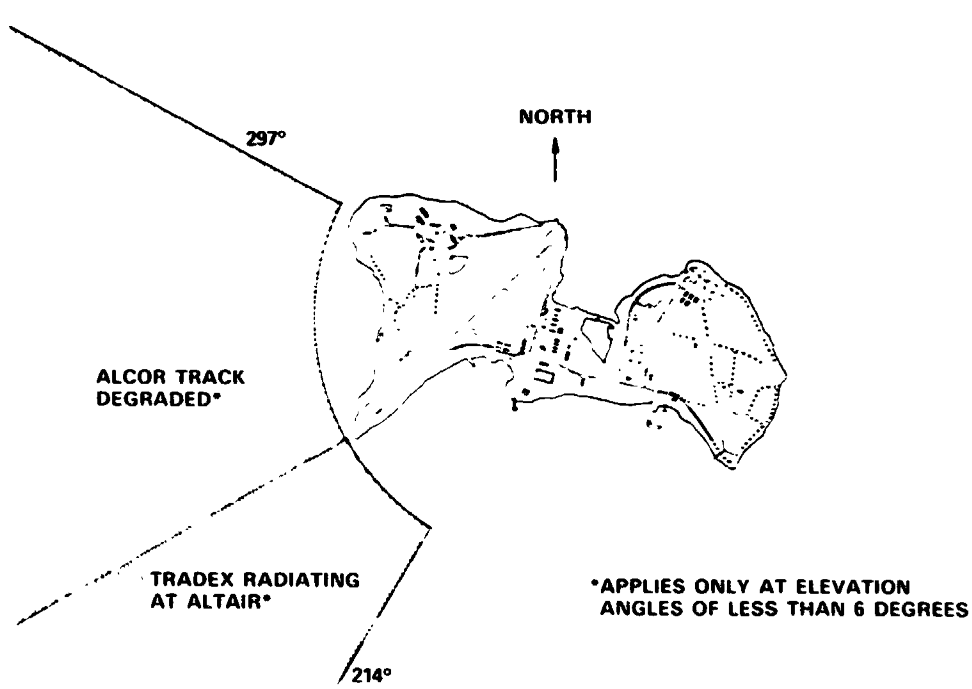
APPROACH CORRIDORS TO KREMS RADARS



*ASSUMED AIRCRAFT OPERATING ZONE RESTRICTION BYPASSED.

FIGURE 15

BYPASSING KREMS INHIBITED ZONES EXPANDS APPROACH CORRIDOR



*APPLIES ONLY AT ELEVATION ANGLES OF LESS THAN 6 DEGREES

FIGURE 16

tracking is degraded for westward approaches because ALCOR is radiating through some buildings. For west-south west approaches the TRADEX is radiating directly at ALTAIR. Although this is not known to be a problem, a detailed study of the geometry would be required to assess electromagnetic interference effects. With all inhibit zones bypassed, the approach corridors to Roi-Namur are as shown in Figure 16. Unfortunately, the western corridors remain inhibited for elevations less than 6 degrees. This means, for example, that for an impact 50 NMI downrange of the radar, track would have to be terminated at an altitude of 65 kft.

For the remainder of this study, it was assumed that the only viable approach corridor from the southwest was between azimuths of 175 to 214 deg in the southwest corridor. As a consequence, continuous coverage by the KREMS radars is guaranteed. In addition, through interface meetings with SAFSCOM, it was determined that no problems exist in relocating telemetry receivers, optical trackers, and other ground based equipment to provide full instrumentation coverage of reentries through the southwest corridor.

During this study, two impact points were used at Kwajalein. These are compared below:

Impact Point	Alt (ft)	Lat (deg)	Long (deg)
Original	0.0	9.36N	168E
Final	0.0	8.90N	167.40E

Both impact points are shown in Figure 14. The original impact point corresponds to the typical impact point for a ground launched payload from VAFB. It applies to the plane change and ground track data given in Section 7. The final impact point represents a point well within the KREMS southwest corridor and provides a radar range and aspect angle comparable to the northeast corridor approach. Section 16 data for cases 1, 2 and 3 were generated with the final Kwajalein impact point.

Case 6 of Section 16 required impact near the Site Defense Radar on Meck Island. The location used for the Site Defense Radar was 9°11' North latitude, 167°43' 32.5" West longitude. Because it is a phased array radar, its radar coverage is limited to north-northeast look azimuths between 345 and 75 deg azimuth. Therefore, only approaches from the north were considered for case 6.

6.0 POKER FLAT IMPACT POINT

A range users manual, Reference (7), for Poker Flat was obtained and provided information to define the Poker Flat impact point. The location of the Poker Flat range is about 20 miles northeast of Fairbanks. Figure 17 shows this location on a map of Alaska. Poker Flat is one of the many auroral field stations in Alaska and serves as a scientific rocket range for the Geophysical Institute of the University of Alaska. The Alaskan pipeline passes between Poker Flat and Fairbanks and could represent a range safety consideration for deorbiting payloads into the area. A more detailed area map is shown in Figure 18 which locates Fairbanks and Poker Flat more exactly. Figures 19 and 20 show the flight zones which are used to impact research sounding rockets at present. These zones provide impact areas 165 NMI in length. The impact point which has been selected for example cases 3 and 4 in Section 16 is marked on Figure 19.

During this study, two impact points were used at Poker Flat. These are compared below:

Impact Point	Alt (ft)	Lat. (deg)	Long. (deg)
Original Poker Flat	0.0	64.5N	144W
Final Poker Flat	0.0	67.0N	146W

The original impact point was used prior to receiving Reference (7). It applies to the plane change and ground track data given in Section 7. Section 16 data was generated with the final Poker Flat impact point. No restriction on approach azimuth was assumed for either impact point.

ALASKAN AURORAL FIELD STATIONS

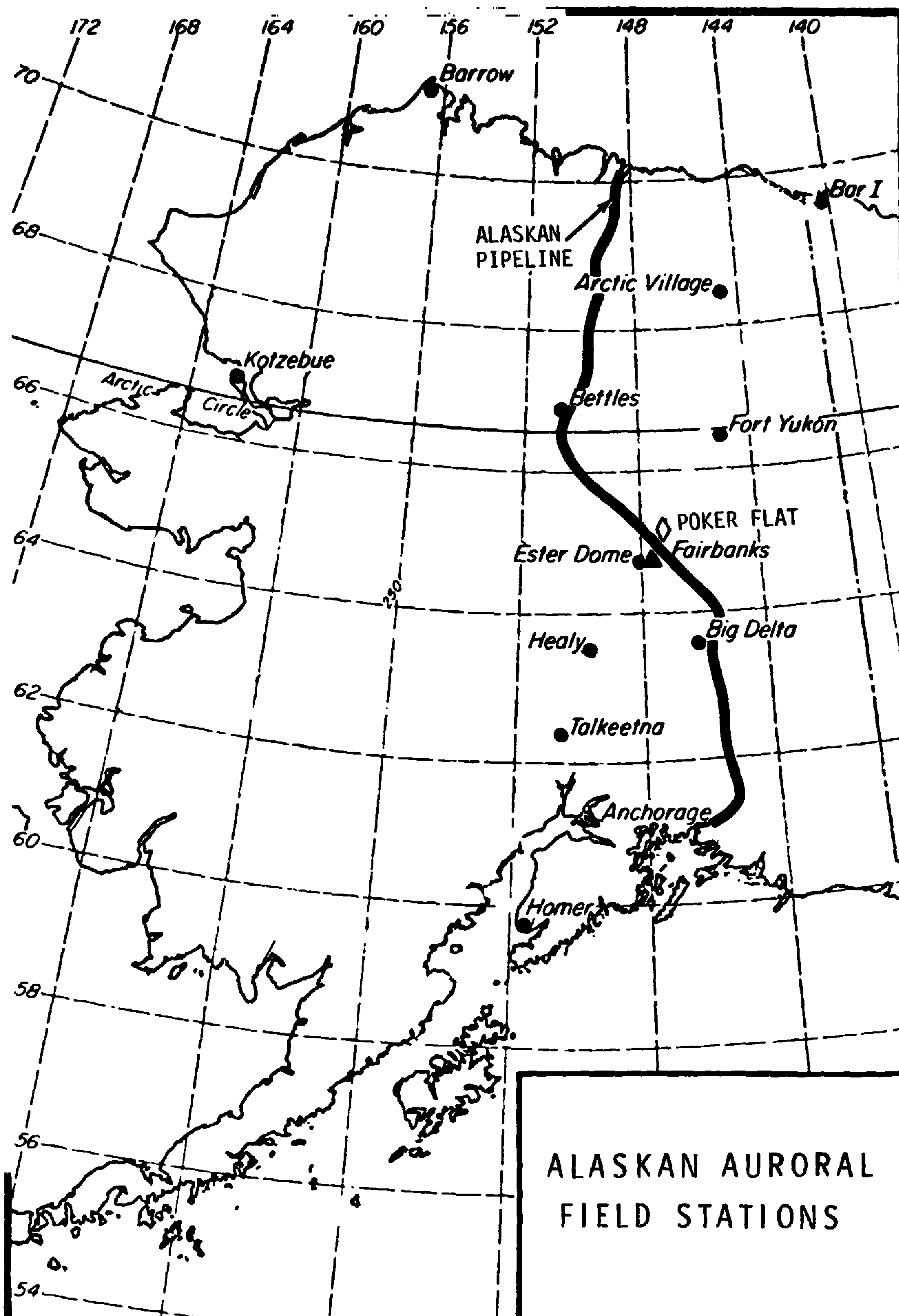


FIGURE 17

POKER FLAT – FAIRBANKS MAP

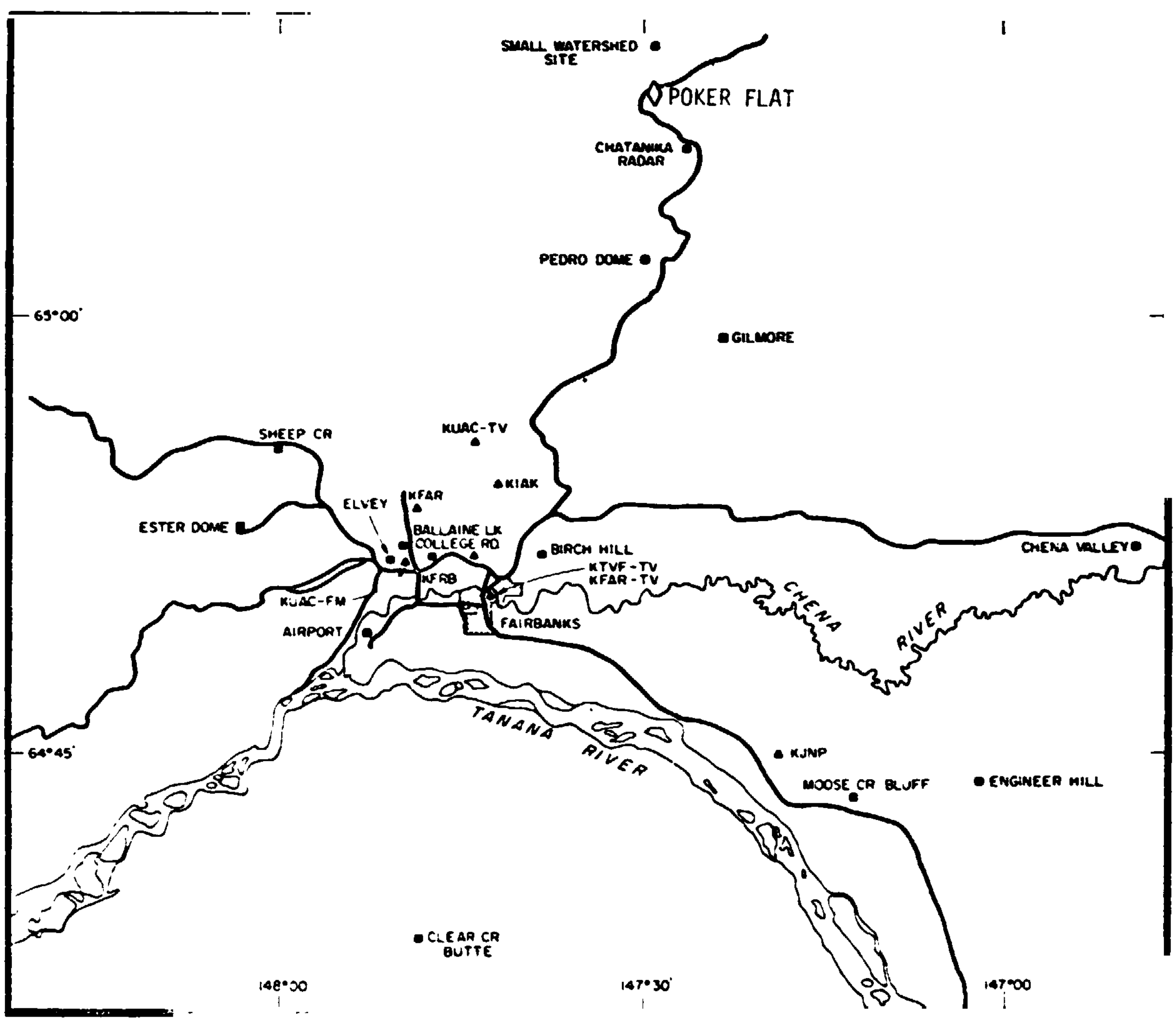


FIGURE 18

POKER FLAT FLIGHT ZONES 2, 3 & 4

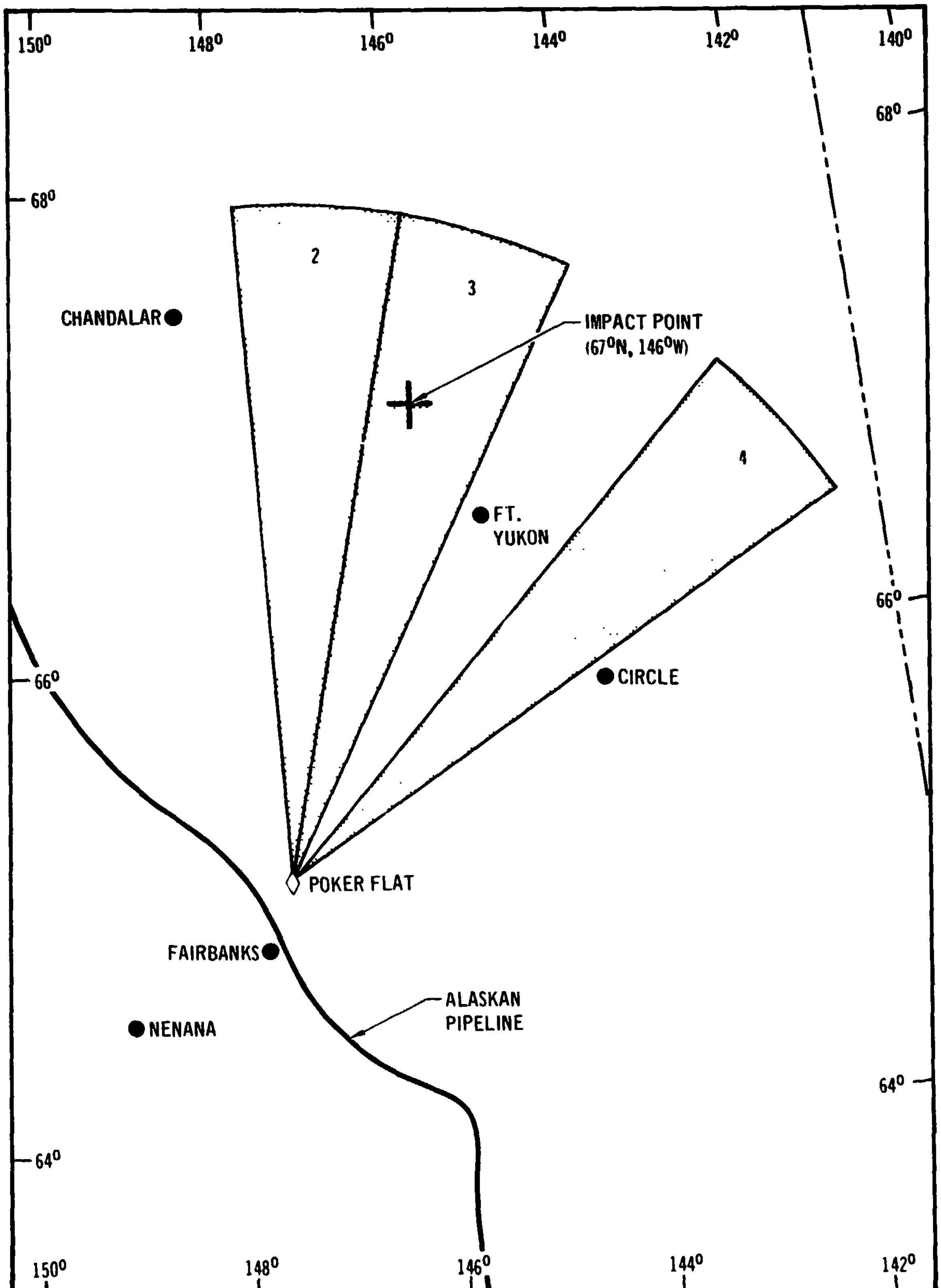


FIGURE 19

POKER FLAT FLIGHT ZONES 1 & 5

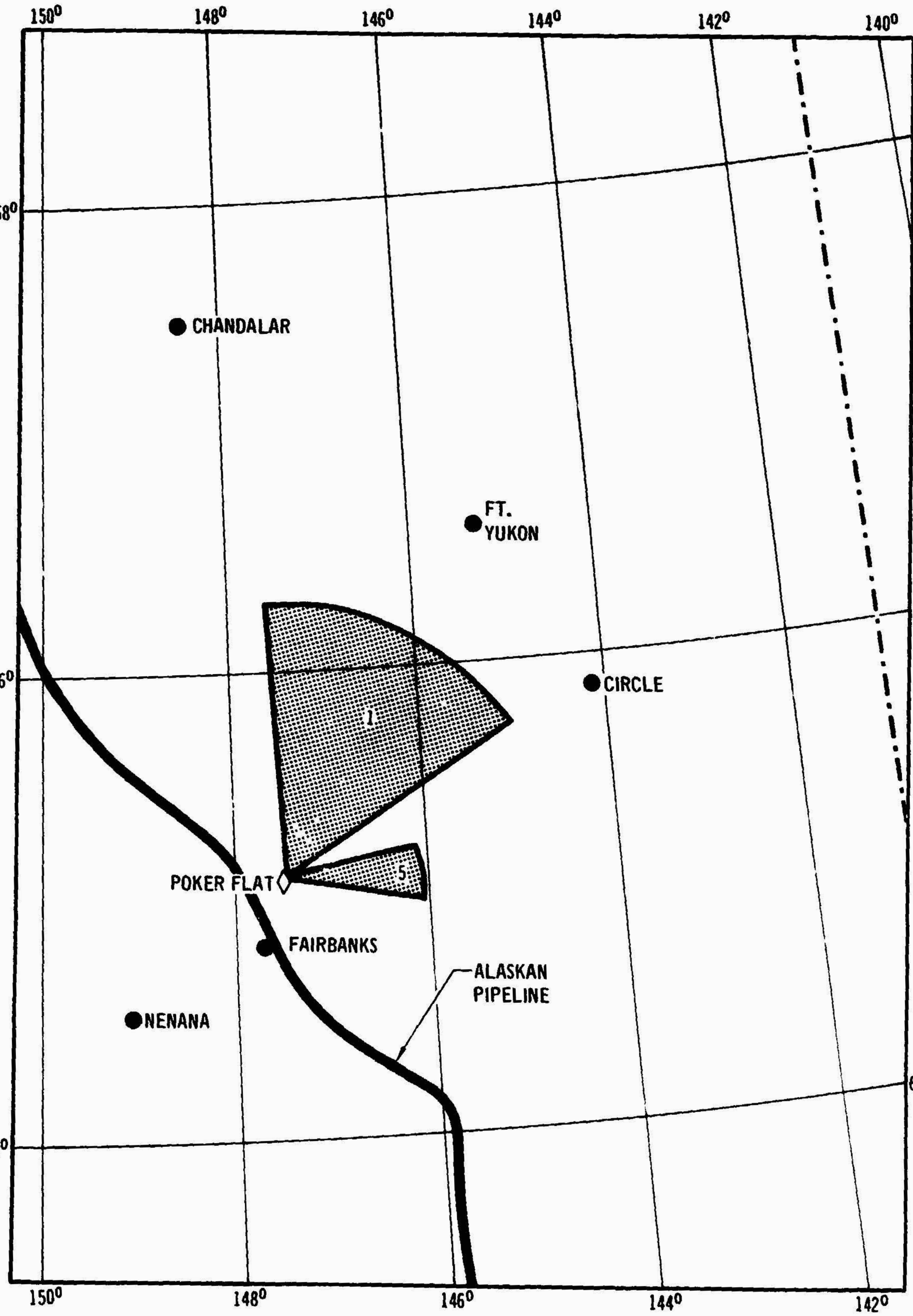


FIGURE 20

7. DELIVERY STRATEGIES

Several delivery strategies were developed to investigate the trajectory shaping required to place multiple Shuttle payloads on a ballistic reentry trajectory to a precise impact point. These strategies and results are discussed in Sections 7.1 through 7.4. The results are used to define delivery system performance in Section 9 and to establish mission scenarios in Section 16.

Several ground rules were established to allow parametric consideration of delivery strategies. These were:

1. Impulsive ΔV Analysis - Velocity increments, ΔV, required for maneuvers were assumed to be applied instantaneously. This assumption is valid for small orbital velocity changes or high thrust/weight ratio boosters. Section 9 discusses the limitations of impulsive ΔV assumptions with respect to typical boosters.
2. Inertial Entry Conditions - Because the number of impact areas and approach azimuths was virtually unlimited during the initial parametrics, inertial reentry conditions were considered to reduce the number of parametrics. These results are applicable to all impact points and approach azimuths if the inertial to relative coordinate transformation is made before applying the data.
3. Spherical Earth - For the purpose of parametric analysis, a spherical earth with uniform gravitational field was assumed. More sophisticated earth models are not warranted except for very detailed mission planning which is not addressed in this report.

Given these ground rules, an approach to developing payload delivery strategies was developed. A typical DoD experiment mission, Figure 21, may require three classes of payload deployment system (PDS) burns. These are:

1. Deorbit Burn - This is an inplane burn required to place the PDS on a ballistic trajectory to impact.
2. Plane Change Burn - This burn changes the payload azimuth to insure impact at the required location and azimuth. It is required in general because the Shuttle orbit does not necessarily provide the exact launch point and azimuth at PDS deployment to hit the impact point.
3. RV Spacing Burn - In order to space the multiple payloads at the pierce point, each payload is placed on a slightly different trajectory by applying a ΔV to each payload. This is done after the PDS is on the ballistic trajectory.

TYPICAL DOD EXPERIMENT MISSION

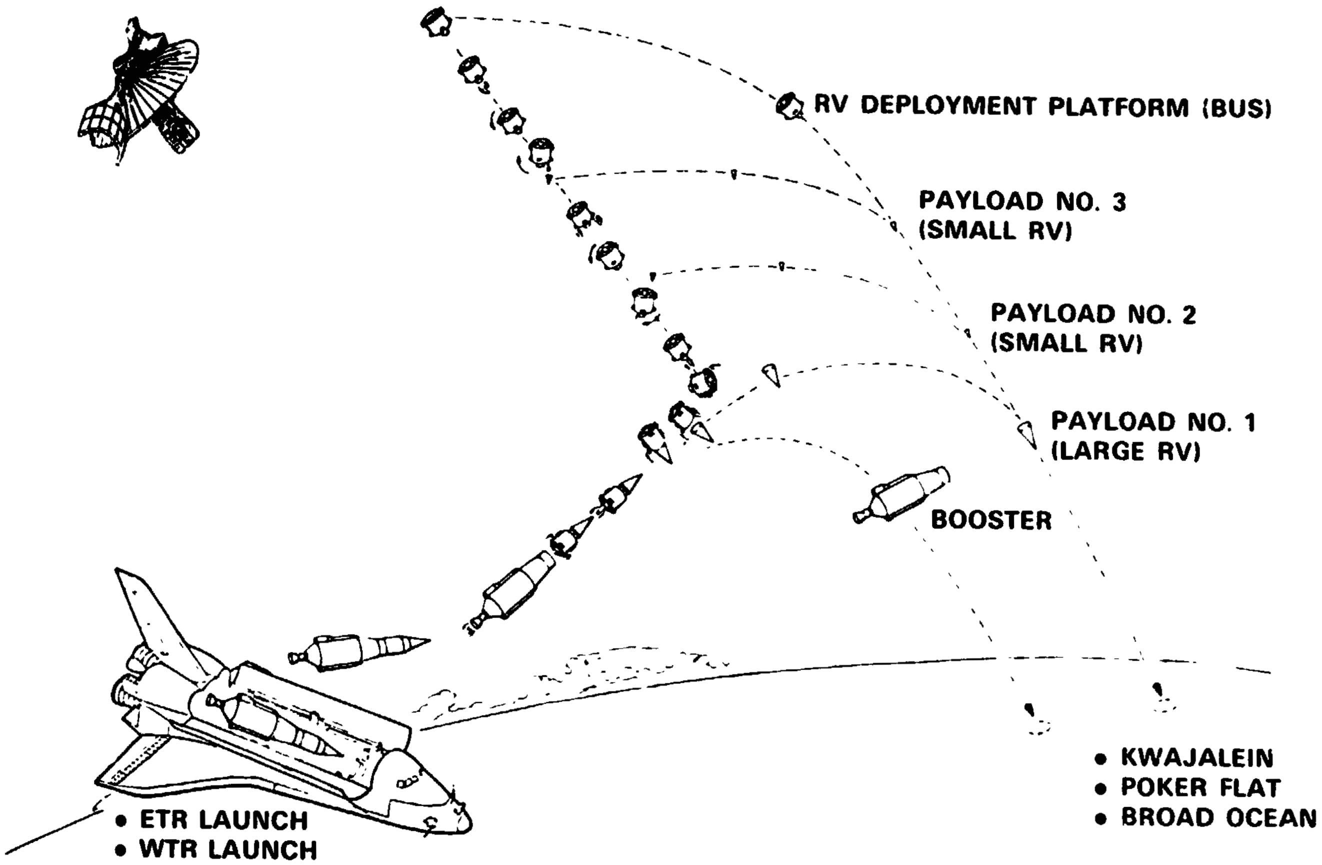


FIGURE 21

Each of these three classes of PDS burns has several options to achieve the desired result. In this study, the three classes of burns were considered separately. For specific missions, such as those studied in Section 16, the results are combined to establish Transtage requirements. The following sections discuss each of the three PDS burns and the combination of them to define a mission sequence.

7.1 Deorbit - The deorbit (inplane) burn strategies evolved by considering a typical ground launch trajectory as shown in Figure 22. Note that the Shuttle orbit altitude is low compared to the RV apogee altitude. This allows the three distinct deorbit strategies illustrated in Figure 23 to be considered.

A. Direct Upleg Insertion - A PDS burn is made from the Shuttle orbit, and places the PDS on the ascending leg or upleg of the ballistic trajectory. This provides maximum trajectory simulation and requires slightly less deorbit time than a ground launch. However, a large velocity change is required if the ballistic entry conditions require low velocity or steep flight path angle.

B. Direct Downleg Insertion - A considerable reduction in response time can be achieved by performing the PDS burn on the descending leg or downleg of the ballistic trajectory. For a typical 160 NMI shuttle orbit altitude, the time to payload pierce is between 0.5 to 7 min depending upon the desired reentry velocity but the impulsive ΔV requirements for this maneuver are the same as strategy A. Therefore, to complete the burn before pierce, a high thrust level booster is required. In addition, payload reentry time spacing at the pierce point requires large ΔV spacing burns.

C. Hohmann to Apogee Insertion - Impulsive ΔV requirements can be minimized by performing a Hohmann transfer burn of the PDS 180 degrees away from the ballistic trajectory apogee. At apogee, a second burn is applied to deorbit the PDS. The combined ΔV for these two burns is much less than that for strategies A and B. However, the total mission time becomes quite long because of the Hohmann transfer orbit.

A good approximation to the deorbit ΔV requirements for strategy A and B can be obtained by assuming that the velocity and flight path angle at pierce are equal to those at the Shuttle orbit altitude. Then the deorbit ΔV requirements are given by the geometric relation:

TYPICAL RV LAUNCH TRAJECTORY

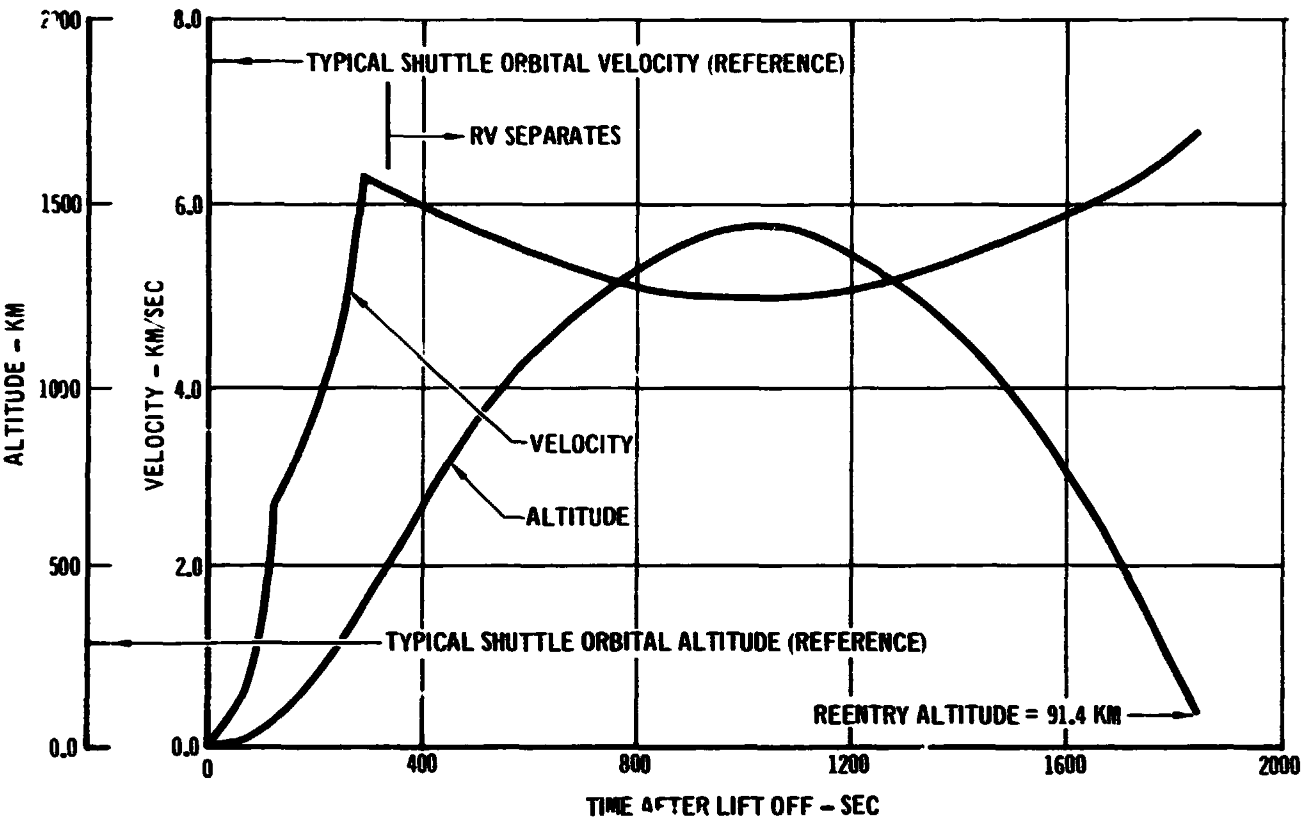
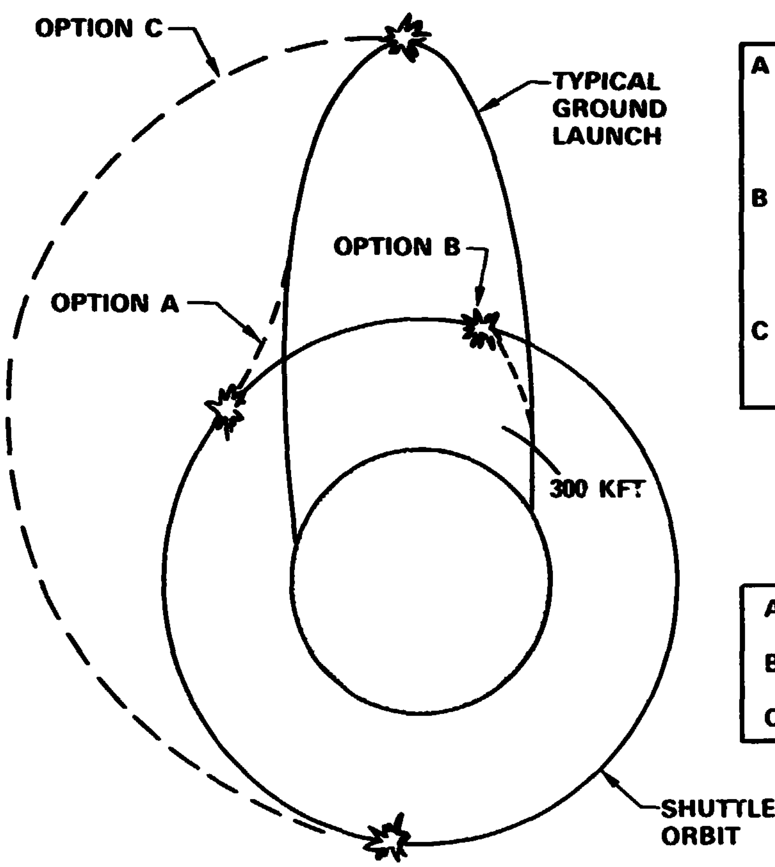


FIGURE 22

DEORBIT MANEUVER STRATEGY OPTIONS



(GEOMETRY DISTORTED FOR CLARITY)

OPTION DESCRIPTION

A DIRECT UPLEG INSERTION
- (+) MAX TRAJECTORY SIMULATION
- (+) SHORT RESPONSE
- (−) HIGH ΔV

B DIRECT DOWNLEG INSERTION
- (+) VERY SHORT RESPONSE
- (−) HIGH ΔV AND THRUST
- (−) BOOSTER SEPARATION DIFFICULT

C HOHMANN TO APOGEE INSERTION
- (+) MINIMUM ΔV
- (−) LONG RESPONSE

	ΔV (KFT/SEC)	RANGE (NMI)	TIME TO PIERCE (MIN)
A	2-24	1,500-7,000	8-60
B	2-24	60-1,550	0.5-7
C	1.5-15	12,000-15,000	45-100

FIGURE 23

$$V = \sqrt{V_0^2 + V_E^2 - 2V_0V_E \cos \gamma_E} \qquad \text{Eq. 1}$$

where V_0 = Shuttle orbital velocity

V_E = Deorbit velocity

γ_E = Deorbit flight path angle

A plot of this equation for flight path angles of interest is given in Figure 24. For low reentry speeds or steep flight path angles the ΔV is large. On the other hand, high reentry speeds and shallow flight path angles required low ΔV. These reentry conditions require little change from the Shuttle orbit conditions. Therefore, the most promising range of reentry conditions for strategy A or B is in the 20-25 kfps velocity range and shallow flight path angles between -5 to -15 deg.

The results of a more rigorous analysis of strategy A, B and C deorbit ΔV requirements are shown in Figures 25 and 26. For these cases, the impulsive ΔV is computed as described in Appendices A and B. This analysis is an exact computation of the impulsive ΔV requirements.

A comparison of Figures 25 and 26 indicate the significant reduction in total deorbit ΔV achievable through use of the Hohmann transfer Strategy C at steep reentry angles. For instance, the ΔV requirements for reentry at 25 kft/sec, -40 deg flight path angle are 17 and 8 kfps for strategy A and C respectively.

In conclusion, strategy C impulsive ΔV requirements are always less than that of Strategy A. The difference is particularly important at steep flight path angles where the strategy A requirements become excessive.

The ΔV requirements principally affect booster sizing. Other parameters such as deorbit times and range to impact affect mission planning and coordination. The deorbit times associated with each strategy are given in Figures 27 through 29. The deorbit time is defined as the time elapsed from the first burn initiation at Shuttle orbit altitude until the payload reenters at the pierce point. Appendices A and B describe the computation technique for the various strategies.

Strategy A deorbit times in Figure 27 are slightly less than ground launch times. The difference is due to the time it takes a ground launch to attain the Shuttle orbit altitude. Typically, this time is 4-5 minutes. Shallow angle, low velocity reentries require the least time because the Shuttle orbit is near the ballistic trajectory apogee for these conditions. As a consequence, the ascending leg of the trajectory is minimal. In fact, at very shallow flight path angles and low speeds, the deorbit trajectory apogee is below the Shuttle orbit altitude and a single deorbit burn to achieve the required reentry conditions is not possible.

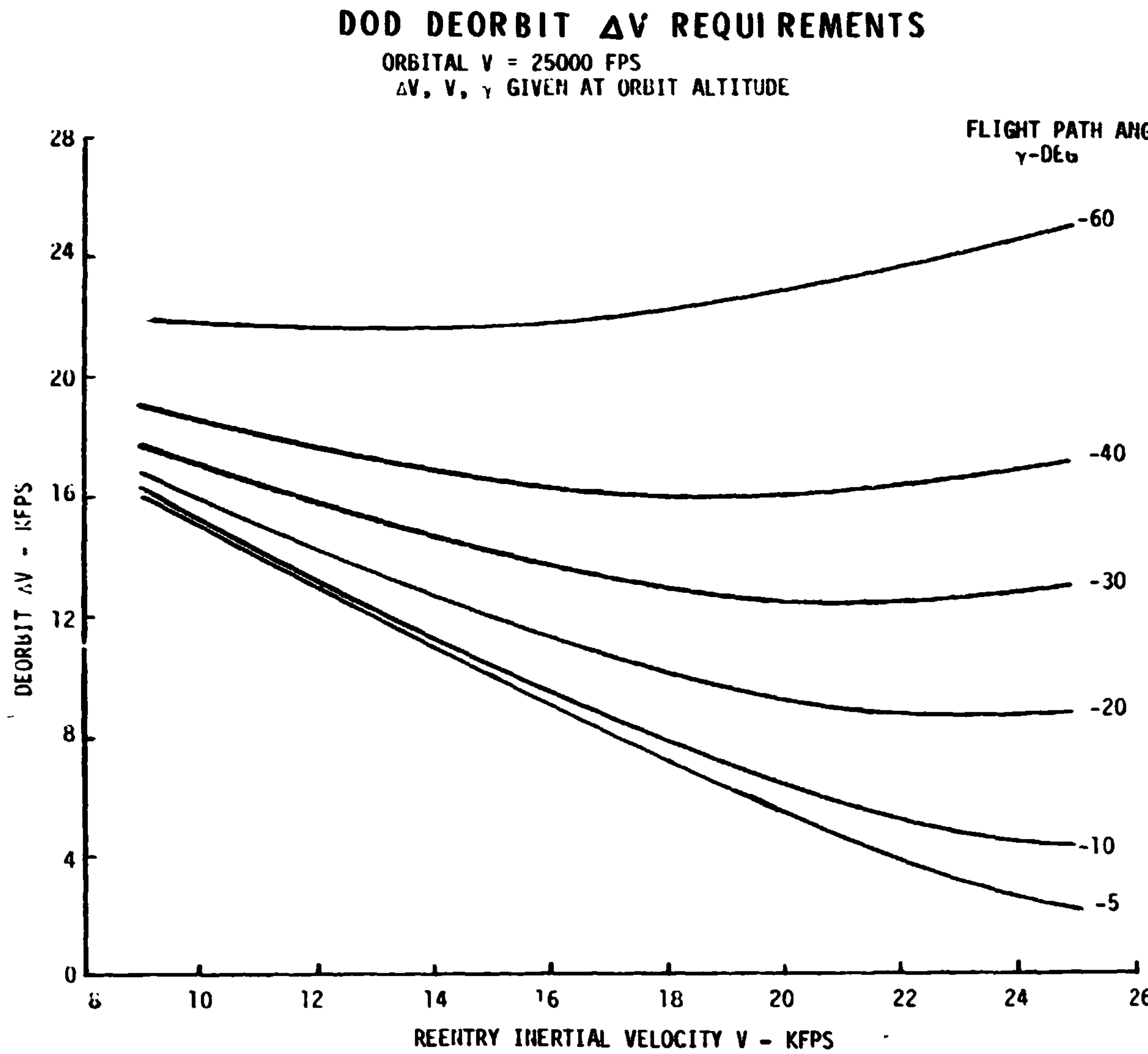

DOD DEORBIT ΔV REQUIREMENTS
ORBITAL V = 25000 FPS
ΔV, V, γ GIVEN AT ORBIT ALTITUDE
FLIGHT PATH ANGLE
γ-DEG
-60
-40
-30
-20
-10
-5
DEORBIT ΔV - KFPS
0
4
8
12
16
20
24
28
8
10
12
14
16
18
20
22
24
26
REENTRY INERTIAL VELOCITY V - KFPS

FIGURE 24

STRATEGY A OR B INPLANE ΔV REQUIREMENTS (UPLEG DEORBIT)

- ΔV'S ARE GREATER THAN 15 KFT/SEC IN THE REQUIRED V-γ RANGE

SHUTTLE ORBIT ALTITUDE – 160 NMI
PIERCE POINT ALTITUDE – 300 KFT

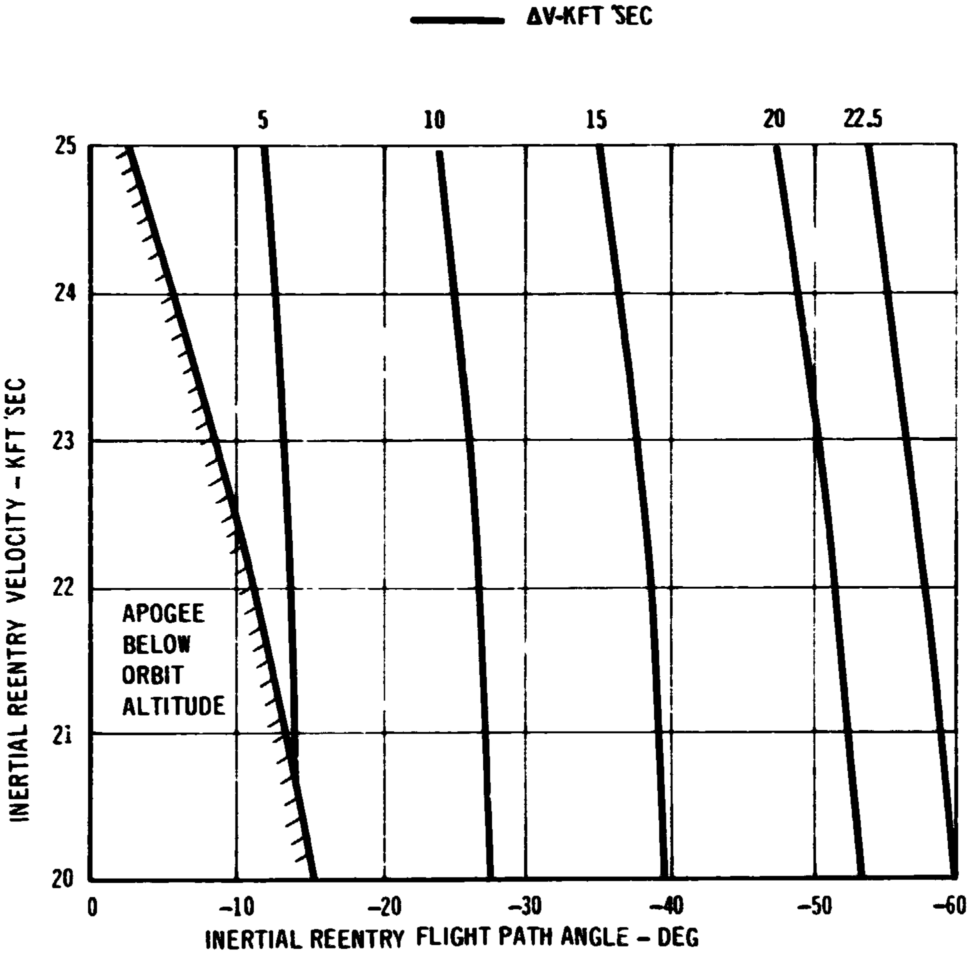


FIGURE 25

STRATEGY C INPLANE ΔV REQUIREMENTS (HOHMANN)

- ΔV'S ARE GREATER THAN 10 KFT/SEC IN THE REQUIRED V-γ RANGE

SHUTTLE ORBIT ALTITUDE – 160 NMI
PIERCE POINT ALTITUDE – 300 KFT

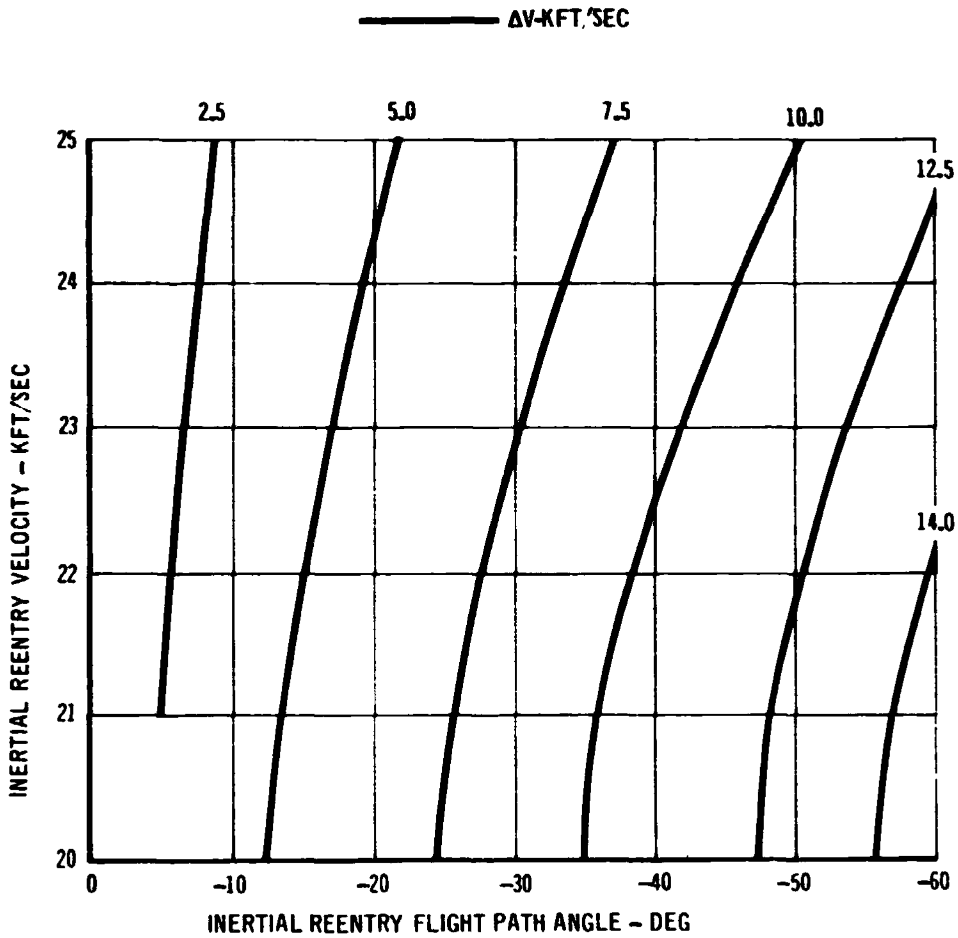


FIGURE 26

UPLEG DEORBIT TIMES (STRATEGY A)

(UPLEG DEORBIT TIMES OF 8 TO 60 MINUTES ARE REQUIRED)

SHUTTLE ORBIT ALTITUDE = 160 NMI
PIERCE POINT ALTITUDE = 300 KFT

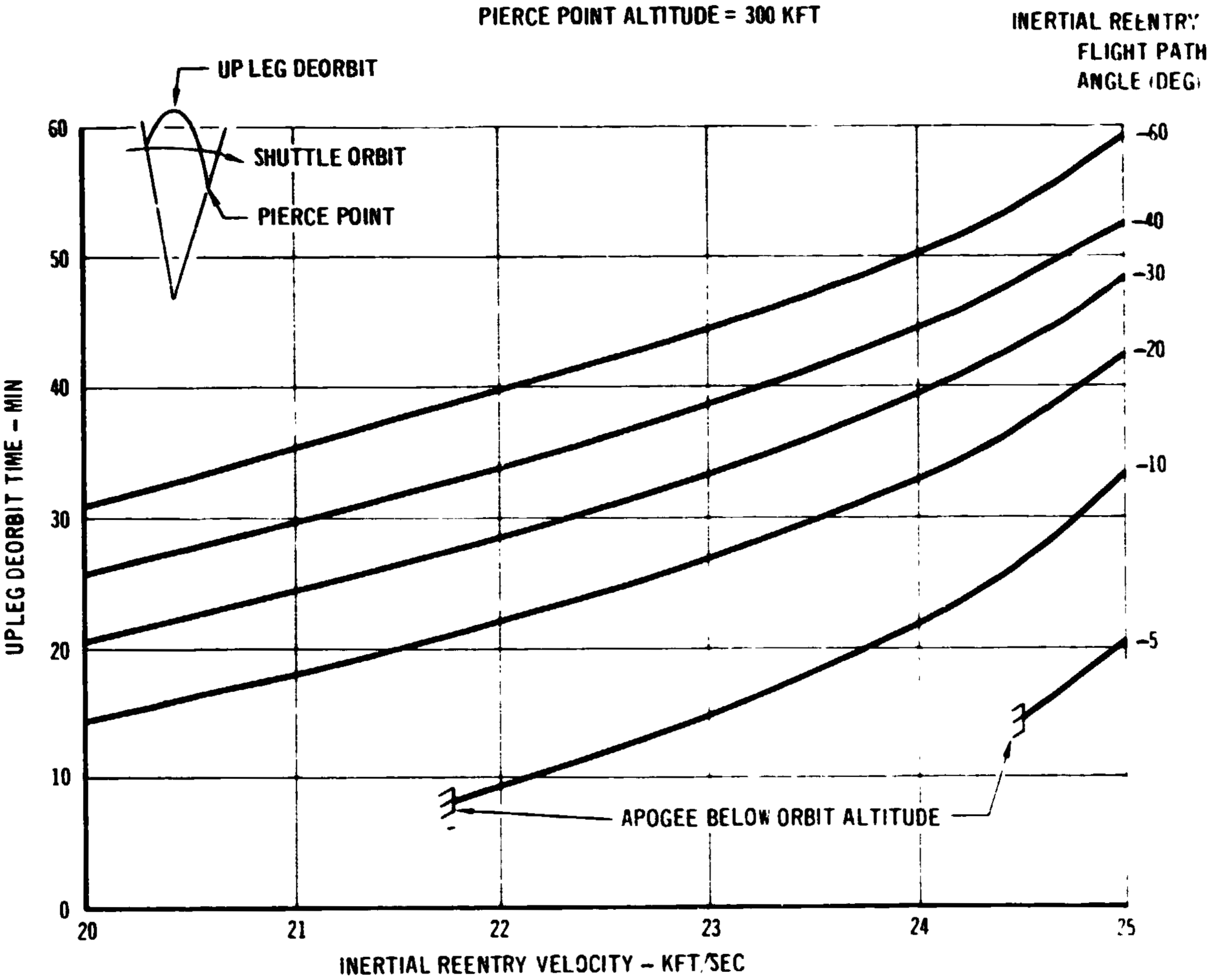


FIGURE 27

DOWNLEG TIMES (STRATEGY B)

(DOWNLEG DEORBIT TIMES OF 30 TO 400 SECONDS ARE REQUIRED)

SHUTTLE ORBIT ALTITUDE = 160 NMI
PIERCE POINT ALTITUDE = 300 KFT

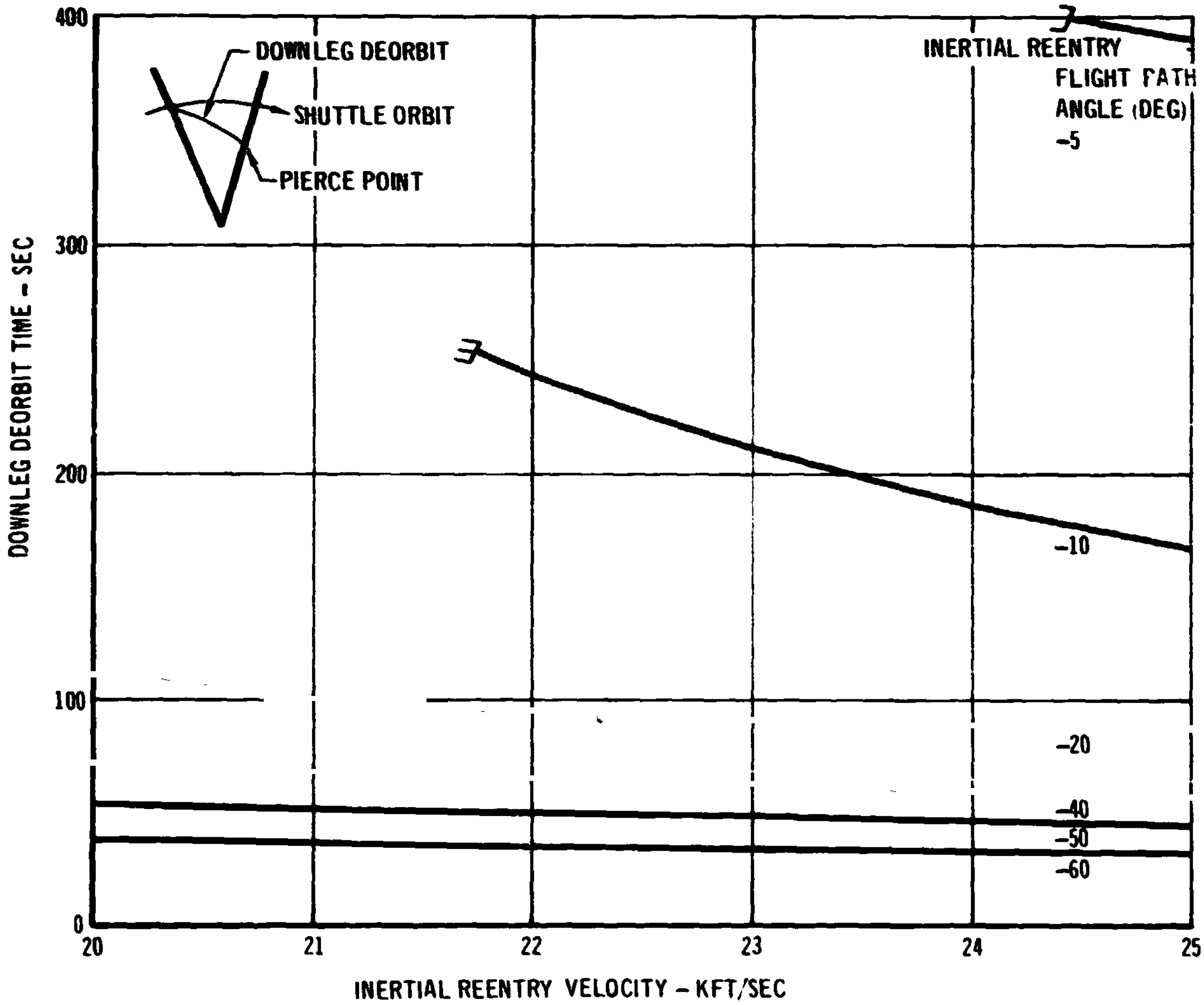


FIGURE 28

HOHMANN TRANSFER TOTAL DEORBIT TIMES (STRATEGY C)

(TIMES OF 45 TO 100 MINUTES ARE REQUIRED)

SHUTTLE ORBIT ALTITUDE = 160 NMI
PIERCE POINT ALTITUDE = 300 KFT

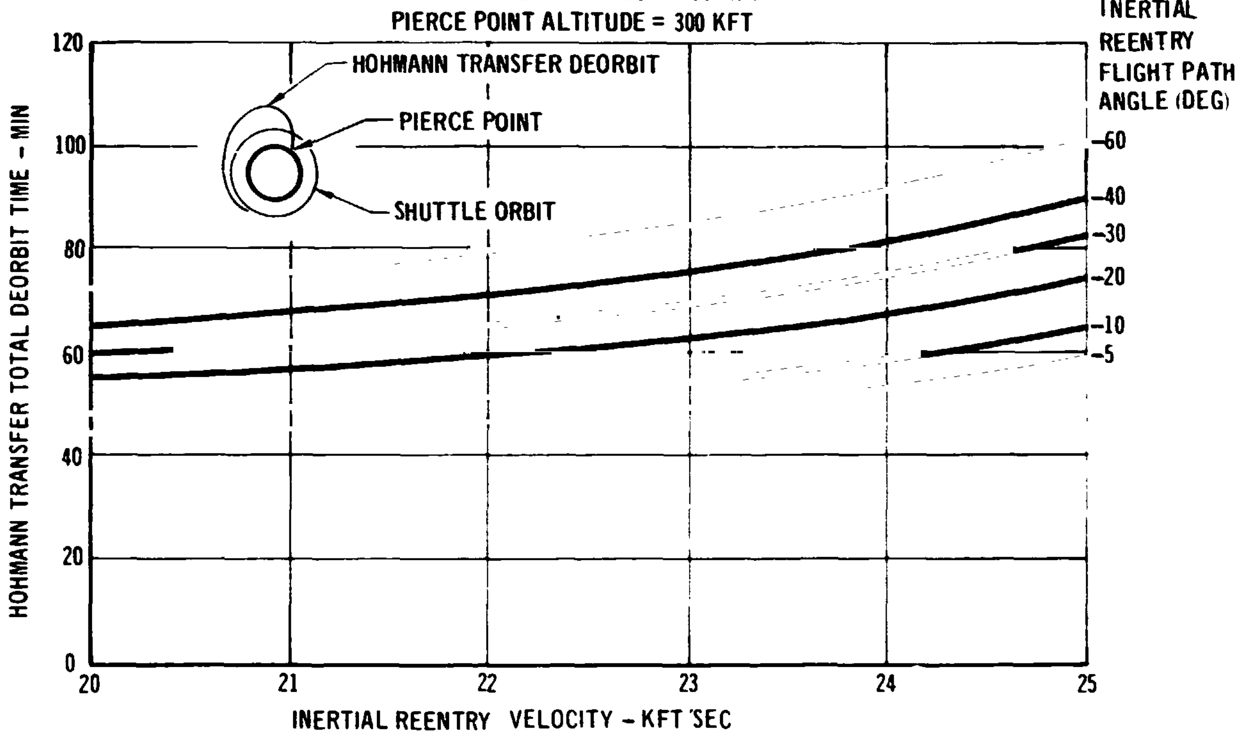


FIGURE 29

Strategy B deorbit times in Figure 28 are much less than strategy A and are typically 30 to 400 seconds over the range of velocities and flight path angles investigated. Minimum times occur at steep reentry angles and high velocities. Because this maneuver puts the payload on the descending leg, the steep and fast reentries reach the pierce altitude quickest.

Strategy C deorbit times in Figure 29 are much longer than strategies A and B because of the Hohmann transfer transit time. Low speed, shallow angle reentries require minimum times because the Hohmann transfer orbit is nearly circular and the apogee is close to the pierce point altitude.

Ground ranges to the pierce point for strategy A and B given in Figures 30 and 31 provide an indication as to where these maneuvers occur in relation to the pierce point. Strategy C,because of the Hohmann transfer,is 12,000 to 15,000 NMI away from pierce and is not shown. For Strategy A and a given reentry velocity, the ground range to pierce increases as flight path angle steepens and then begins to decrease. This occurs because the upleg portion of the trajectory gets longer as flight path angle steepens. Eventually, this effect is compensated for by the shortening of the downleg as the flight path angle steepens even more so.

For the down range deorbit ranges of Figure 31, the longest ranges exist for shallow flight path angles where the trajectories become more circular in shape. At very steep angles, ranges of less than 100 NMI from pierce are achieved.

In summary, Strategies A and B deorbit maneuvers are most attractive for shallow flight path angles at high velocity where the ΔV requirements are not severe. Strategy C provides the capability to achieve steeper flight path angles with lower ΔV than Strategies A and B.

7.2 Plane Change - As a general rule, plane change of the PDS is required to achieve a given latitude, longitude and azimuth at the pierce point. For this study, the plane change is the first burn made by the PDS after deployment from Shuttle. The plane change maneuver changes the PDS orbit inclination from that of the Shuttle to an inclination which will pass through the required deorbit burn point latitude and longitude. The PDS orbit remains circular at the Shuttle orbit altitude. Less energy is required for a plane change maneuver executed at apogee of the deorbit trajectory. However, a plane change before the deorbit burn allows almost full simulation of the deorbit trajectory. In addition, payload spacing burns can be completed prior to apogee. With a PDS plane change at apogee, payload spacing

GROUND RANGE TO PIERCE FOR AN UPLEG DEORBIT

(STRATEGY A)

(UPLEG DEORBIT MANEUVER MUST OCCUR 1500 TO 7000 NMI FROM PIERCE POINT)

SHUTTLE ORBIT ALTITUDE – 160 NMI
PIERCE POINT ALTITUDE – 300 KFT

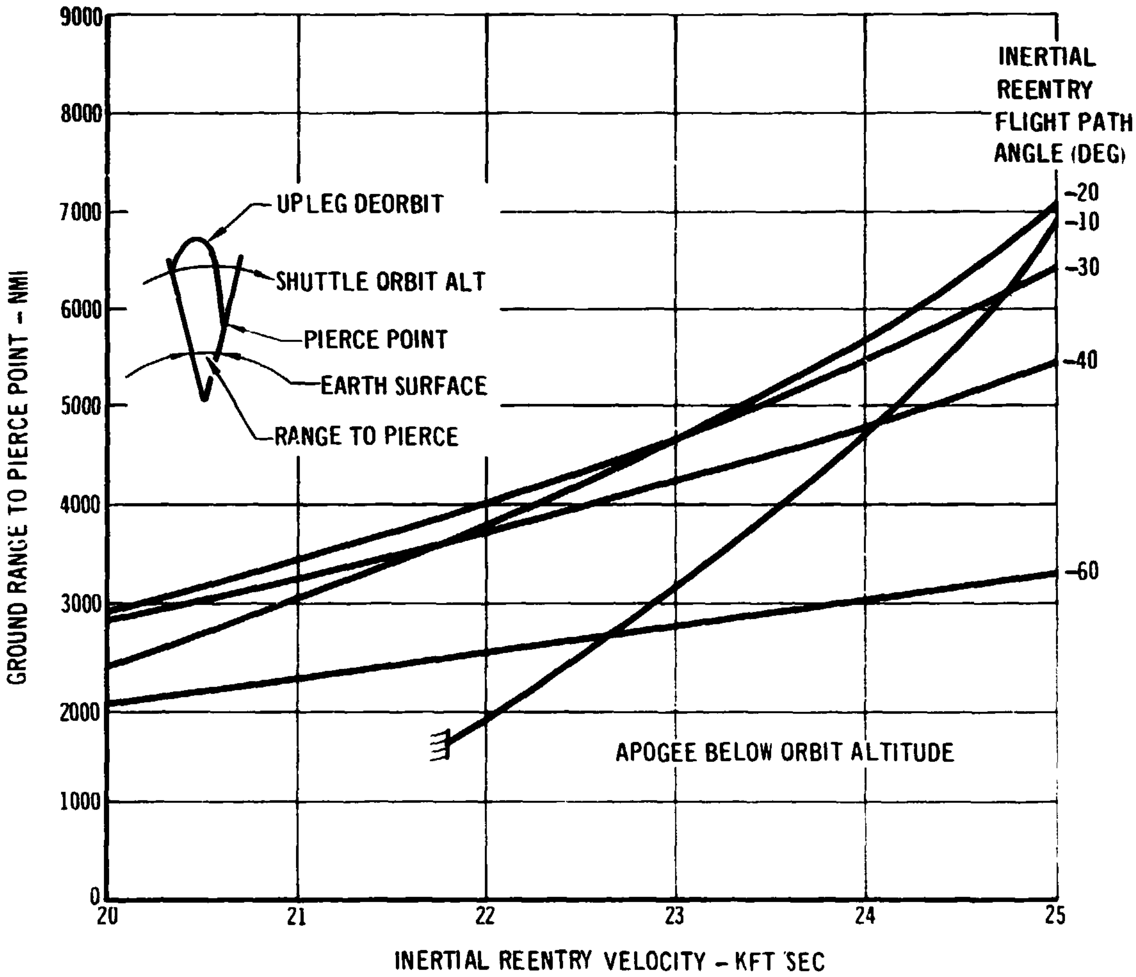


FIGURE 30

GROUND RANGE TO PIERCE POINT FOR A DOWNLEG DEORBIT (STRATEGY B)

(DOWNLEG DEORBIT MANEUVER MUST OCCUR 60 TO 1550 NMI FROM PIERCE POINT)

SHUTTLE ORBIT ALTITUDE –160 NMI
PIERCE POINT ALTITUDE – 300 KFT

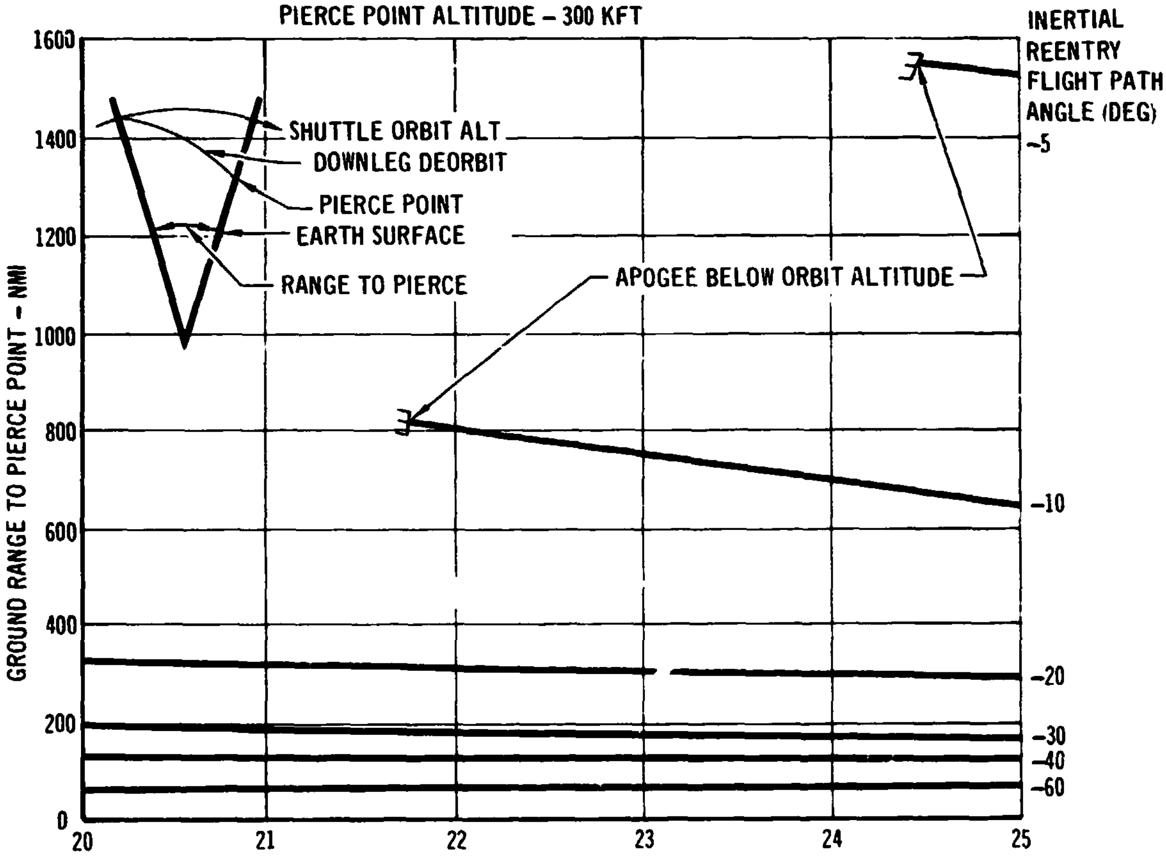


FIGURE 31

must occur on the downleg portion of the trajectory. This increases spacing ΔV requirements and, in some cases, the 90 second spacing at pierce cannot be achieved.

The plane change ΔV determination scheme was first considered for the general case of overflying the target area. This provided some insight into the locations on the Shuttle orbit for which plane change maneuvers might be made. Figure 32 provides a schematic summary of the strategy. For the selected pierce point at KMR a range of allowable approach azimuths can be defined so as not to violate the KREMS inhibit zones. The window for plane change can then be defined on the Shuttle orbit as shown. The heading change and required ΔV in this plane change window can then be determined.

This analysis applied to a KSC launch with pierce at KMR results in ΔV requirements as shown in Figure 33. At a given KSC launch azimuth, there are a range of ΔV's applied at the proper point on the Shuttle trajectory which will allow overflight of KMR through a corridor. As the launch azimuth increases (more southward launch), eventually the overflight through the southwest corridor is not feasible. The results shown favor orbit 4 for a reduced plane change ΔV requirement. This figure also indicates that for minimum ΔV a northern azimuth Shuttle launch is desirable. These results guided the analysis of the example cases described in Section 16.

The example cases will be treated in more detail because the pierce point conditions were fixed. However, three options existed for determining the plane change requirements. These were:

1. Combined Burn - The intercept of the loci of deorbit points and the Shuttle ground track defines a location at which a single burn deorbit can be made with the deorbit maneuver and plane change occurring simultaneously. The solution for these loci proceeds as follows. Given the impact point latitude and longitude and the pierce point altitude, relative velocity, flight path angle, and a selected azimuth, the latitude and longitude of the pierce point must be computed as shown in Figure 34. Figure 35 shows the resulting pierce point latitude and longitude as a function of relative flight path angle at pierce for both Kwajalein and Poker Flat. Note that at the shallow flight path angles, pierce points are significantly displaced in latitude and longitude from the impact point. Once the pierce point location is determined, it serves as the target point to calculate the deorbit burn locations for the selected pierce point azimuths. If a Shuttle orbit track passes through the loci of deorbit burn points, then a single burn plane change and deorbit maneuver is

OUT-OF-PLANE ΔV DETERMINATION SCHEME
KSC LAUNCH

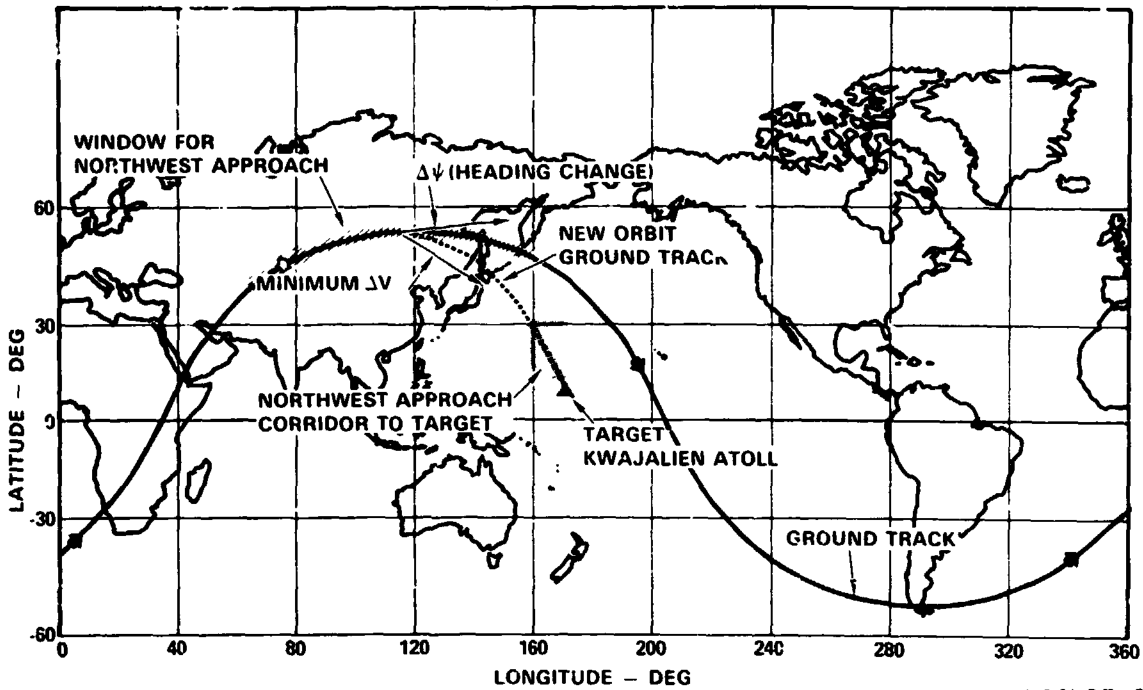


FIGURE 32

OUT-OF-PLANE ΔV REQUIREMENTS FOR SOUTH-WEST APPROACH TO KWAJALEIN

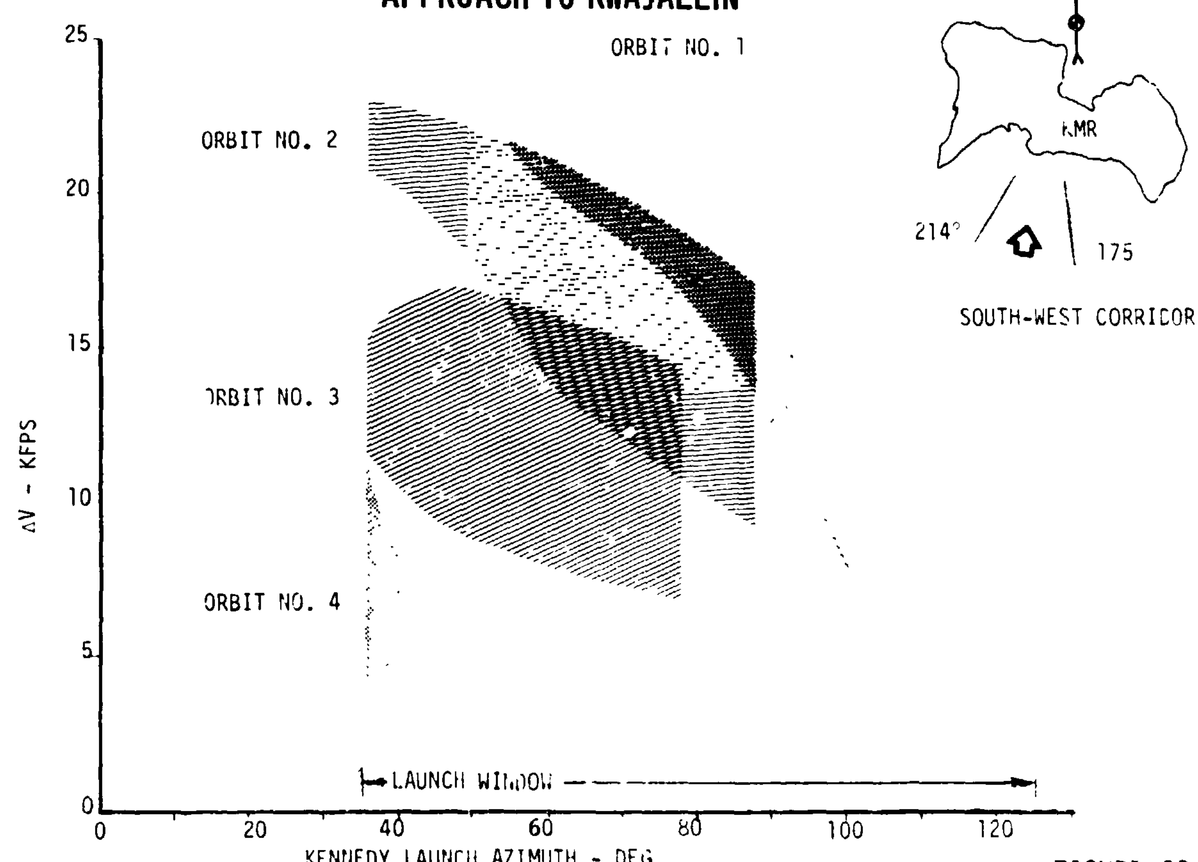


FIGURE 33

PIERCE POINT LATITUDE AND LONGITUDE DETERMINATION

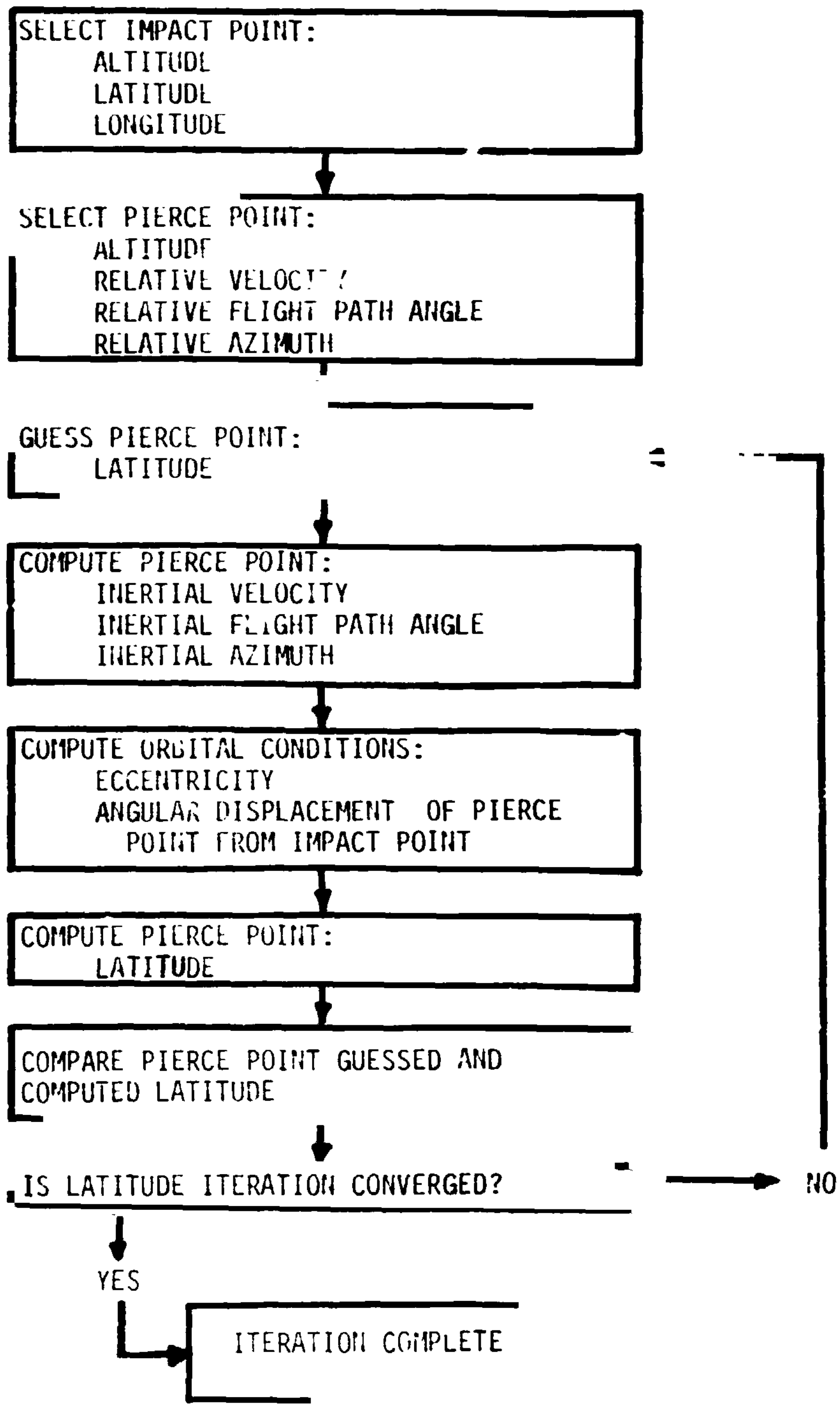
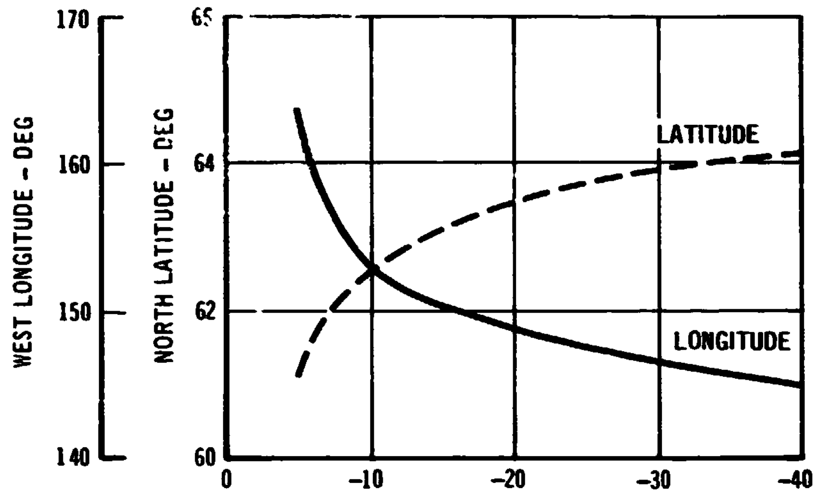


FIGURE 34

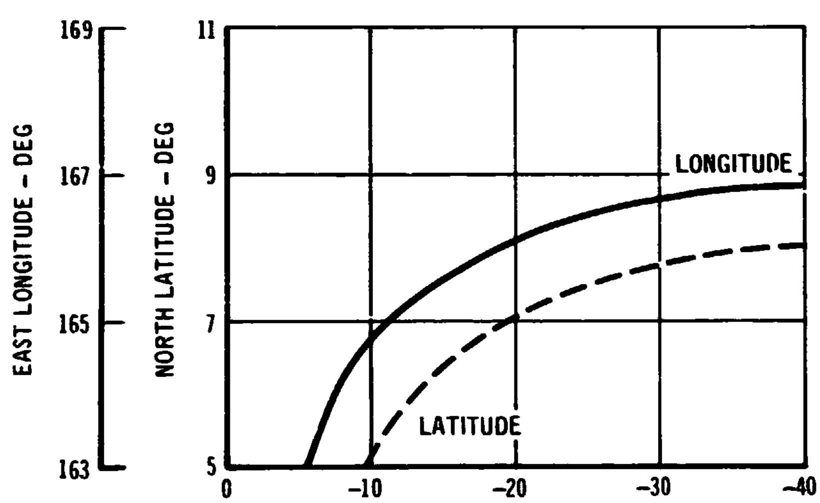
PIERCE POINT LATITUDE AND LONGITUDE

POKER FLAT
PIERCE POINT LATITUDE AND LONGITUDE



- IMPACT
  LATITUDE = 64.5°N
  LONGITUDE = 144.0°W
- PIERCE
  REL. VEL. = 25 KFT/SEC
  AZIMUTH = 60°

KWAJALEIN
PIERCE POINT LATITUDE AND LONGITUDE



- IMPACT
  LATITUDE = 8.90°N
  LONGITUDE = 167.40°E
- PIERCE
  REL. VEL. = 25 KFT/SEC
  AZIMUTH = 34°

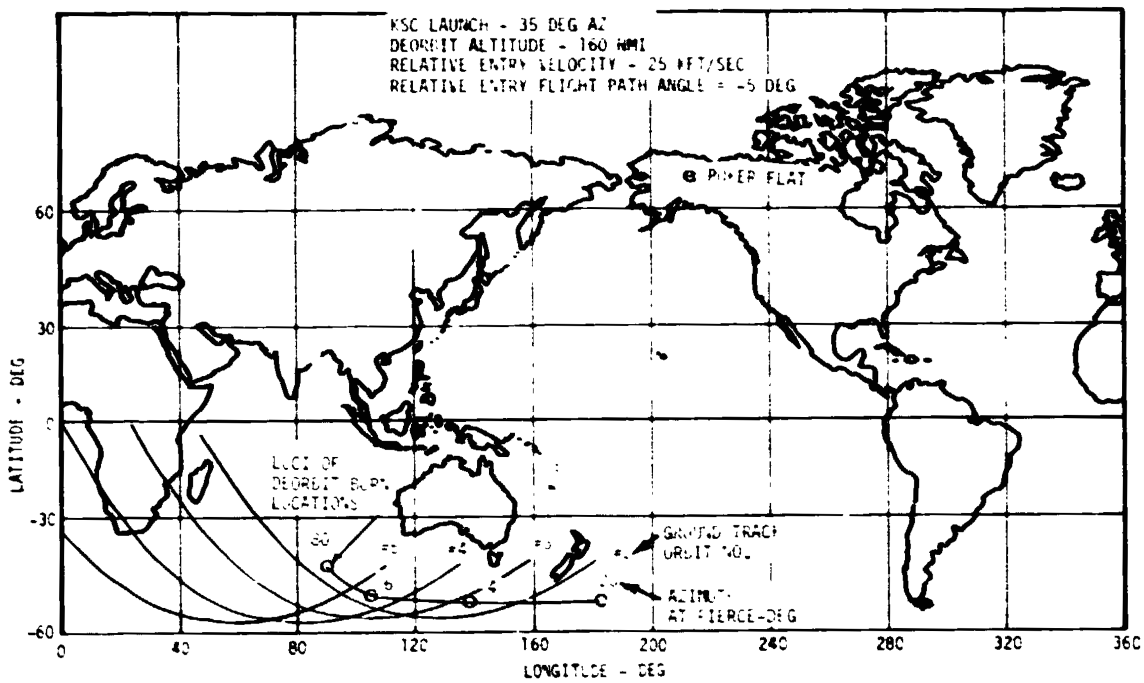
FIGURE 35

possible at that point. For the example shown in Figure 36, four orbits intersect the loci. In general an azimuth change is required at that point. Figure 37 shows the change in azimuths required for each orbit opportunity. The resultant combined ΔV is shown in Figure 38. Excessive ΔV results because the azimuth change from orbits 2 through 5 is excessive for this example. However, cases 1 and 6 of Section 16 were workable with this combined burn approach.

2. Single Plane Change - A second strategy for plane change involves performing a plane change which puts the payload at the deorbit point with the required azimuth. This solution begins by computing the deorbit burn locations as previously described. The amount of PDS plane change required from a particular location on the Shuttle orbit to pass through this deorbit burn location is then computed. The azimuth achieved by the PDS at the deorbit point can then be compared to the azimuth required at the deorbit point. If the azimuths are equal, then a single plane change maneuver initiated at the appropriate location on the Shuttle orbit is feasible. Case 1 first opportunity data shown in Section 16 is an example of the single plane change strategy. Note that the plane change ΔV is not minimum for this strategy.
3. Optimum Plane Change - The ΔV's can be reduced by another plane change strategy which allows for a single burn plane change at the optimum point. This is accomplished by changing the plane 90 degrees away from the deorbit point. As described for the combined burn strategy, the solution begins by computing the deorbit burn locations. The optimum plane change location is located on the Shuttle orbit and the resultant azimuth at the deorbit point determined. The azimuth required and the azimuth achieved at the deorbit point are then plotted as a function of relative pierce point azimuth. The pierce azimuth at which the required and achieved azimuths are equal corresponds to the pierce azimuth which minimizes plane change requirements. Cases 2 through 5 plane change requirements in Section 16 were determined in this manner.

A special application of plane change strategy 3 was used to define plane change requirements for the broad ocean area. Hawaii was selected as the impact area because it provides a base for logistics support and tracking coverage. In addition, a due east launch (maximum payload) from KSC provides an orbit which passes only slightly north of Hawaii for the first 4 orbits as shown in Figure 39.

DETERMINATION OF SINGLE BURN PLANE CHANGE & DEORBIT LOCATIONS FOR POKER FLAT IMPACT



NOTE: INTERCEPT OF GROUND TRACK AND LOCI OF DEORBIT BURNS IS POINT AT WHICH COMBINED BURN OCCURS

FIGURE 36

SINGLE BURN DEORBIT/PLANE CHANGE – PIERCE POINT AZIMUTH

- o 160 NMI SHUTTLE ORBIT
- o KSC LAUNCH -35 DEG AZIMUTH
- o POKER FLATS IMPACT
- o V_E = 25 KFT/SEC
- o γ_E = -5 DEG

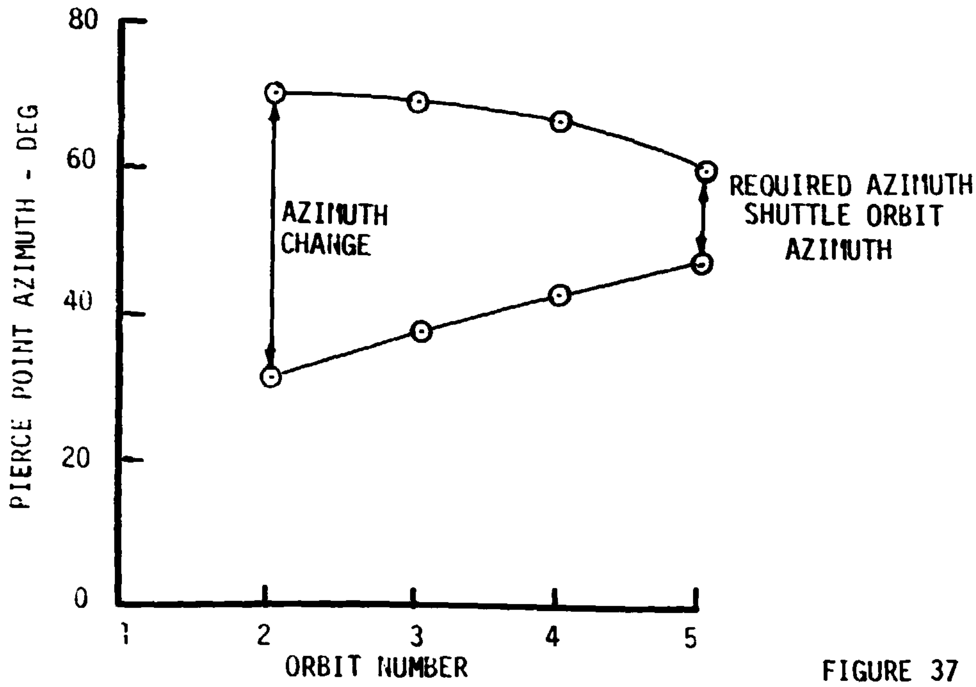


FIGURE 37

SINGLE BURN DEORBIT/PLANE CHANGE ΔV REQUIREMENTS

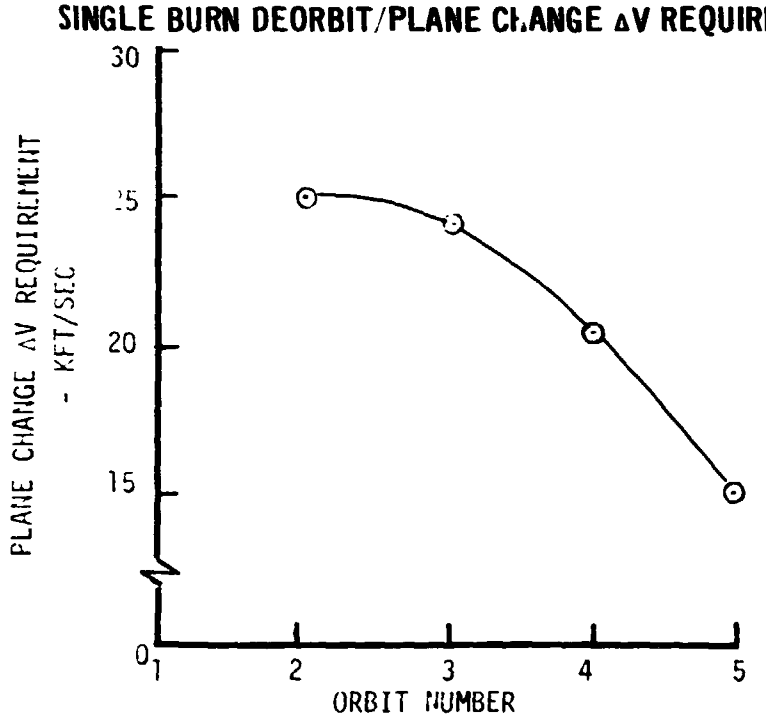


FIGURE 38

OPTIMUM BROAD OCEAN AREA SELECTION – STRATEGY

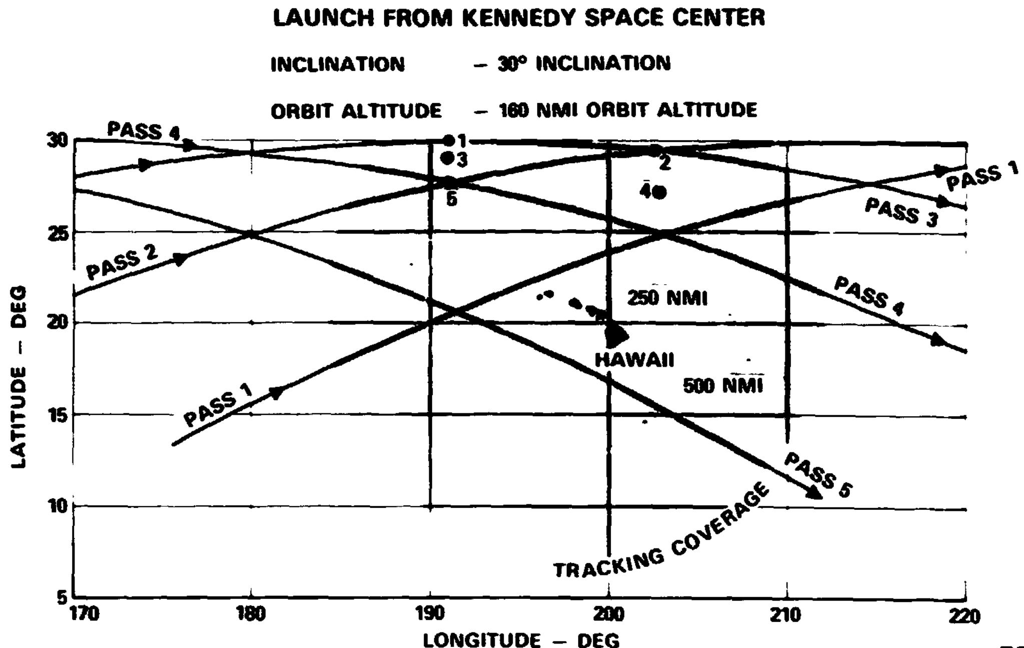


FIGURE 39

ΔV REQUIREMENTS FOR MULTIPLE OPPORTUNITIES AT HAWAII

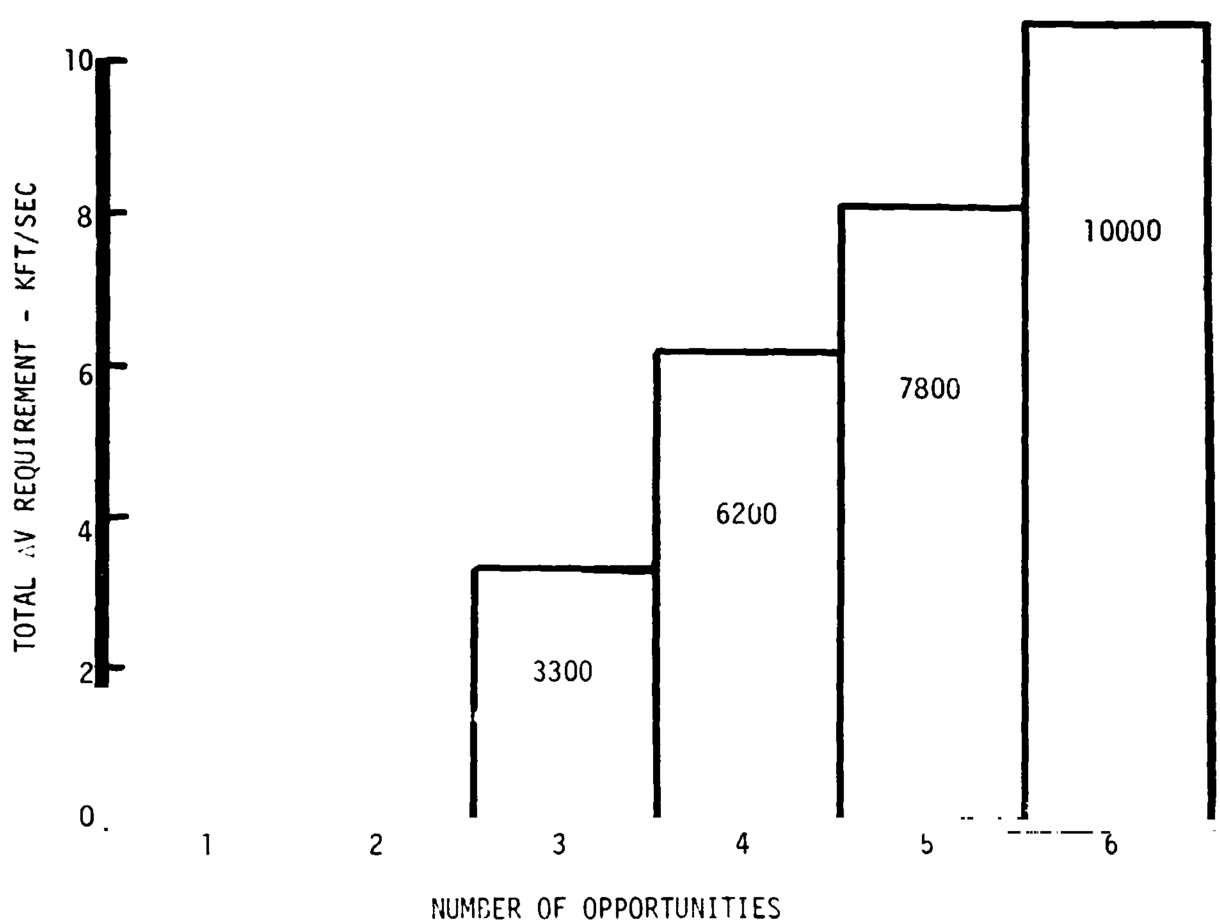


FIGURE 40

One possible strategy to maximize opportunities of impact at Hawaii is to deploy the PDS on the first orbit and continually change its orbit to pass over pre-selected impact points. If points 1 and 2 of Figure 39 are selected, no ΔV is required since overflight occurs on the Shuttle orbit. For three consecutive passes, point 3 provides the minimum ΔV. A similar result applies to points 4 and 5. Figure 40 shows the combined ΔV to maintain overflight on consecutive orbits of the PDS.

In summary, three strategies were developed for plane change. The combined burn strategy eliminates the two burn requirement and can be performed by a single stage solid rocket booster. The single plane change strategy is sometimes necessary if the pierce point allowable azimuths are severely constrained. Optimum plane change strategy is workable if azimuth constraints at the pierce point are not severe. Generally, each pierce point condition must be evaluated separately to determine which strategy is applicable.

7.3 RV Spacing - Additional PDS burns are required to provide time-spacing of multiple payloads and the booster at the pierce point. To insure complete KREMS radar coverage of each payload approximately 90 second spacing is required. This is sufficient time to track the lead payload to impact, re-elevate, and reacquire the trailing payload before it reenters at 300 kft altitude.

The deployment scenario is shown in Figure 41 for both the upleg deorbit (Strategy A) and the Hohmann transfer deorbit (Strategy C). For Strategy A the first burn is the deorbit burn and places the PDS on the first payload trajectory. If no further burns are made, the PDS will reenter simultaneously with the first payload. After completion of the first burn, the first RV is separated from the PDS. A second burn is made which places the PDS on a trajectory which will reenter with the required spacing after the first payload. This trajectory has a slightly higher apogee and steeper flight path angle compared to the first payload. Additional burns are made until all payloads are deployed. A final burn is made to space the PDS the required time behind the last payload.

The burn sequence is analogous for Strategy C, except the deorbit burn corresponds to the second burn of the Hohmann transfer. Typically subsequent burns require higher ΔV compared to Strategy A to achieve the same spacing. The higher ΔV results because the burns must be made closer to the pierce point, i.e., on the downleg of the trajectory. The ΔV increase is particularly large for Strategy B downleg deorbits. In fact, for most reentry conditions, 90 second spacing is not possible for Strategy B. As a consequence, Strategy B is best suited for single payload deliveries.

The spacing ΔV requirements discussed in this section assume that each payload is targeted for the same pierce point in inertial space. As a consequence, these are inplane maneuvers. For an eastward launch from KSC, this results in each subsequent payload impacting southwest of the previous payload due to the earth's rotation and steeper flight path angles required for spacing. In Section 16, the spacing burns include a small out-of-plane burn which targets each payload to the same impact point. The spacing requirement is maintained at 300 kft altitude but at a different pierce point latitude and longitude for each payload.

The timing between burns was assumed to be 450 sec for the deorbit burn and 60 sec for subsequent spacing burns. This allows sufficient time for PDS maneuver, deployment, and reorientation prior to the next burn. For shallow flight path angle, high velocity reentries the deorbit burn time is much less than 450 sec.

TWO R/V DEPLOYMENT SCENARIO

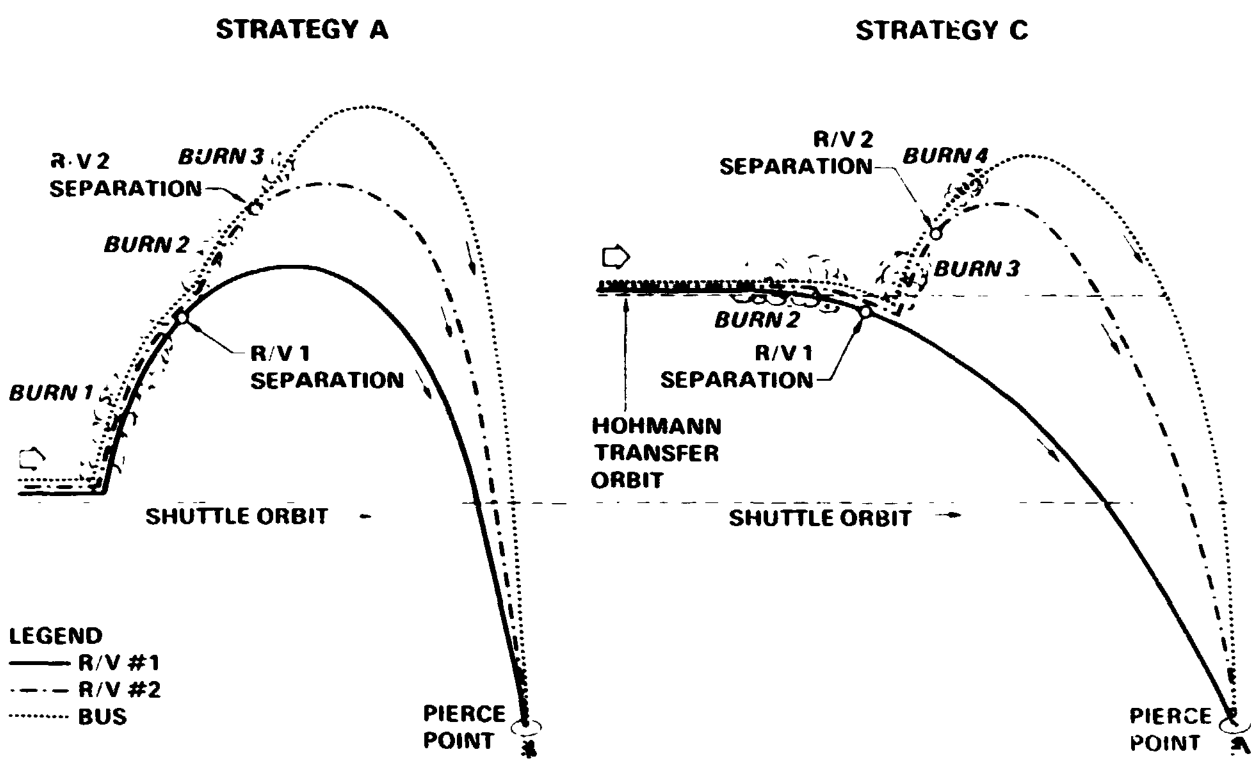


FIGURE 41

However, 450 sec was allowed for that burn throughout this section. In Section 16, this time was reduced for cases 2 through 5.

Figures 42 and 43 present the ΔV requirements as a function of burn number for inplane spacing maneuvers and 90 sec spacing at pierce. Region R includes reentry velocities between 20 and 25 kft/sec and flight path angles between -20 and -40 deg; Region F extends the flight path angle to -5 deg; and Region S to -60 deg. In general, Strategy C spacing ΔV's of Figure 43 are higher than Strategy A of Figure 42 because the Strategy C burns occur closer to the pierce point. In either case, spacing of payloads at shallow flight path angles requires high ΔV because of the short time of flight involved. On the other hand, at high velocities and steep flight path angles, the trajectory times between the burns and pierce are large and only small ΔV's are required to effect the 90 sec spacing.

Spacing is achieved by changing the entry conditions to provide a longer deorbit time for subsequent payloads. Figure 44 compares the consecutive payload reentry velocities with a first payload velocity of 25 kft/sec and a range of flight path angles. At the steeper angles between -20 and -60 degrees, subsequent payloads reenter at slightly greater velocities for strategy A. For strategy C and strategy A at shallow flight path angles, subsequent payload velocities decrease with payload number. The decrease is especially dramatic at -5 deg flight path angles where velocities less than 20 kft per result. In effect, the 90 sec spacing can only be achieved by significantly slowing down the payloads. The slow down is even more severe for Strategy C.

The 90 sec spacing imposes a severe requirement for strategy C. Figure 45 demonstrates the sensitivity of ΔV to reduced spacing requirements. Reducing the spacing time to 60 sec, reduces ΔV for all payloads to slightly greater than 1.0 kft/sec for the entry conditions shown. Another approach to reducing ΔV requirements is to do all the spacing burns simultaneously.
There is a significant advantage for this if a large number of payloads is to be deployed. Nevertheless, the strategy C requirements are always greater than strategy A or the ground launch as shown.

Often there is a requirement to space the booster not only in time but in range at the pierce point. Figure 46 presents typical booster spacing ΔV requirements as a function of first payload velocity and flight path angle. Strategy C ΔV requirements are excessive again because of the proximity to the pierce point.

STRATEGY A PAYLOAD SPACING ΔV REQUIREMENTS ENVELOPE

- ΔV'S OF 500-3000 FT/SEC ARE REQUIRED PER PAYLOAD IN REGION R

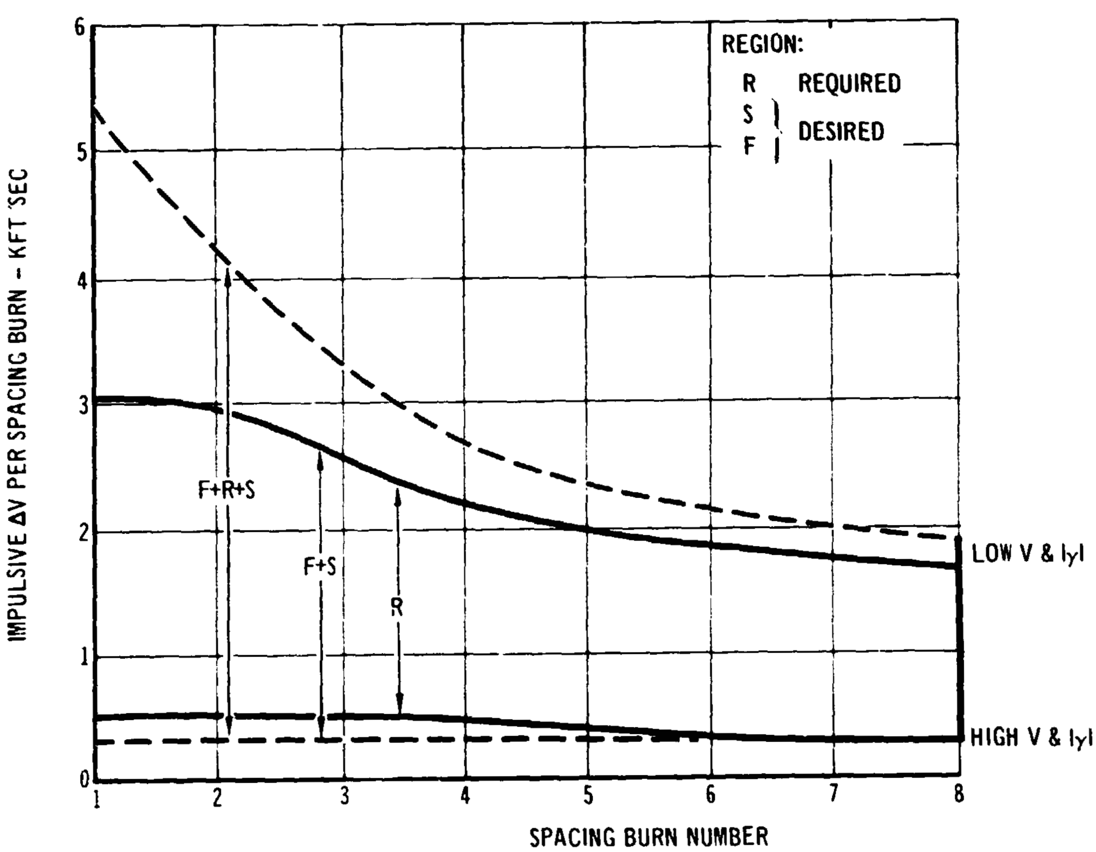


FIGURE 42

STRATEGY C PAYLOAD SPACING ΔV REQUIREMENTS ENVELOPE

- ΔV'S MUCH HIGHER THAN STRATEGY A ARE REQUIRED

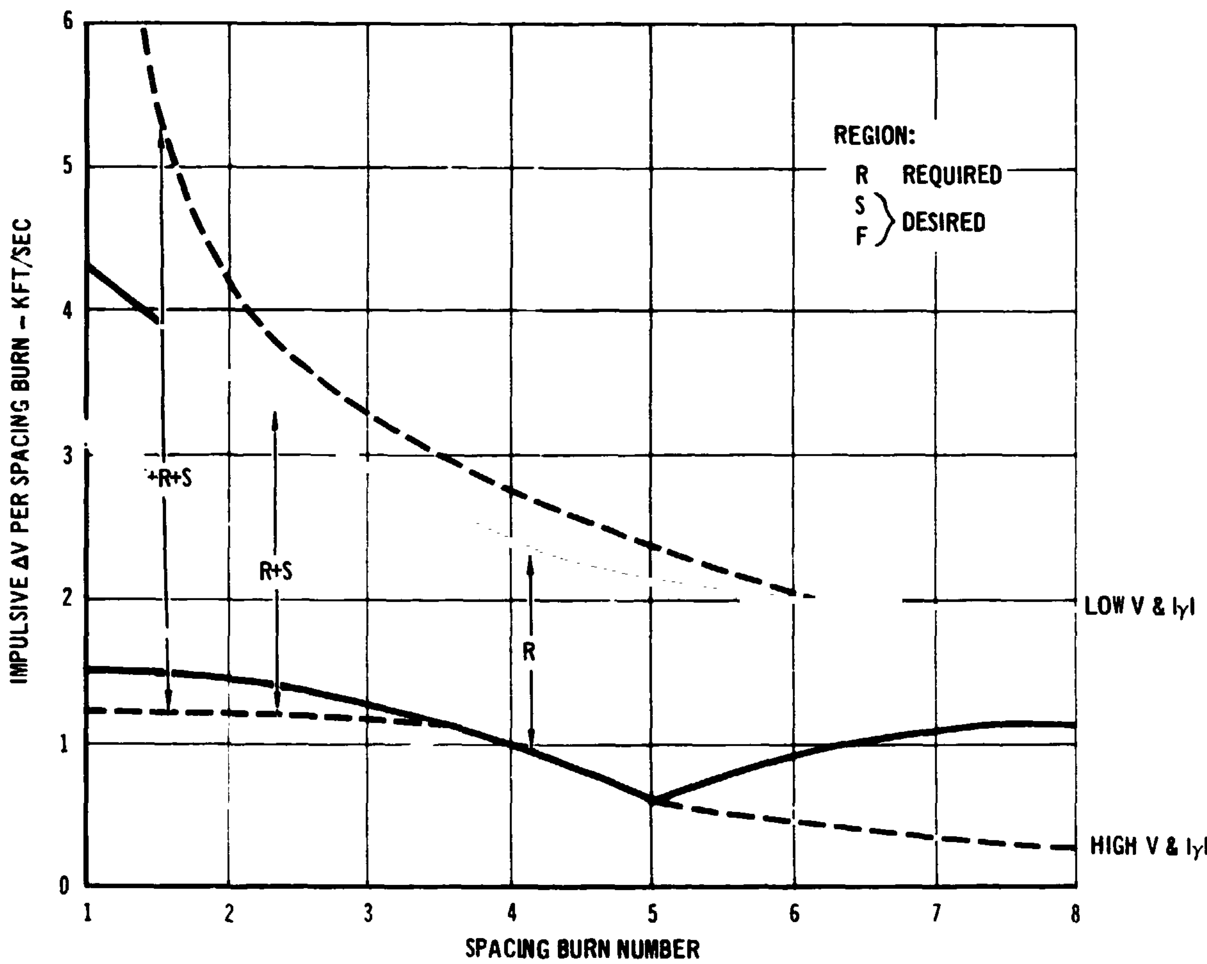
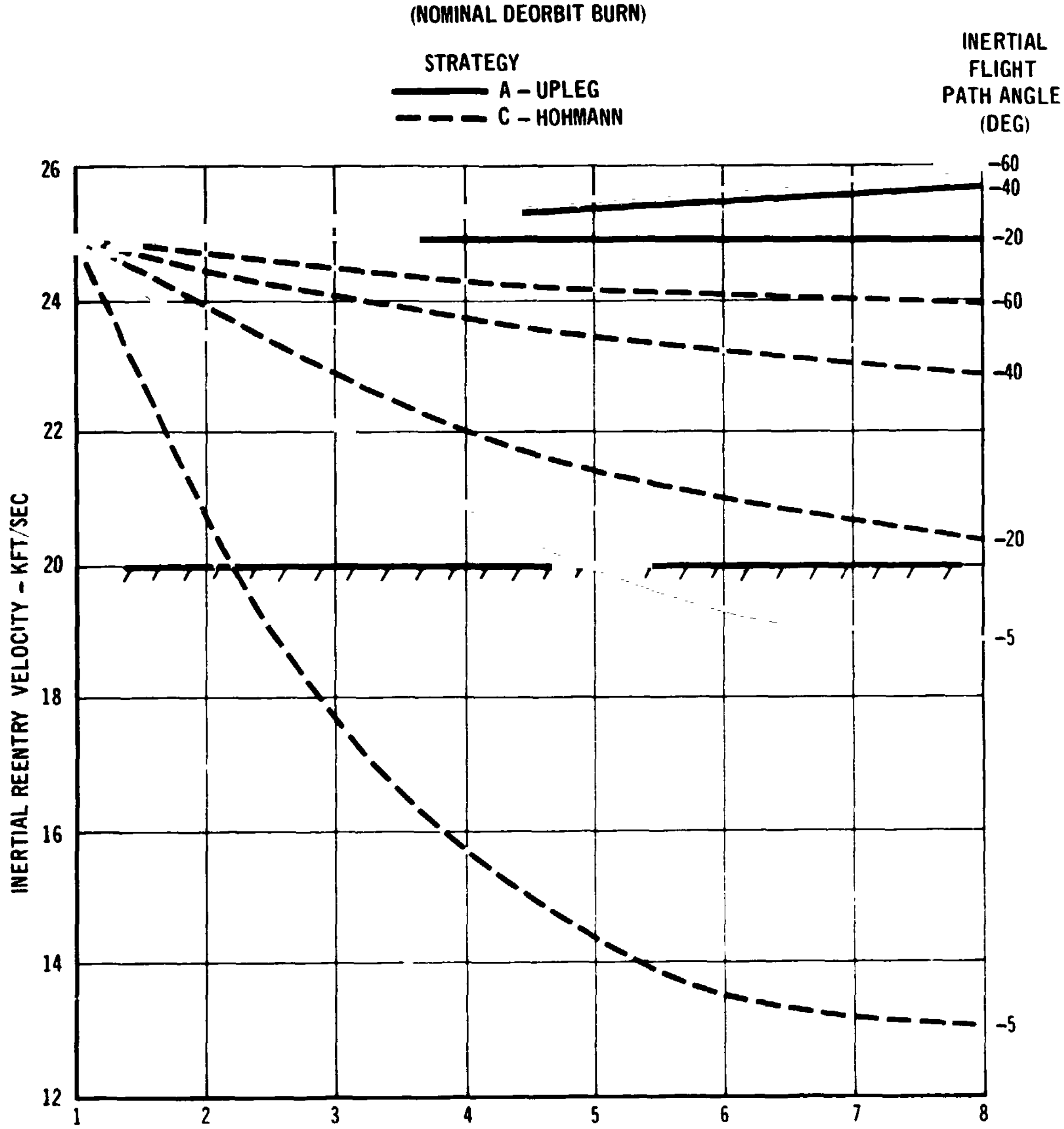


FIGURE 43

COMPARISON OF STRATEGY A AND C MULTIPLE PAYLOAD REENTRY VELOCITIES

(NOMINAL DEORBIT BURN)



• STRATEGY C ENTRY VELOCITY VARIES SIGNIFICANTLY WITH PAYLOAD NUMBER

FIGURE 44

COMPARISON OF PAYLOAD SPACING ΔV REQUIREMENTS

- STRATEGY C – HOHMANN ΔV REQUIREMENTS CAN BE REDUCED
- WILL BE GREATER THAN GROUND LAUNCH

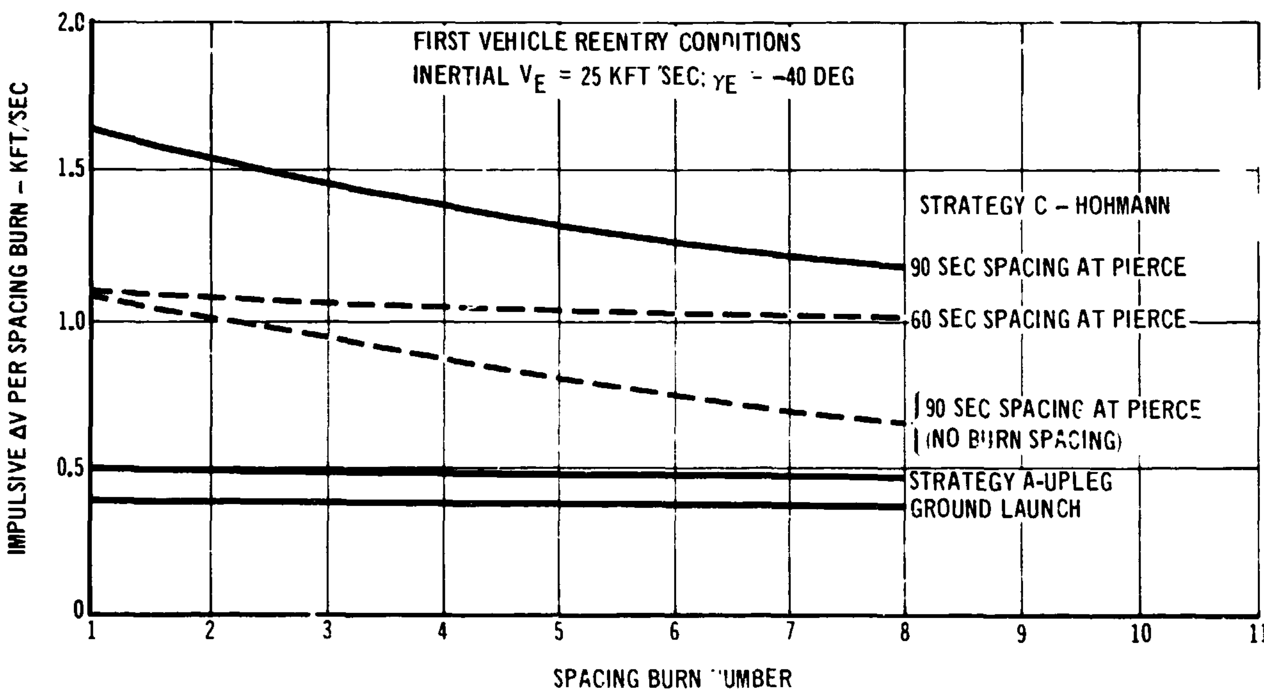


FIGURE 45

BOOSTER DEORBIT ΔV REQUIREMENTS

- LOW ANGLE AND VELOCITY REQUIREMENTS ARE EXCESSIVE
  - ASSUMPTIONS
    BOOSTER PAYLOAD DEORBIT BURN & MANEUVER REQUIRES 450 SEC
    BOOSTER REENTERS 90 SEC BEFORE FIRST PAYLOAD 100 NM UPRANGE

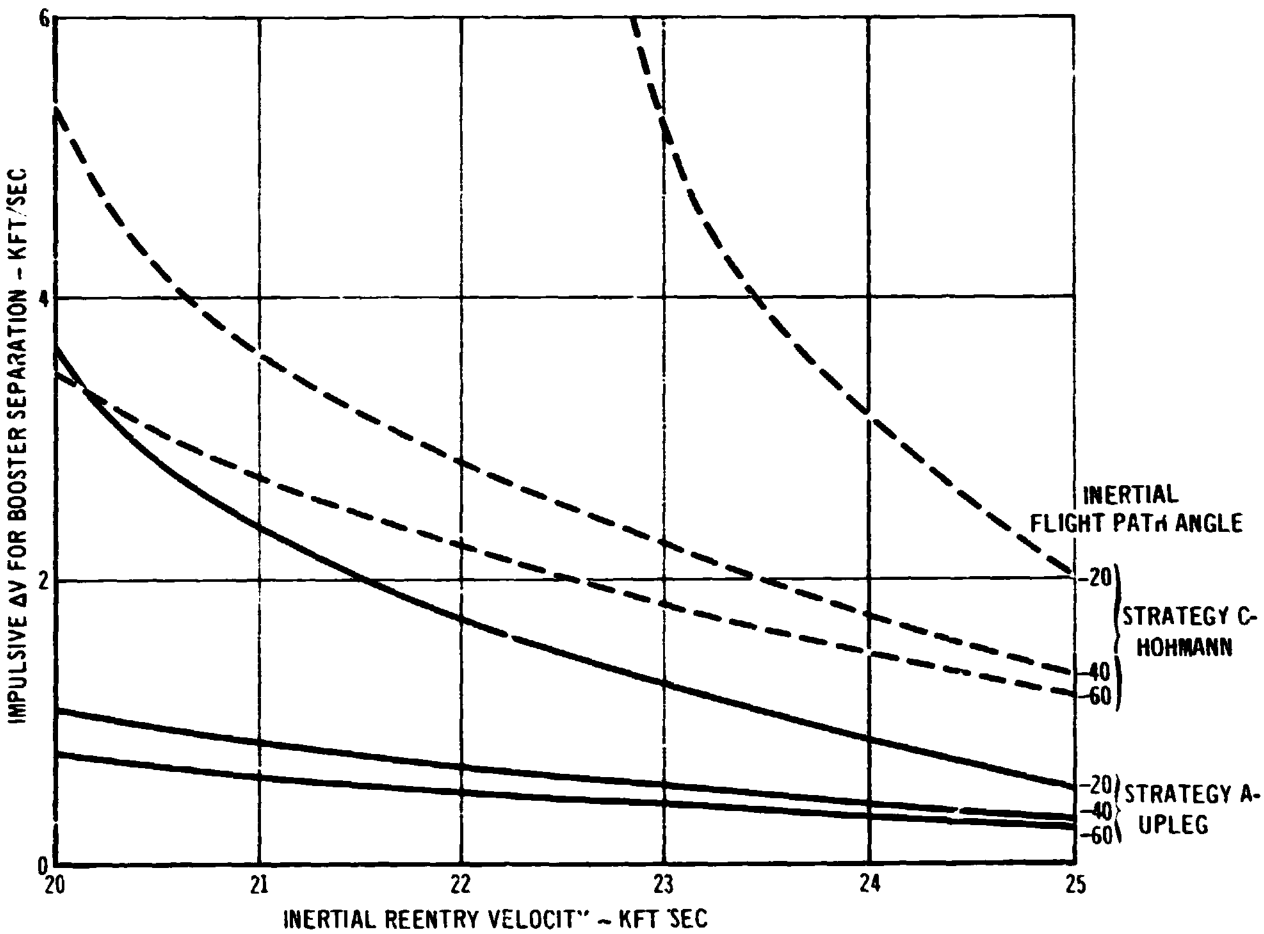


FIGURE 46

In conclusion, typical spacing ΔV's of 500-3000 fps per payload are required for an upleg deorbit. The strategy C-Hohmann transfer spacing ΔV's are much higher. In either case, the high velocity/steep flight path angle require lower ΔV's. However, spacing and initial deorbit ΔV combine to significantly increase booster ΔV requirements for many heavy vehicles entering at low speed/steep flight path angles.

7.4 Ground Tracks - Ground tracks for the deorbit trajectories and the Shuttle orbits were generated to establish Shuttle and payload locations for the communications and range safety analyses in Sections 12 and 15 respectively and the plane change requirements discussed in Section 7.2. First, typical Shuttle orbit ground tracks are considered; then deorbit ground track sensitivity to entry conditions; and finally, conclusions based upon the deorbit, plane change, spacing and ground track analyses are presented.

7.4.1 Shuttle Orbital Ground Tracks - A due east launch from KSC appears attractive because it provides maximum payload and first orbit overflight of KMR as shown in Figure 47. In addition, plane change, upleg deorbit, and downleg deorbit can also be accomplished in the first orbit. Unfortunately, there is probably insufficient time between Shuttle launch and PDS first burn for pre-launch checkout and preparation. The second and third orbits also provide deployment opportunities and the potential for initial Hohmann transfer burns in the vicinity of the U.S. or South America. This could be attractive for ground coverage purposes. However, the payloads would reenter through the inhibit corridors at KMR. Unless the inhibit zones at KMR are bypassed completely, the due east launch from KSC is not the best launch azimuth.

A better orbit results from a launch at the maximum northward azimuth of 35 deg. Figure 48 shows the resulting ground tracks for orbits 1-6 and 10-14. The shadowed region of the ground tracks represent regions in which a PDS plane change can be performed to provide KMR overflight and azimuths within the southwest and northwest corridor. The stars represent the optimum locations for plane change.

Orbits number 4, 5, 12 and 13 pass to the east or west of KMR. These provide deorbit opportunities with minimum plane change requirements as described in Figure 33 of Section 7.2. Orbit 4 provides the best opportunity with respect to maneuver locations and overflight of tracking stations. The deorbit maneuver for KMR impact of cases 1, 2 and 3 in Section 16 is accomplished during orbit 4. Poker Flat is too far north for overflight although PDS plane changes on orbits 5 or 6 can provide impact there as describe in Section 16.

ORBITAL GROUND TRACKS FOR WESTERN APPROACH TO KWAJALEIN

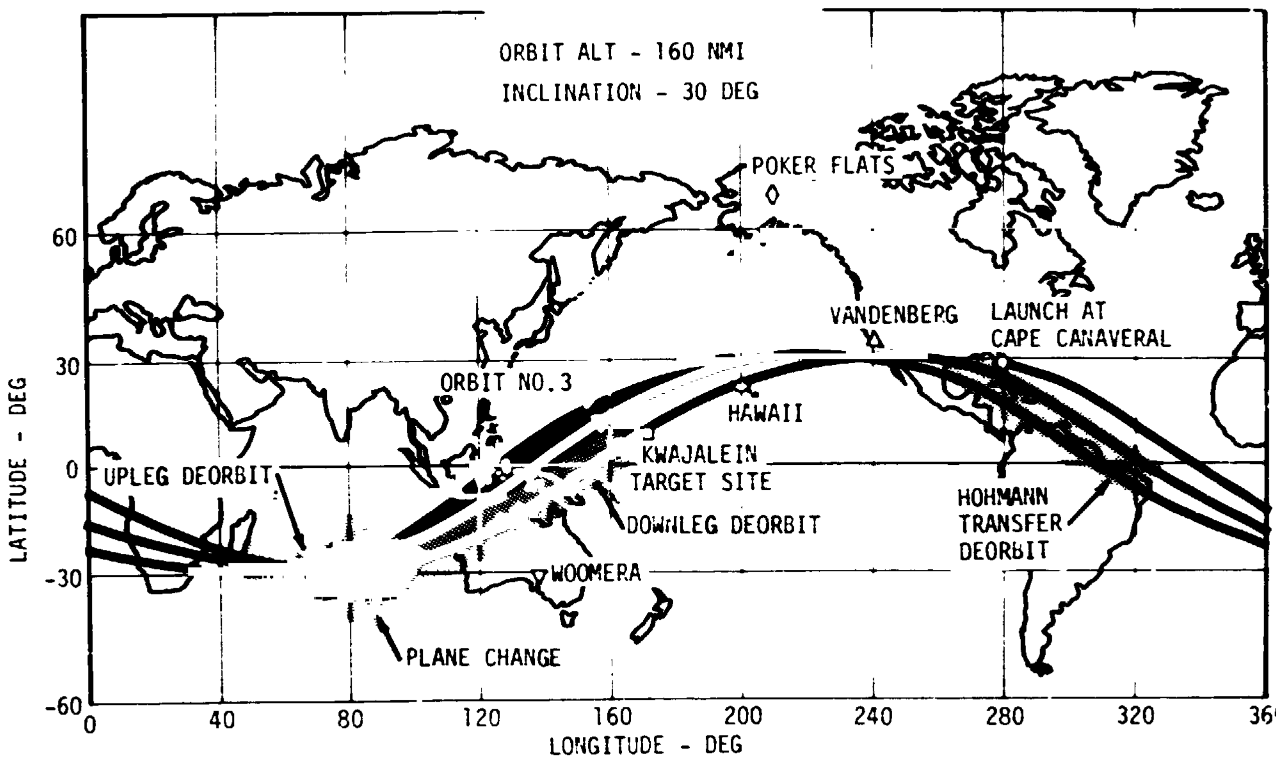


FIGURE 47

TYPICAL GROUND TRACK FOR EASTERN LAUNCH

APPROACH OPPORTUNITIES TO KMR

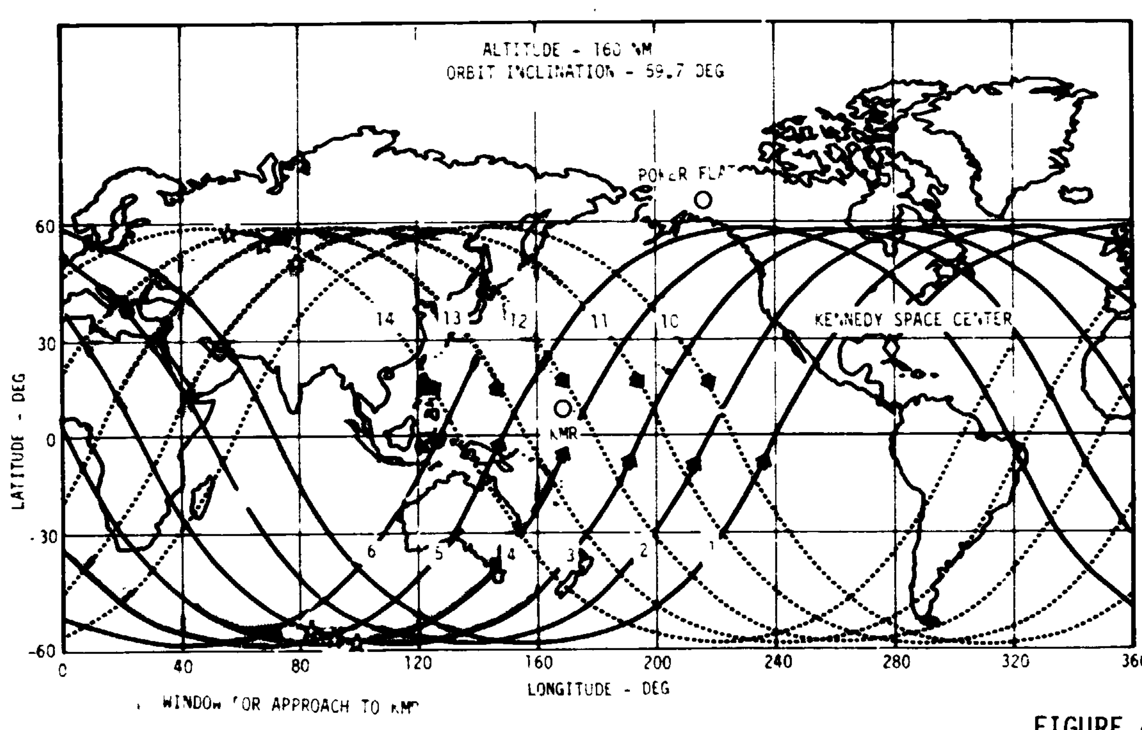


FIGURE 48

On the other hand, Poker Flat overflight can be achieved by launches out of VAFB. The ground tracks of orbits 1 through 5 and 12 through 14 for a 72 deg inclination orbit are shown in Figure 49. The windows and optimum maneuver locations for plane change are identified as described previously. Orbits 2, 12 and 13 provide near overflight of Poker Flat. The plane change for orbit 2 occurs over Central Asia; for orbits 12 and 13 it occurs south of the equator. For cases 4 and 5 of Section 16 the deorbit maneuver is made in orbits 12, 13 and 14.

A due south launch from VAFB provides overflight of the north pole as shown for orbits 1 through 10 in Figure 50. Orbits 1, 2, 9 and 10 provide near overflight of Poker Flat; orbit 3 provides near overflight of KMR. Orbit 3 was selected for the first opportunity deorbit of case 6 in Section 16. Approaching Poker Flat from the south on orbits 9 and 10 provides an open corridor over the Pacific ocean with overflight of Hawaii on orbit 10. Although this approach was not developed in Section 16, it is a viable approach to Poker Flat for a VAFB launch.

The above ground track data aided in the identification of Shuttle orbits for which the PDS deorbit maneuvers could be performed. Figure 51 summarizes the orbits which provide the best opportunities for deorbit at the various impact sites. Which of these orbits provide minimum plane change ΔV and appropriate approach azimuth is a function of the required deorbit burn location.

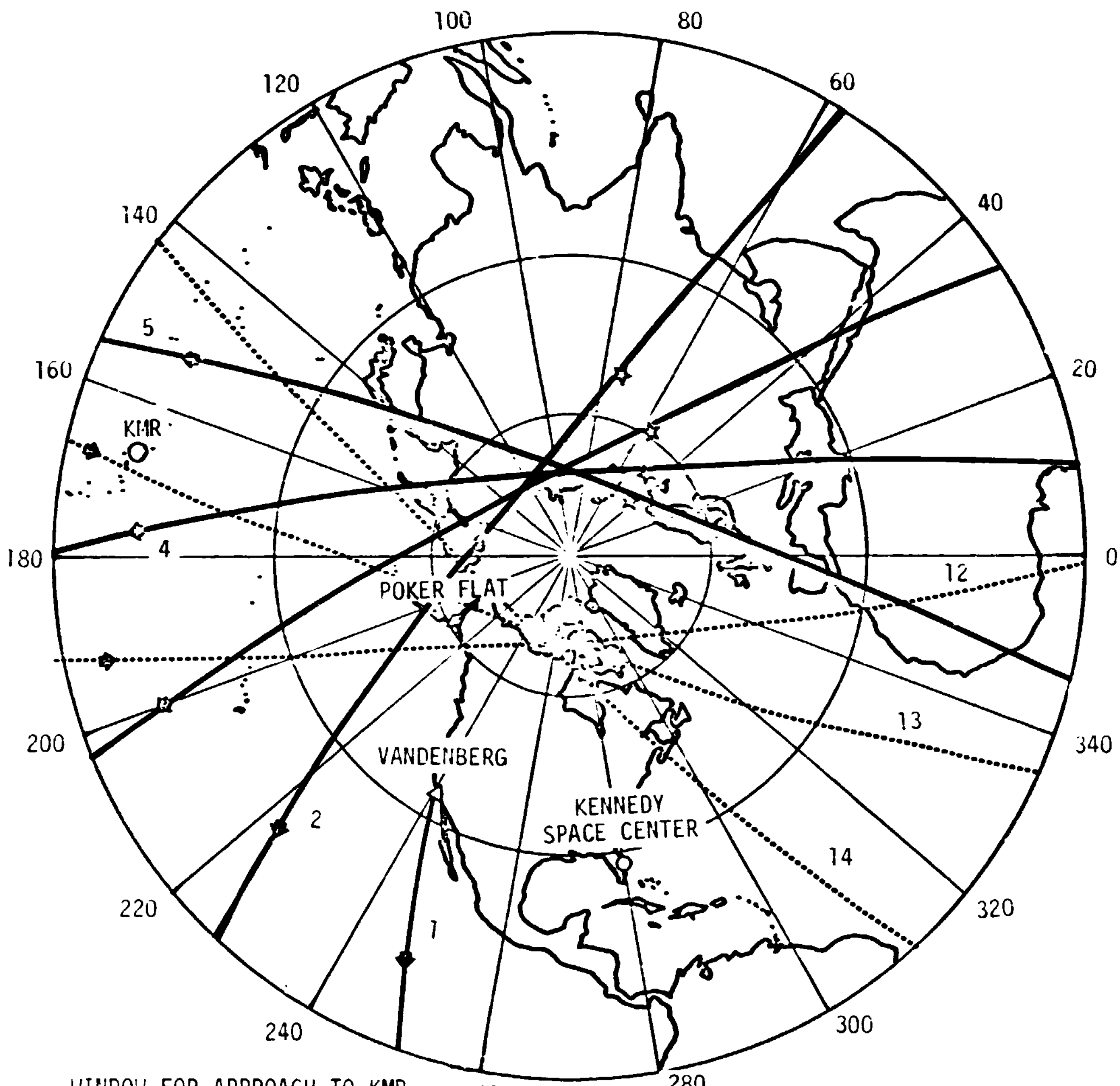
7.4.2 Deorbit Location Sensitivities - Figure 52 gives an example of the locations of the deorbit burn for relative azimuth of 60 and 34 deg at Poker Flat and KMR, respectively. The relative reentry velocity is 25 kfps at both locations. The Poker Flat deorbit burns occur south of Australia for shallow flight path angles and over it for moderate angles. The KMR deorbit burns occur near the southern tip of Africa for shallow angles and over the Indian Ocean for moderate angles. For the shallow reentry angles, the range from deorbit maneuver to impact is large due to the high inertial velocities (greater than 26000 ft/sec in some cases) required to achieve relative velocities of 25000 ft/sec. These higher energy orbits require more range from deorbit maneuver to impact.

The sensitivity of deorbit burn location to relative approach azimuth and reentry velocity for a -5 degree path angle at Poker Flat is shown in Figure 53. Range from pierce is relatively insensitive to pierce azimuth. However,

TYPICAL GROUND TRACK FROM VANDENBERG LAUNCH APPROACH OPPORTUNITIES TO KMR

ALTITUDE - 160 NM

ORBIT INCLINATION - 72.0 DEG



WINDOW FOR APPROACH TO KMR

☆ MINIMUM OUT OF PLANE ΔV

FIGURE 49

TYPICAL GROUND TRACK FROM VANDENBERG LAUNCH APPROACH OPPORTUNITIES TO POKER FLAT

NORTHERN HEMISPHERE
POLAR ORBIT – 160 NMI

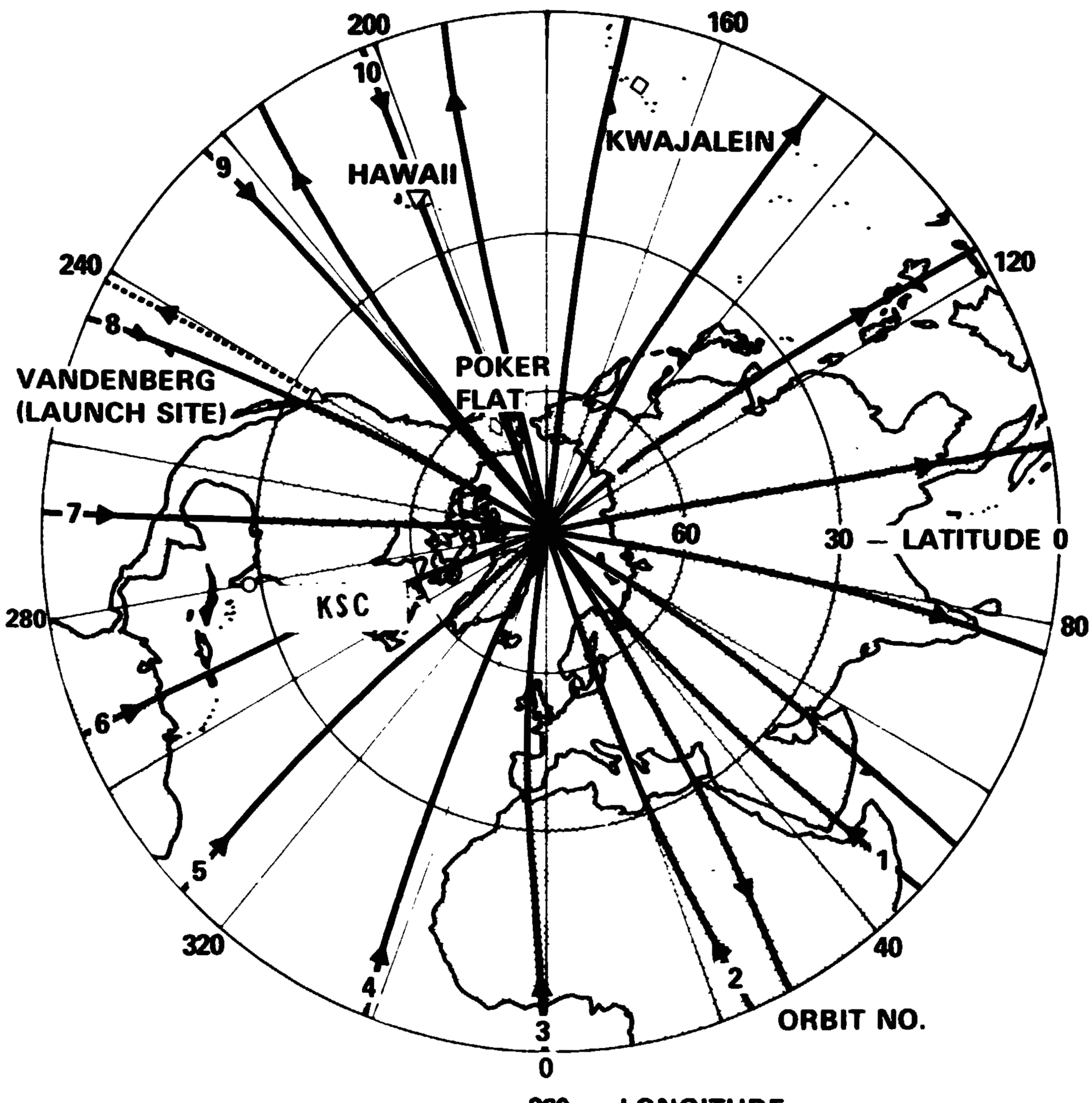


FIGURE 50

PREFERRED ORBITS FOR PDS DEORBIT (160 NMI SHUTTLE ORBIT ALTITUDE)

LAUNCH SITE	IMPACT AREA	SHUTTLE ORBIT INCLINATION (DEG)	PREFERRED ORBITS FOR DEORBIT
KSC	KMR	57	4, 5, 12, 13
	POKER FLAT	57	5, 6
	HAWAII	30	1 - 6
VAFB	KMR	72	4, 5, 13, 14
	KMR	90	3, 4, 11, 12
	POKER FLAT	72	2, 12, 13
	POKER FLAT	90	1, 2, 9, 10

FIGURE 51

SENSITIVITY OF DEORBIT BURN LOCATIONS TO REENTRY FLIGHT PATH ANGLE FOR KWAJALEIN AND POKER FLAT

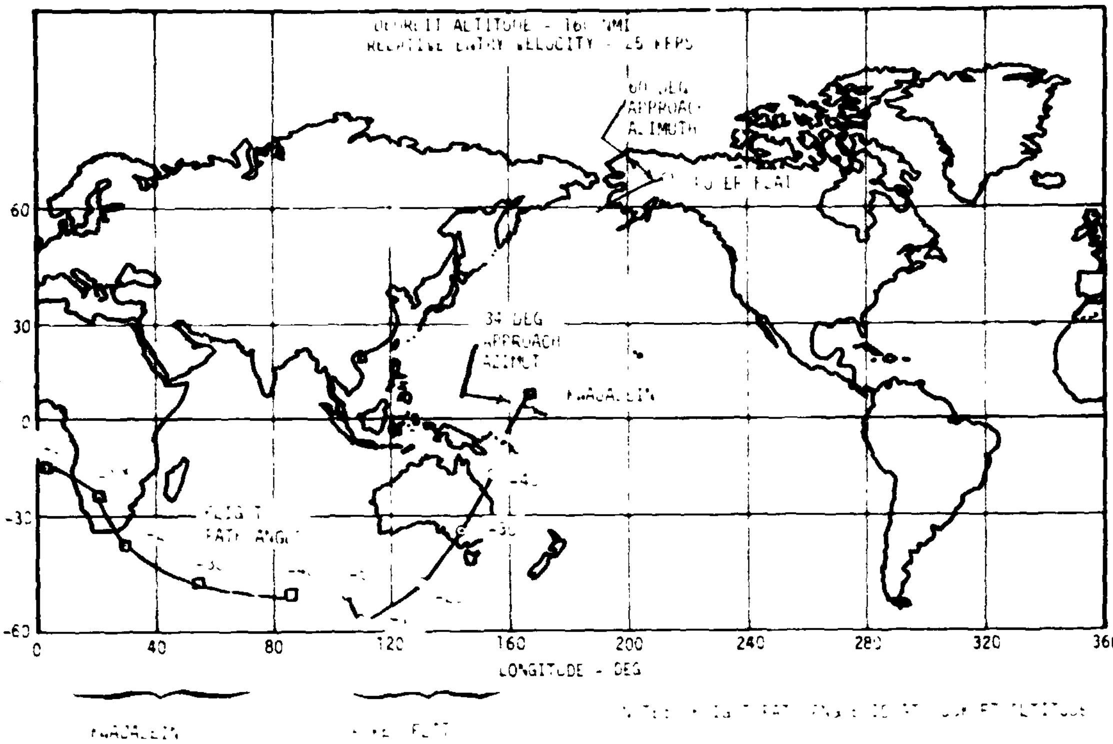


FIGURE 52

SENSITIVITY OF DEORBIT MANEUVER LOCATIONS TO APPROACH AZIMUTH & REENTRY VELOCITY

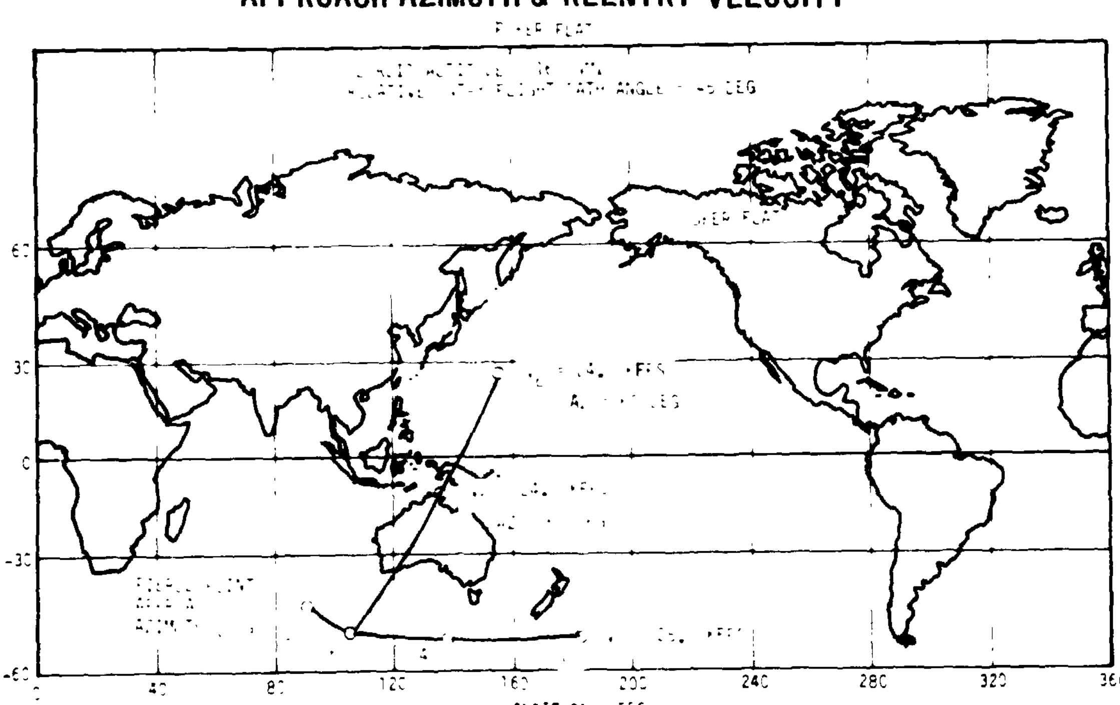


FIGURE 53

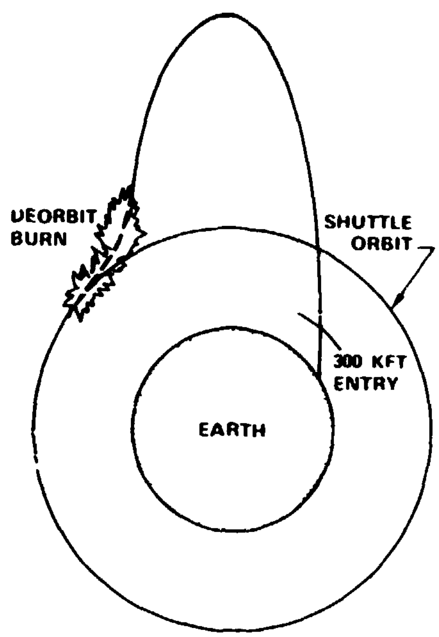
a change of reentry velocity from 25000 to 24000 fps changes maneuver range by more than a factor of two. As a consequence, if deorbit burns for Poker Flat are to be performed within line of site of Kwajalein or Guam, the relative reentry velocities would be constrained to approximately 24000 fps. The analyses of Section 7.1 were done in inertial coordinates since the approach azimuth at pierce was unspecified during the initial study. These original cases correspond to approximately 24500 fps relative reentry conditions at a 60 degree approach to Poker Flat. In combination the reentry velocity and azimuth selection allows locating the deorbit burn at favorable latitudes and longitudes. However, this may not be practical for two reasons. First, the reentry velocity may be fixed by the experiment requirements. Second, and more importantly, the plane change requirements may be excessive for particular locations of the deorbit points. These considerations are unique to each particular case.

7.4.3 Delivery Strategy Conclusions - By analyzing deorbit, plane change, payload spacing and ground track requirements, the unique capabilities of Shuttle described in Figure 54 were identified. Because the PDS deorbit maneuver can be initiated at any point on a Shuttle orbit, the simulation of 1500-7100 NMI trajectories is possible. (The Shuttle is a mobile launch platform.) In addition, full coverage by the KREMS radars is possible by selecting approach azimuths at KMR within the allowable corridors. Impact at Poker Flat is achievable from either a VAFB or a KSC Shuttle launch. Payload deployment on consecutive orbits provides multiple pass deorbit opportunities without severe plane change ΔV requirements. In many cases, multiple payload spacing maneuvers from an upleg deorbit require ΔV comparable to ground launched systems.

The above conclusions provided the basis from which much of the detailed example case requirements described in the following sections evolved.

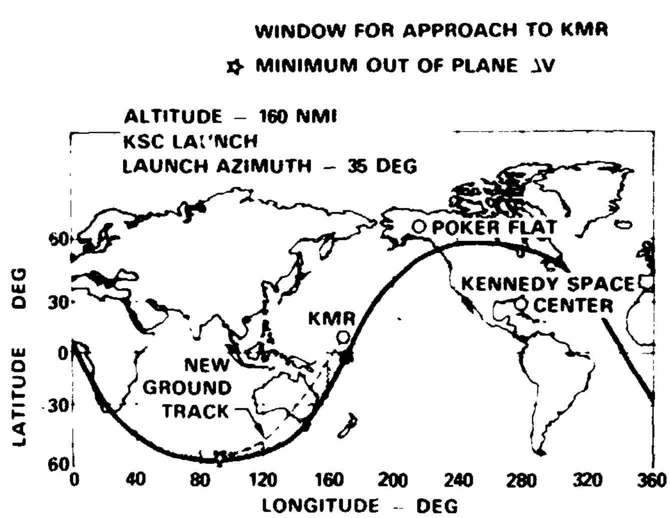
SHUTTLE/PDS MANEUVER STRATEGY STUDIES HAVE IDENTIFIED UNIQUE CAPABILITIES

DEORBIT



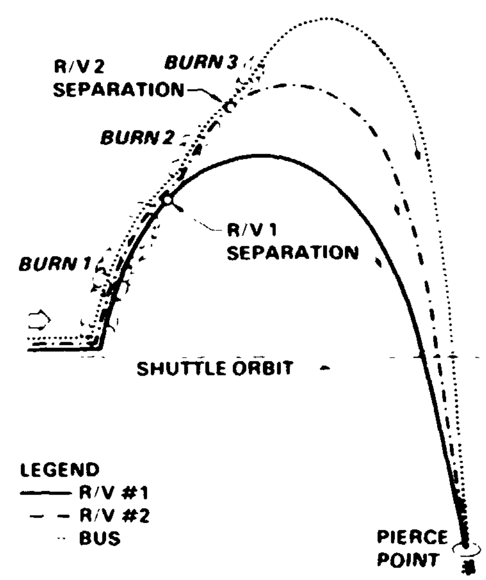
- ALLOWS TRAJECTORY RANGE SIMULATIONS OF 1500-7100 NMI

PLANE CHANGE



- PROVIDES FULL KREMS COVERAGE
- PROVIDES MULTIPLE PASS DEORBIT OPPORTUNITIES AT KWAJALEIN, HAWAII & POKER FLAT

PAYLOAD SPACING



- IMPROVES UPON MULTIPLE PAYLOAD NUMBER & WEIGHT OF GROUND LAUNCH SYSTEMS

FIGURE 54

8.0 BOOSTER CAPABILITIES

The physical characteristics and performance capabilities of typical Shuttle launched boosters are summarized in this section. The purpose of these data is to provide booster characteristics for use in defining PDS performance in Sections 9 and 16. Figure 55 shows the candidates boosters and assembled vehicles in relation to the Shuttle.

These boosters fall into four categories which cover the existing technology range of upper stage performance and physical size. Each class is illustrated by a representative design. Category 1 is a cryogenic propellant class of booster such as the Centaur. This class represents the highest available performance and the biggest size. Category 2, an existing storable propellant booster, such as the Transtage, is presented as an example of intermediate size and performance. Agena and Delta also fall into this category. Category 3 is a storable propellant design based on using components from the Shuttle auxiliary propulsion system. It is the minibus described in Section 9 and allows maximum opportunity for shared payload launches of the Shuttle. Category 4, a solid propellant booster, is shown using the best available definition of the recently selected Interim Upper Stage (IUS) concept. Category 2 or 3 boosters are best suited for the DoD type missions.

In the following sections, each booster class is described by a survey of dimensional mass, and propulsion characteristics. The performance capabilities in terms of payload mass versus velocity increment is provided in Section 9 for representative boosters.

8.1 Existing Cryogenic Booster - The Centaur, which is the only cryogenic (O_2/H_2) stage currently in use, is shown as the example design for this class of booster. Figure 56 presents the mass, dimensional and propulsion characteristics of the Centaur. It is a large booster almost 32 ft in length with a launch mass of nearly 38000 lbm. For many of the DoD missions it is oversized. However, it has application to DoD mission requiring reentries at low velocity or steep flight path angles.

8.2 Existing Storable Booster - The Transtage is selected as the example design for this booster class on the basis that it provides maximum performance capability with minimum modification. Other designs considered were the Delta and Agena stages. Physical and propulsion characteristics of the Transtage are described in Figure 57. The relatively short length of less than 15 feet provides efficient packaging in the Shuttle payload bay and considerable space for additional

BOOSTER CANDIDATES

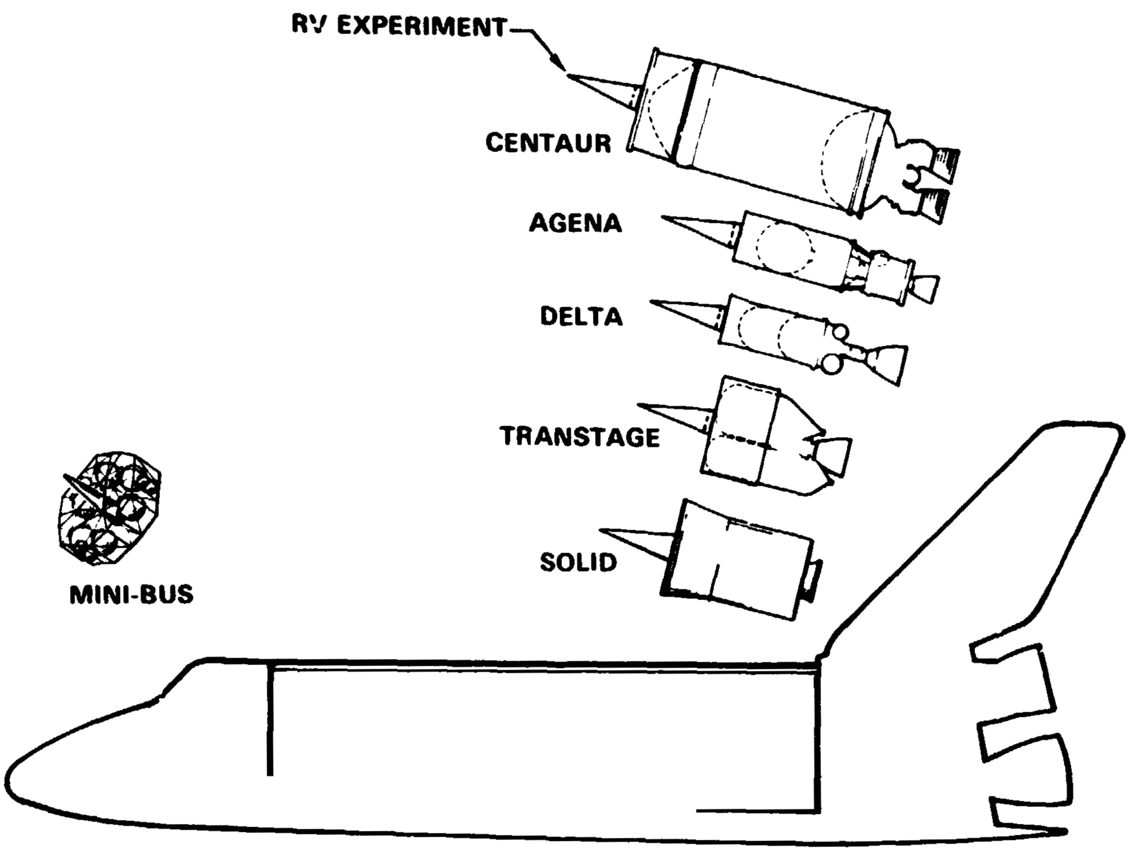
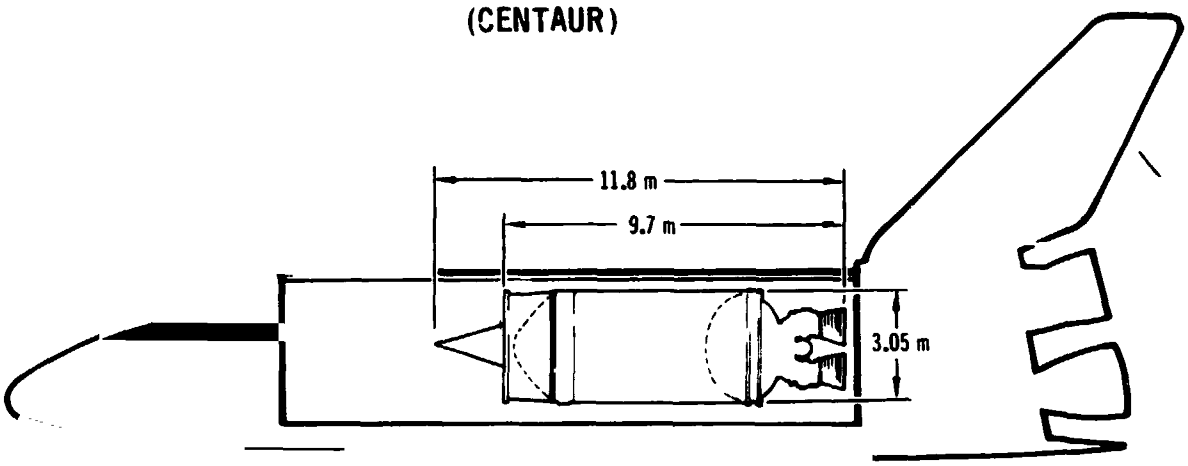


FIGURE 55

TYPICAL CRYOGENIC PROPELLANT BOOSTER DESCRIPTION (CENTAUR)

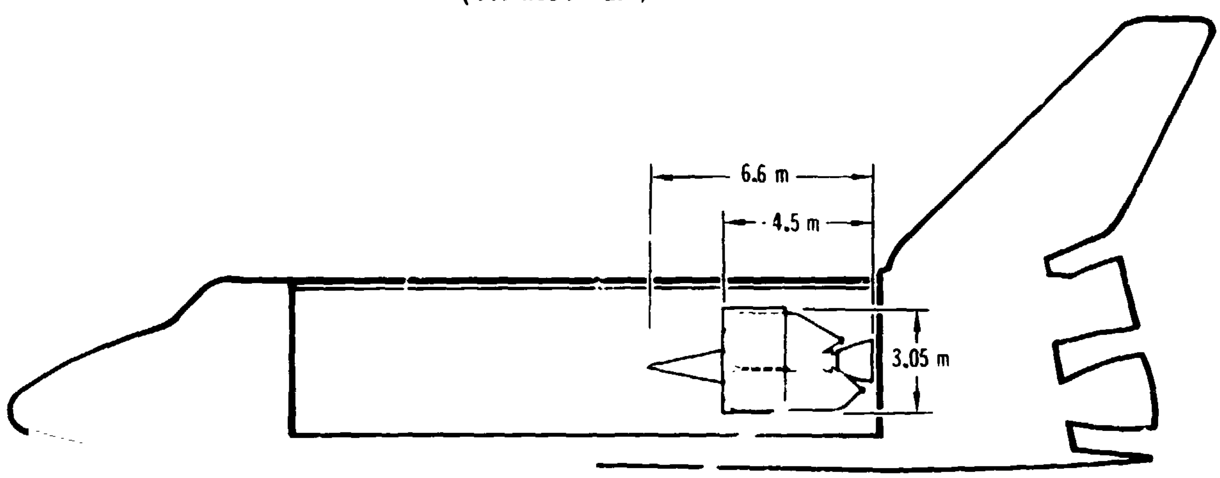


MASS	kg	(lb)	THRUST N (lb)	I_{SP} m/sec (sec)
INTERSTAGE *	86.7	(191.2)	129,900 (29200)	4311 (439.6)
INERT **	2495.0	(5501.0)		
BURNOUT	3703.0	(8164.2)		
EXPENDED	13532.0	(29833.0)		
IGNITION	17235.0	(37997.2)		

* INCLUDES 2ND STAGE SPIN TABLE
** INCLUDES AVIONICS

FIGURE 56

TYPICAL EARTH STORABLE PROPELLANT BOOSTER DESCRIPTION (TRANSTAGE)



MASS	kg	(lb)	THRUST N (lb)	I_{sp} m/sec (sec)
INTERSTAGE *	86.7	(191.2)	69980 (15733)	2955 (301.3)
INERT **	1701.0	(3750.7)		
BURNOUT	2909.0	(6413.9)		
EXPENDED	10447.0	(23032.0)		
IGNITION	13356.0	(29445.9)		

* INCLUDES 2ND STAGE SPIN TABLE
** INCLUDES AVIONICS

FIGURE 57

payload sharing. This class of booster has a wide range of applicability to DoD mission requirements. Its compactness, multiple burns capability, and payload capacity, make it a strong candidate for the PDS booster. Throughout this report the Transtage is used as the PDS booster.

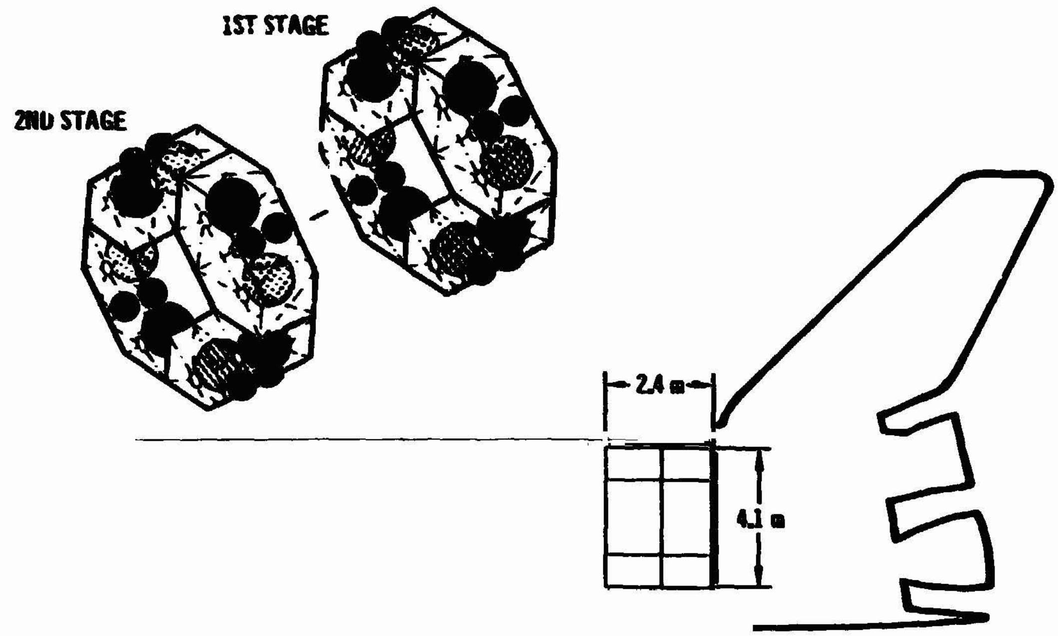
8.3 Short Length Existing Component Storable Booster - A multi-stage velocity package composed of Shuttle Auxiliary Propulsion System Components is shown to illustrate a short length, high ΔV class of storable (N_2O_4/MMH) boosters. The compact velocity package is an example of how existing components can be configured to best utilize the wide, length limited shape of the Shuttle payload bay. This maximizes opportunity for shared payload launches of the Shuttle. In contrast, the other booster classes represent relatively long, narrow upper stages because they were originally designed for ground launched, expendable boosters.

Figure 58 describes the physical and propulsion characteristics of a two stage velocity package. The two stages are identical and composed of tank, engine and flow control components being developed for the Reaction Control System (RCS) of Shuttle. More detail of this concept as it is best configured to meet DoD mission requirements is presented in Section 11. It is best suited for the high velocity-shallow flight path angle reentry missions.

8.4 Existing Solid Motor Booster - A preliminary version of the Interim Upper Stage (IUS) is presented as an example of the solid propellant class of booster. The configuration shown is necessarily preliminary since the Air Force is in the process of awarding a contract to define the final IUS characteristics. Figure 59 summarizes the characteristics of a two stage IUS. A third stage will in general be required to provide the payload spacing burns for DoD payloads. The two stage SRB will provide only the deorbit and plane change burns. This SRB concept is similar in size and total impulse to the Transtage. Therefore, many of the missions presented in this report can be performed by an SRB if a third stage or spacing is included.

In conclusion, existing boosters are available which can meet the DoD needs of the 1980's. Their performance for particular missions are presented in the next section.

TYPICAL "SHORT LENGTH" BOOSTER DESCRIPTION (SHUTTLE COMPONENT VELOCITY PACKAGE)

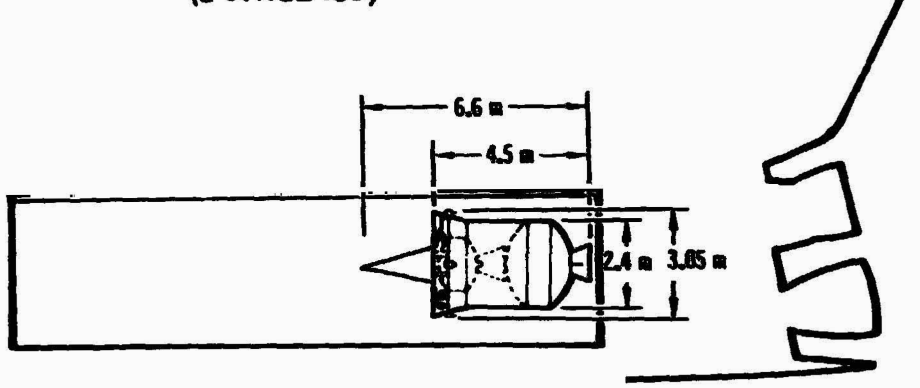


STAGE	MASS	kg (lb)	THRUST N (lb)	I_{sp} m/sec (sec)
SECOND STAGE	INTERSTAGE *	349.0 (770.0)	15984 (3600)	2834 (289)
	INERT	1316.0 (2902.0)		
	BURNOUT	2786.3 (6144.0)		
	EXPENDED	3931.0 (8667.0)		
	IGNITION	6717.3 (14811.0)		
FIRST STAGE	INTERSTAGE	45.0 (100.0)	15984 (3600)	2834 (289)
	INERT	1316.0 (2902.0)		
	BURNOUT	8078.3 (17813.0)		
	EXPENDED	3931.0 (8667.0)		
	IGNITION	12009.3 (26480.0)		

* INCLUDES AVIONICS AND 3RD STAGE SPIN TABLE

FIGURE 58

TYPICAL SOLID PROPELLANT BOOSTER DESCRIPTION
(2 STAGE IUS)



STAGE	MASS	kg	(lb)	THRUST N (lb)	I_{sp} m/sec (sec)
SECOND STAGE	INTERSTAGE *	86.7	(191.2)	62,720 (14100)	2909 (296.6)
	INERT **	648.5	(1430.0)		
	BURNOUT	1856.3	(4093.2)		
	EXPENDED	2161.9	(4767.0)		
	IGNITION	4018.2	(8860.2)		
FIRST STAGE	INTERSTAGE	–	–	186800 (32000)	2838 (289.4)
	INERT	932.9	(2057.0)		
	BURNOUT	4951.1	(10917.2)		
	EXPENDED	9144.7	(20164.0)		
	IGNITION	14095.8	(31081.2)		

* INCLUDES 3RD STAGE SPIN TABLE
** INCLUDES AVIONICS

FIGURE 59

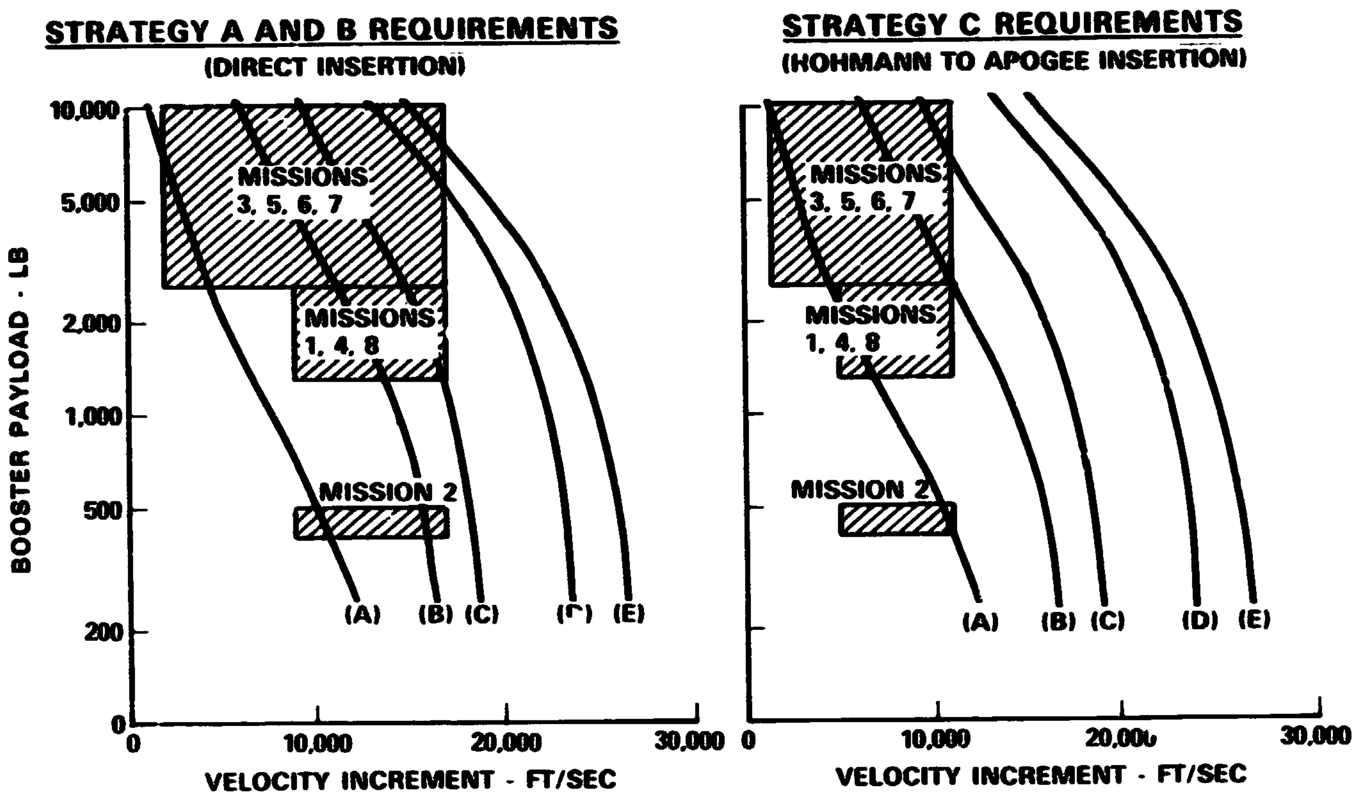
9. DELIVERY SYSTEM SIZING AND PERFORMANCE

The performance of the payload delivery system, PDS, which consists of the booster and/or bus, spin separation system, and RV's was considered as a function of reentry velocity and flight path angle. The initial study emphasis defined booster performance as a function of inertial reentry velocities between 20 and 25 kft/sec and flight path angles between -5 and -60 deg. The results of this study led to the detailed definition of Transtage performance (Section 9.2) for relative reentry velocities of 25 kft/sec and selected flight path angles. The definition of the Minibus concept to take advantage of the low ΔV requirements for shallow flight path angles followed and is described in Section 9.3. Finally, the assessment of the impulsive ΔV assumption was investigated as described in Section 9.4.

9.1 Booster Performance - The ΔV requirements from Sections 7.1 through 7.3 were compared with Delta, Transtage, and Centaur boosters described in Section 8 and the Burner IIA and Tandem Transtage. The Burner IIA is a small, solid rocket booster and the Tandem Transtage is a growth Transtage which achieves higher total impulse by increased propellant load. This was an initial screening to assess the type of booster required to meet the classified mission requirements of Reference 2. Figure 60 shows the comparison of these booster payload weight-ΔV capability for strategies A, B and C of Section 7.1. The rectangles identified by specific missions from Reference 2 represent the range of payload weights and ΔV's expected. Payload weight includes the spin separation system weight estimate. All velocity increment-booster payload combinations to the left of a booster performance line are possible. The line represents an upper bound on booster performance. For example, for strategies A and B the Transtage can provide up to approximately 10 kft/sec ΔV for a 10000-lb payload and 17 kft/sec for a 1000-lb payload. Note that strategy C ΔV requirements are less than A and B because only the deorbit burn requirements are considered. The inclusion of payload spacing burns would increase Strategy C requirements more than Strategy A.

Several conclusions are evident from Figure 60. The Centaur and Tandem Transtage have sufficient ΔV to perform the majority of the missions identified. Transtage and Delta can perform most of the low to moderate payload missions and some of the large payload-low ΔV missions. The Burner IIA can perform little of the Strategy A missions and only the small payload Strategy C missions. The upper left side of these plots represents the high velocity shallow flight path angle reentry missions. All boosters shown can meet to some degree this part of the mission requirements.

BOOSTER PERFORMANCE COMPARISON TO DoD MISSION REQUIREMENTS



(A) BURNER IIA
(B) DELTA
(C) TRANSTAGE
(D) TANDEM TRANSTAGE
(E) CENTAUR

FIGURE 60

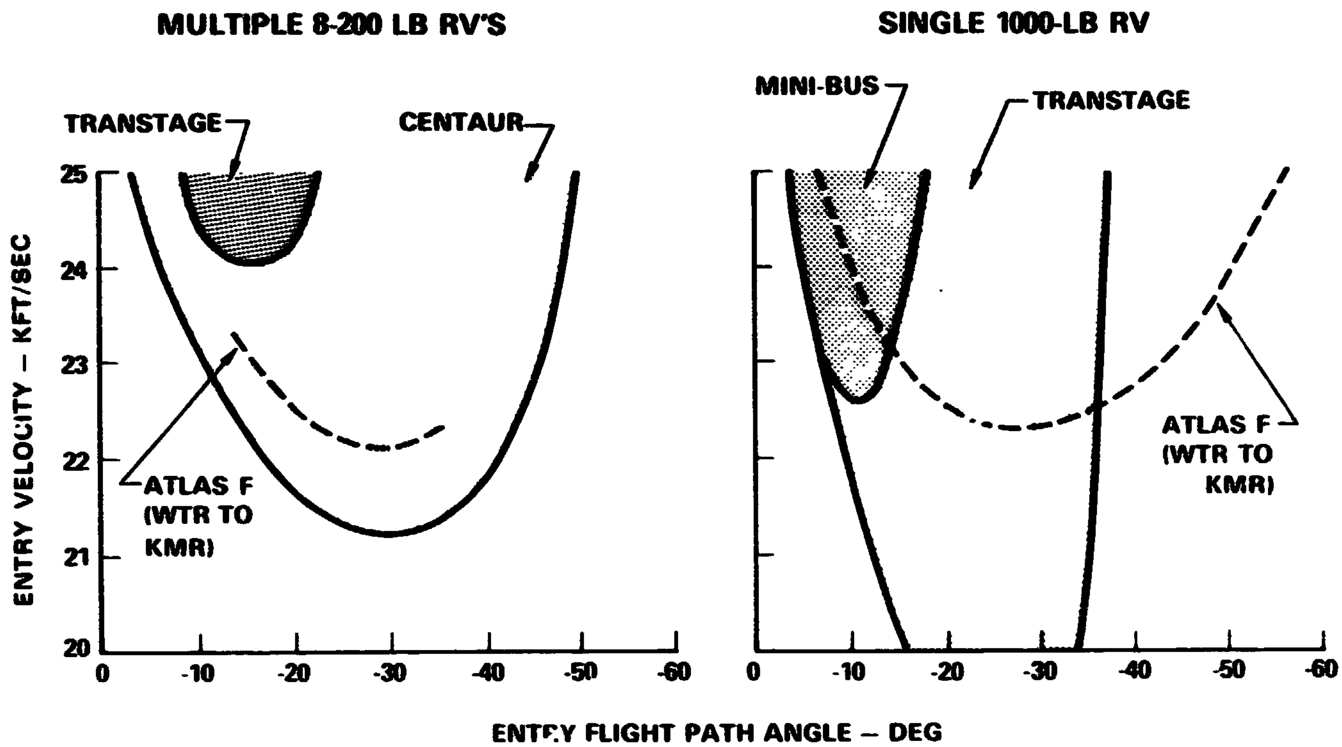
To help identify the range of reentry velocities and flight path angles for which a particular booster could be used, Figure 61 is presented. The left chart for 8 RV's weighing 200 lb each emphasizes the effect of the 90 second spacing requirement on booster performance. The booster is used to provide both the Strategy A deorbit burn and the spacing burns. Any reentry flight path angle-velocity combination within the shaded area is achievable. The left and right boundaries represent the limitations imposed by spacing burn and deorbit burn requirements, respectively. Transtage can provide a small region of high velocity-shallow flight path reentries. On the other hand, Centaur can provide a large area of the V-γ map. These areas are compared with the Atlas ground launch from VAFB to KMR for the same number and spacing of payloads. Because fo the fixed range from launch to impact, the Atlas capability is a line and not an area. (Atlas performance data were obtained from Reference 8.) Shuttle delivered payloads can achieve a wider range of reentry conditions and, in addition, higher reentry velocities compared to a typical ground launch.

The single 1000-lb RV capability, shown on the right hand chart, demonstrates again the high velocity shallow flight path angle capability achievable from Shuttle. Transtage and the Minibus concept are compared. Here the left hand boundary represents the limit for a single burn deorbit maneuver, i.e., the ballistic trajectory apogee is below the Shuttle orbit altitude for shallow flight path angle-low velocity reentry. The Transtage can achieve a wide range of reentry conditions indicating that it is also oversized for much of the conditions. The Minibus is a bus concept, described in Section 9.3,which is designed specifically to provide high velocity-shallow flight path angle reentry conditions. Note that the Atlas F can also provide specific reentry conditions over the same range as Transtage and has additional capability at steep flight path angles. The advantage of Shuttle for these reentry conditions is the capability to achieve payload reentry at other impact points such as Poker Flat.

9.2 Transtage Performance - The above analysis identified that Transtage could indeed provide high velocity-shallow flight path angle capability from a Shuttle launch. The next step was to develop in more detail the overall capability of Transtage. The approach selected involved considering only 25 kft/sec relative reentry velocities and computing the maximum payload weights, the offloading and the excess ΔV as a function of relative flight path angle at the pierce point.

HIGH VELOCITY & SHALLOW REENTRY ANGLES EASILY SIMULATED BY SHUTTLE/PDS DELIVERY

(IMPROVES UPON GROUND LAUNCH CAPABILITIES)



NOTE: 90 SEC SPACING AT PIERCE

FIGURE 61

9.2.1 Sizing Algorithms - To accomplish this task a family of PDS sizing algorithms for the booster and/or bus and the spin separation systems were developed. Figures 62 and 63 provide the spin separation system weight and length algorithms respectively. (The origin of the spin system algorithms is presented in Section 11.) The structural weight in Figure 62 includes the spin table or tube, booster attachment structure, beams and cross members. This is all the weight that is added forward of the booster or bus interface. The spin system length of Figure 63 includes the tube lengths for payloads up to 1000 lbs. The RV's do not protrude significantly from the tubes. Above 1000 lbs a spin table is used and the RV is mounted above it. The RV, therefore, contributes significantly to the spin separation system length. Typical RV lengths are given in Figure 9 of Section 3.

Five possible booster/bus design combinations were identified (Figure 64). They were:

1. Sizeable booster/sizeable bus - The spent booster separates after the out of plane and deorbit burns are completed. The deployable bus provides the payload spacing burns. Both the booster and the bus are sized to make maximum use of the Shuttle payload bay. The resultant design is a new booster and bus capable of delivering the given payload weight and number at the reentry conditions.
2. Fixed Booster/Sizeable Bus - The staging sequence is the same as option 1 above. However, the booster characteristics are fixed. The bus provides additional spacing capability over that of a single stage system. The payload weight and bus design is determined for the given number of payloads and reentry conditions.
3. Sizeable bus - The bus or booster provides propulsion for all burns, i.e., plane change, deorbit and spacing and makes maximum use of the Shuttle payload bay. The resultant design is a new booster capable of delivering the specified payload number and weight at the proper reentry conditions.
4. Fixed Bus - Offloaded - The bus function is performed by the booster. The amount of propellant required for the given payload weight, number and reentry condition is determined.
5. Fixed Bus (Maximum capability) - The bus function is again performed by a fixed booster, e.g., Transtage. For a given reentry condition, the number of payloads, the maximum payload weight is determined assuming all the propellant is used.

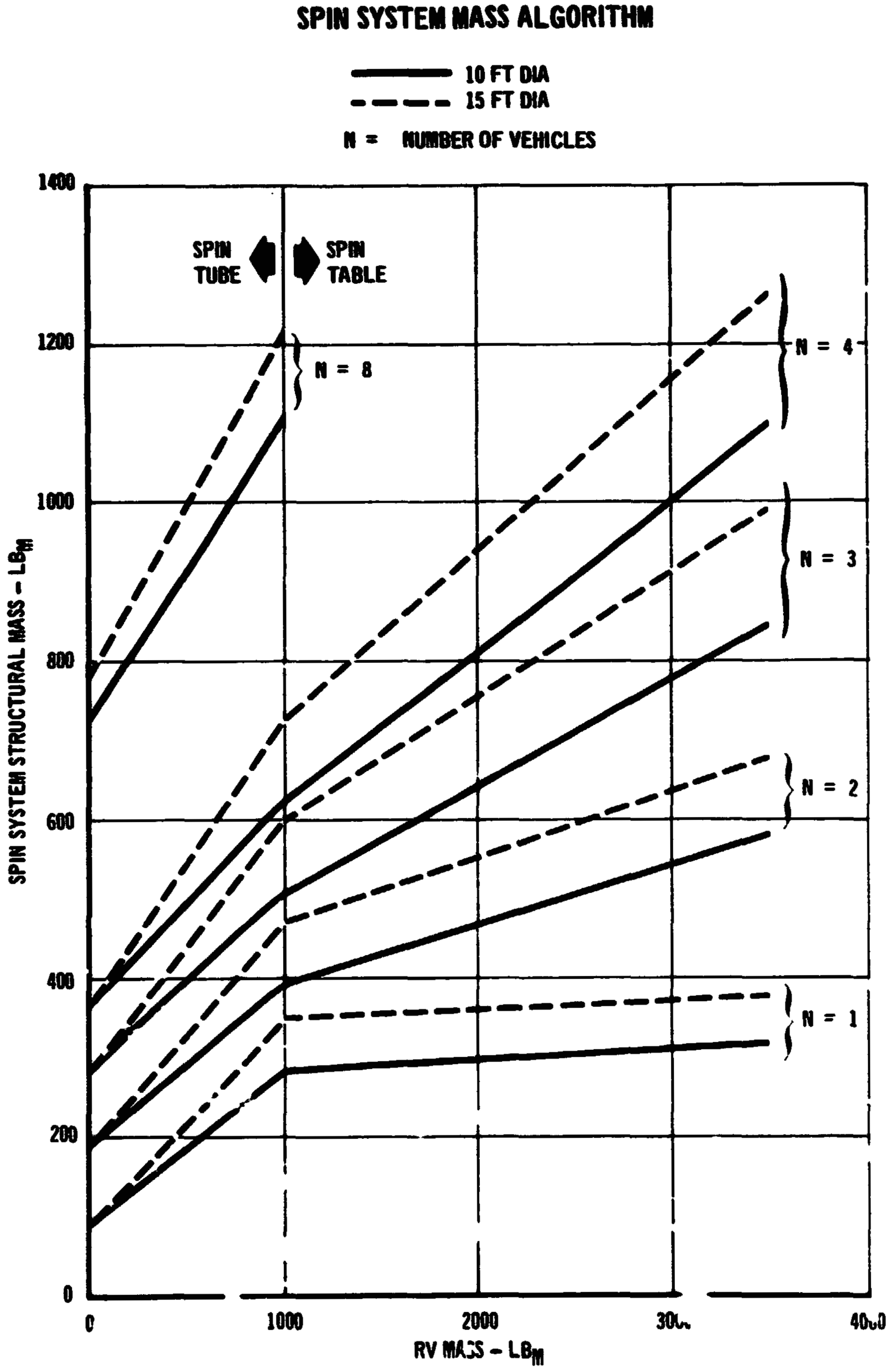

SPIN SYSTEM MASS ALGORITHM
10 FT DIA
15 FT DIA
N = NUMBER OF VEHICLES
SPIN TUBE
SPIN TABLE
N = 8
N = 4
N = 3
N = 2
N = 1
SPIN SYSTEM STRUCTURAL MASS – LB_M
RV MASS – LB_M
0
200
400
600
800
1000
1200
1400
0
1000
2000
4000

FIGURE 62

SPIN SYSTEM LENGTH ALGORITHM

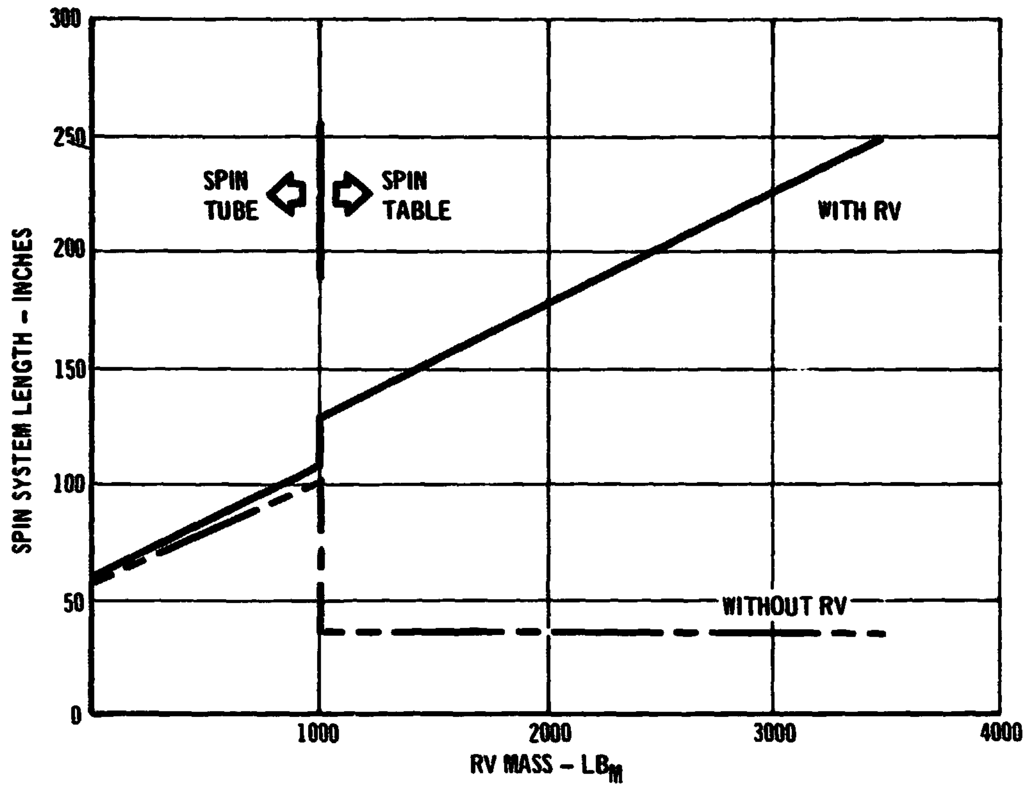


FIGURE 63

BOOSTER/BUS DESIGN OPTIONS

CASE	APPROACH	NO. PAYLOADS	WGT PAYLOADS	WEIGHT		LENGTH	
				BOOSTER	BUS	BOOSTER	BUS
1	Sizeable booster & bus	Fixed	Fixed	Calc	Calc.	Calc.	Calc.
2	Fixed booster-sizeable bus (Maximum capability)	Fixed	Calc.	Fixed	Calc.	Fixed	Calc.
3	Sizeable bus	Fixed	Fixed	--	Calc.	--	Calc.
4	Fixed bus offloaded	Fixed	Fixed	--	Calc.	--	Fixed
5	Fixed bus - (maximum capability)	Fixed	Calc.	--	Fixed	--	Fixed

FIGURE 64

Options 4 and 5 were considered in detail to establish the Transtage capabilities. The propellant quantities and spin system structural sizing were based upon the matrix solution defined in Appendix C. For option 5 both the propellant weight for each burn and the weight of each payload are unknown. For option 4, only the propellant weight is unknown.

9.2.2 Transtage Capability - Figure 65 presents the maximum payload weight the Transtage can deliver to the pierce point with 25 kft/sec relative velocity and the range of flight path angles shown on the abscissa. Spacing of 90 sec between each payload and the Transtage at pierce is assumed. An additional ΔV of 5000 fps for plane change is allowed. Therefore, the results are applicable to both KMR and Poker Flat impact. The solid lines on the chart represent maximum payload weight as a function of flight path angle for 1, 2 or 4 payloads. For example, a 2000 lb payload can be delivered with first payload flight path angles of -24, -17 and -6 deg for 1, 2 and 4 payload configurations respectively. Dashed lines represent constant PDS length which does not vary with payload number. (The payloads are mounted side by side on the spin table.) Payload weights as great as 5000-lb are anticipated and would result in 40 foot-long PDS configurations. For the more typical 1000-lb payload, PDS lengths of 25 ft are to be expected.

Note that the Transtage can deliver up to 4 1000-lb payloads with a flight path angle of -10 deg. This implies considerable excess capability for flight path angles in the -5 deg range. Figure 66 displays this excess capability in terms of excess ΔV and percent offload as a function of flight path angle. These parameters best typify excess capability. For a fixed booster, a low ΔV mission can be performed by unloading excess propellant which reduces the booster launch weight in the Shuttle payload bay. Alternatively, a full propellant load can be carried and this extra propellant used to provide additional maneuver capability, e.g., additional plane change opportunities. In either case, the data shown in Figure 66 reflects the excess capability of Transtage to perform the shallow flight path angle reentry. For example, a single 1000-lb payload at -5 deg flight path angle requires as much as 72% of the propellant to be offloaded, or a full propellant load provides 8700 fps excess ΔV. This is a very inefficient use of an existing booster and points out the need for a smaller booster for this type of mission.

The performance data shown in Figure 65 can also be plotted as a function of total impulse required. To some extent this provides the capability to consider other

MAXIMUM PERFORMANCE CAPABILITY OF TRANSTAGE

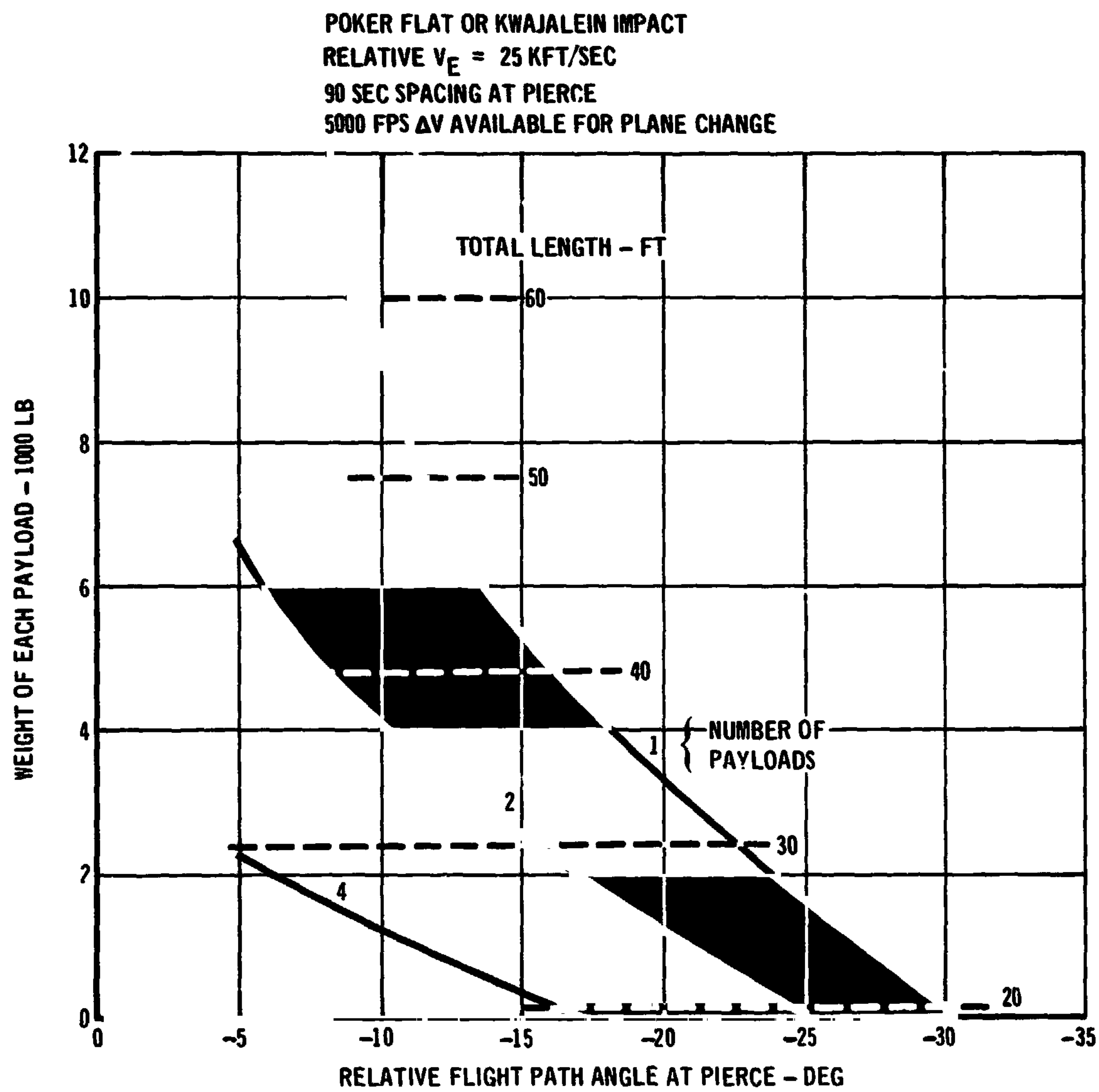


FIGURE 65

TRANSTAGE OFFLOADING & EXCESS ΔV FOR 1000 LB PAYLOADS

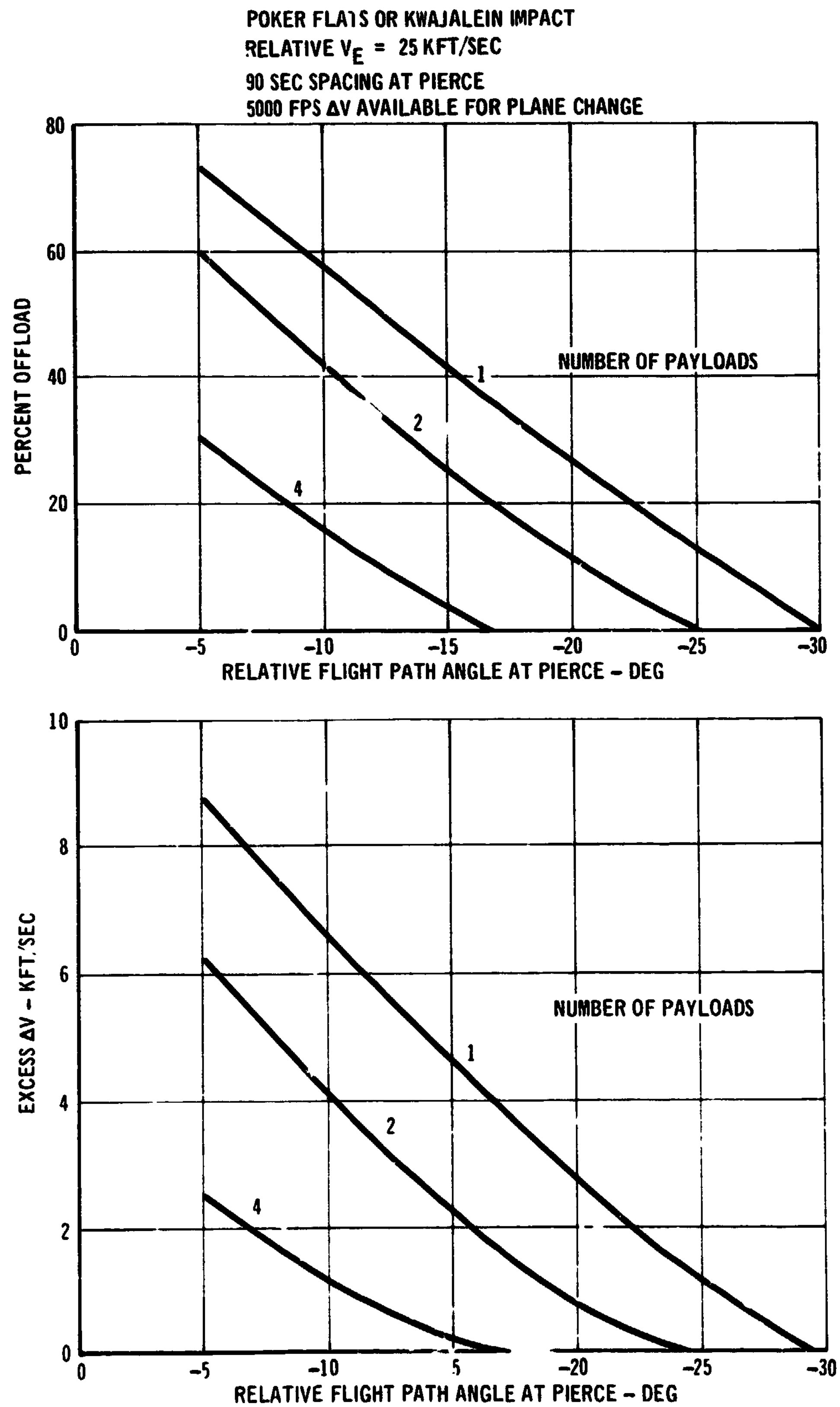


FIGURE 66

booster systems of comparable total impulse for these missions. Figure 67 presents the total weight of equal weight payloads as a function of total impulse required for 1, 2 or 4 payloads. The first payload reenters at 25 kft/sec and the flight path angles of -5, -10, -20 or -30 deg as identified on the figure. The curves change little for either Pr r Flat or KMR entry so the data is applicable to either. A 5000 fps ΔV reserve is sufficient for at least one plane change maneuver at either location. The zero total payload weight corresponds to the Transtage with no added payload. Its burnout weight is 3751 lb. This is the minimum total impulse condition. Maximum total impulse of 6939 klb-sec corresponds to a full propellant load. For a given total payload, shallow flight path angles require less total impulse than steep angles. For instance, for two 1000-lb payloads, the -5 and -20 deg flight path angle total impulse requirements are 2700 and 6300 klb-sec, respectively. Note that the -30 deg flight path angle capability is minimal for only one payload.

Figures 68 and 69 present additional data on the total launch weight and length of the PDS as a function of total impulse required. The Transtage propellant is assumed to be offloaded to reduce launch weight in Figure 68. The minimum launch weight points correspond to no payload and a Transtage burnout weight of 3751. Maximum weight points correspond to the full propellant load and represent the maximum capability of Transtage. The launch length of Figure 69 is the combined length of Transtage, the spin separation system, and the RV's. Minimum length of 14.83 feet is the Transtage alone length. The curves of Figures 68 and 69 allow estimates of the weight and length of the launch configuration for the total impulse requirements of various missions.

9.3 Minibus Sizing and Performance - The performance characteristics of each of the Minibus concepts described in Section 11 was determined for typical deorbit trajectories such as shown in Figure 70. A plane change burn requiring 1200 fps ΔV was assumed to precede the deorbit burn at point 1. (This is somewhat low with respect to the plane change ΔV for the example cases of Section 16.) Payload deployment was assumed at point 2, and a bus spacing burn at point 3. Burns 1 and 2 always are spaced 7.5 minutes in time. The resulting time spacing of the payload and bus at pierce is 90 seconds.

The performance comparisons were achieved by fixing the inertial reentry velocity at 25000 ft/sec, and parametrically investigating the inertial reentry flight path angle. The data of Section 9.2 was in terms of relative conditions at the

TRANSTAGE PERFORMANCE PARAMETRIC – TOTAL WEIGHT OF PAYLOADS

- RELATIVE V_E = 25000 FT/SEC
- 5000 FPS ΔV RESERVE

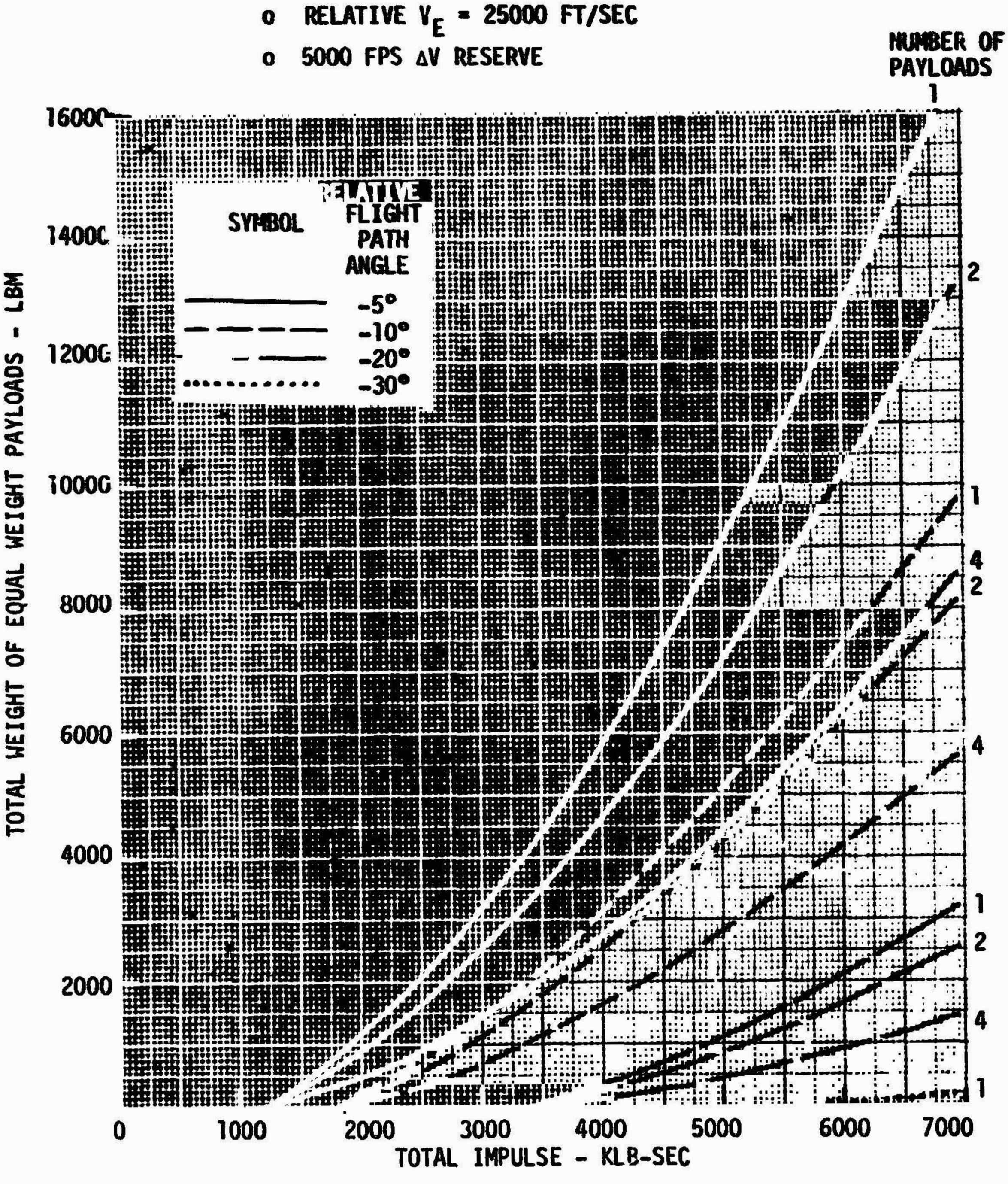


FIGURE 67

C.2

TRANSTAGE PERFORMANCE PARAMETRIC – TOTAL LAUNCH WEIGHT

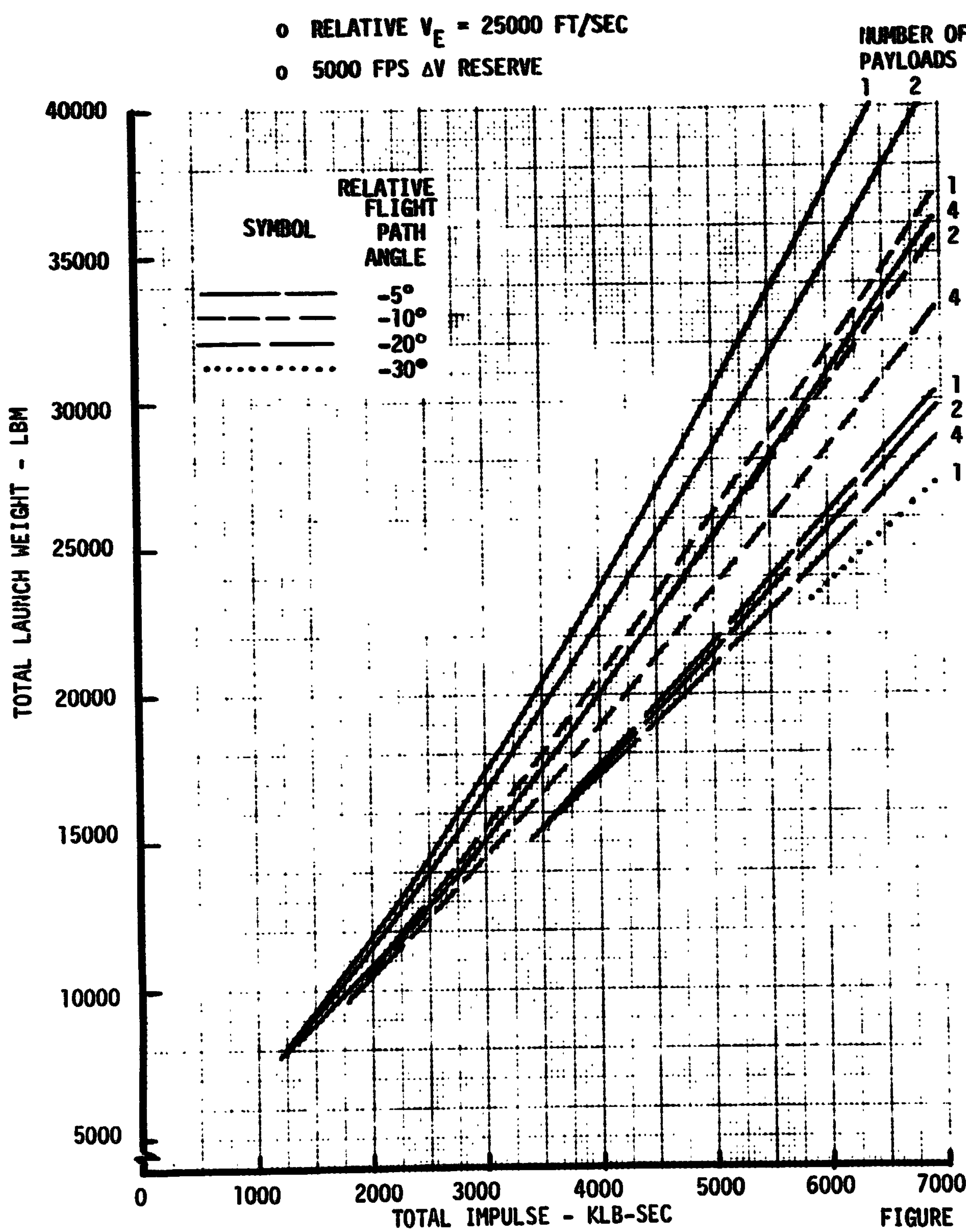


FIGURE 68

TRANSTAGE PERFORMANCE PARAMETRIC – TOTAL LAUNCH LENGTH

- RELATIVE V_E = 25000 FT/SEC
- 5000 FPS ΔV RESERVE

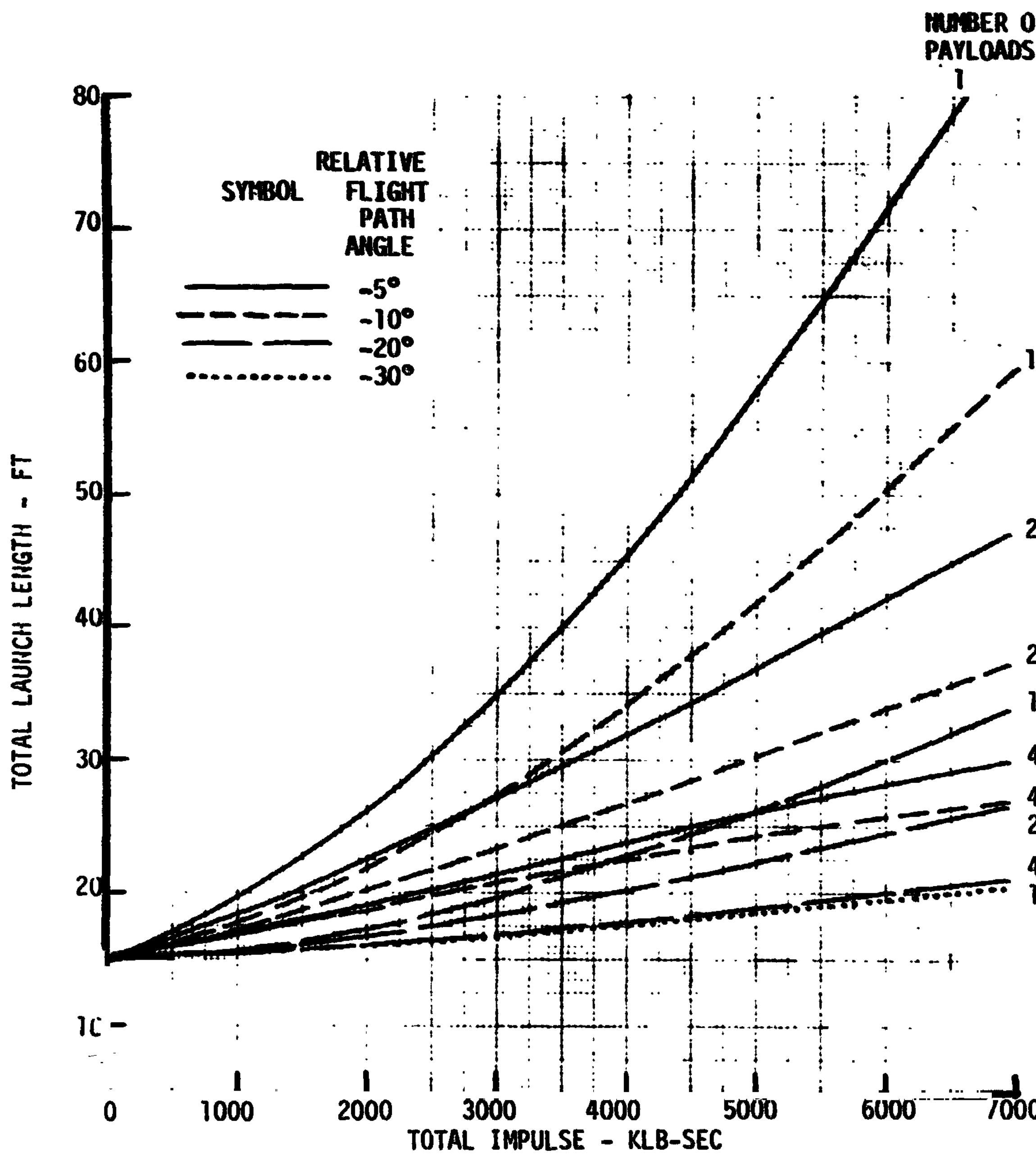
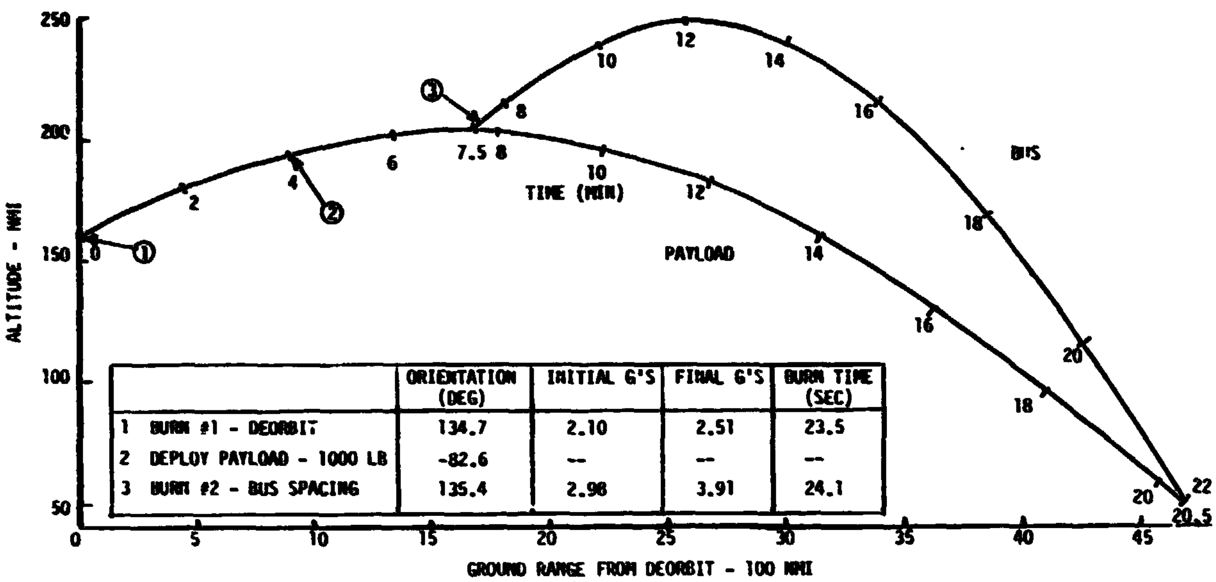


FIGURE 69

EXAMPLE DEORBIT TRAJECTORIES

INERTIAL V_E = 25 KFT/SEC, γ_E = -5 DEG, 1000 LB PAYLOAD



	ORIENTATION (DEG)	INITIAL G'S	FINAL G'S	BURN TIME (SEC)
1 BURN #1 - DEORBIT	134.7	2.10	2.51	23.5
2 DEPLOY PAYLOAD - 1000 LB	-82.6	--	--	--
3 BURN #2 - BUS SPACING	135.4	2.98	3.91	24.1

FIGURE 70

pierce point. Therefore, the Transtage results presented for comparison here differ slightly from those of the previous section. For a fixed payload of 1000 lb and shallow flight path angles, propellant must be offloaded. The percentage offloading required as a function of reentry angle is given in Figure 71. Zero offloading corresponds to the maximum capability of the concept. A high percentage offloading means that the Minibus is oversized for the mission and the lower total impulse concepts are more practical. For these results both the OMS derivative and the Transtage are oversized for reentry flight path angles shallower than -20 deg.

Instead of offloading propellant, the excess propellant can be used for plane change maneuvers before deorbit. If this is done, percent offload is translated into excess ΔV capability as shown in Figure 72. As for offloading, zero excess ΔV occurs at the Minibus maximum capability. To achieve the desired approach azimuth at Kwajalein on two consecutive passes, a very high excess ΔV capability is required and only the OMS derivative and the IUS can achieve this at shallow flight path angles. On the other hand, 2 consecutive passes at Poker Flat from Vandenberg are well within the capabilities of any of the concepts for shallow entry angles.

Figure 73 summarizes the offloaded propellant requirements for each of the concepts assuming a flight path angle of -5 deg at 25 kft/sec velocity. The total launch weight is less than 8500 lb for all concepts. However, the length varies greatly because of the bus length and method of attaching the RV to the bus. RCS derivatives 3 and 4 are minimum length because the RV and spin table can be submerged in the bus. The Transtage is a maximum length because the RV and spin table must be mounted atop the Transtage. The resulting packaging in the Shuttle payload bay for RCS derivative 3 and the Transtage are shown in Figure 74. (The Minibus could also be positioned in the front of the payload bay.) The remaining lengths for shared payloads are 51 and 36-ft for the Minibus and Transtage, respectively. Also shown on the figure is the relative cost for just the launch cost using the Minibus and a Transtage. A rather simple costing procedure based on used cargo bay volume shows the advantage of the Minibus. The recurring cost for the Minibus would be less than the Transtage and hence the launch cost is relatively low for the Minibus. The Minibus has other applications where the energy requirements for certain orbital missions are not excessive and its development cost could be shared.

Excess ΔV capability can also be used to deorbit larger weight payloads. For a full propellant load, the maximum payload weight possible as a function of entry angle is shown in Figure 75. These results include the change in weight of the

MINIBUS SIZING/PERFORMANCE STUDY
OFFLOADING REQUIREMENTS

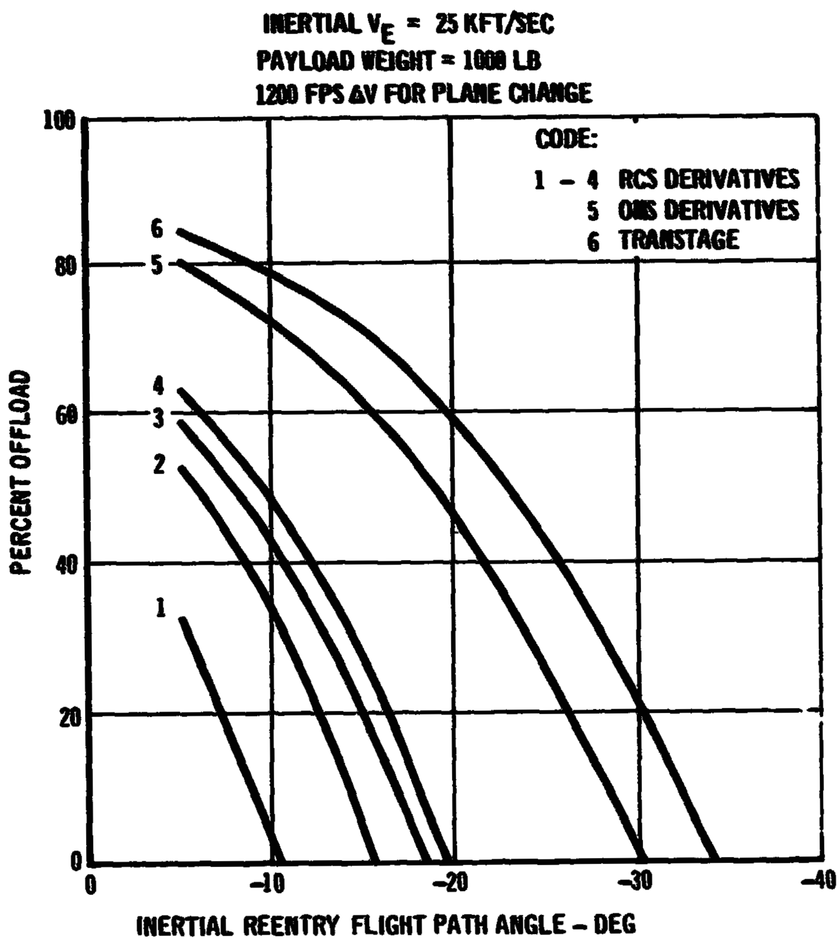


FIGURE 71

MINIBUS SIZING/PERFORMANCE STUDY
EXCESS ΔV CAPABILITY

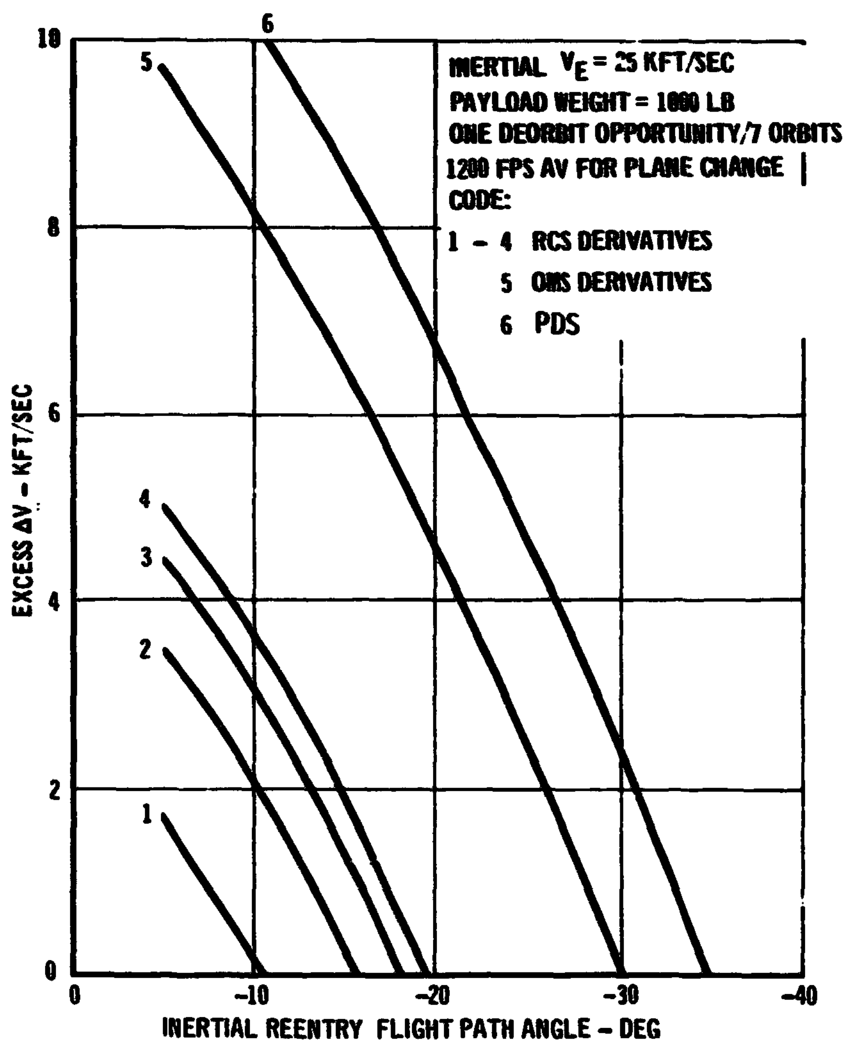


FIGURE 72

SUMMARY OF MINIBUS LAUNCH CHARACTERISTICS

INERTIAL V_E = 25 KFT/SEC; γ_E = -5 DEG; OFFLOADED

MINIBUS CONCEPT →	1	2	3	4	5	6
INERT WEIGHT (LB)	1025	1751	2477	3202	2418	3751
PROPELLANT WEIGHT (LB)	1432	2028	2623	3218	3221	3474
SPIN SYSTEM WEIGHT (LB)	270	270	270	270	270	270
RV WEIGHT (LB)	1000	1000	1000	1000	1000	1000
TOTAL LAUNCH WEIGHT (LB)	3727	5049	6370	7690	6909	8495
% OFFLOADED	34	53	60	63	73	85
SHUTTLE PAYLOAD BAY LENGTH USED (FT)	14	14	9	9	16	24

CODE:

1 - 4 RCS DERIVATIVES

5 OMS DERIVATIVES

6 TRANSTAGE

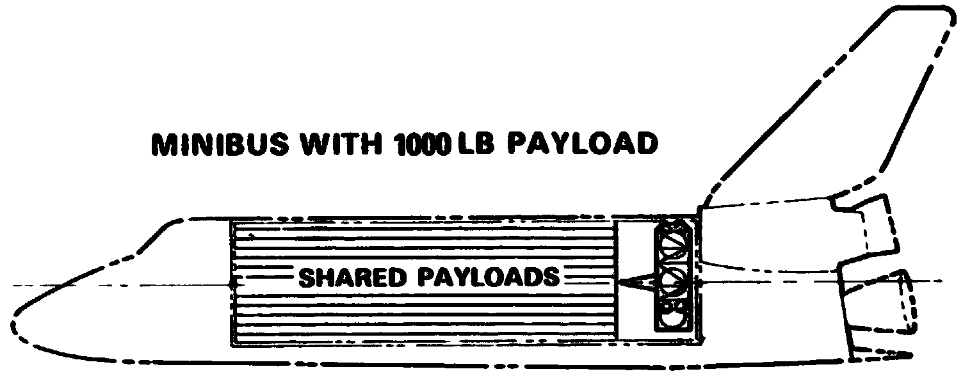
FIGURE 73

COMPARISON OF MINIBUS AND TRANSTAGE SHUTTLE PAYLOAD BAY UTILIZATION

LAUNCH COST

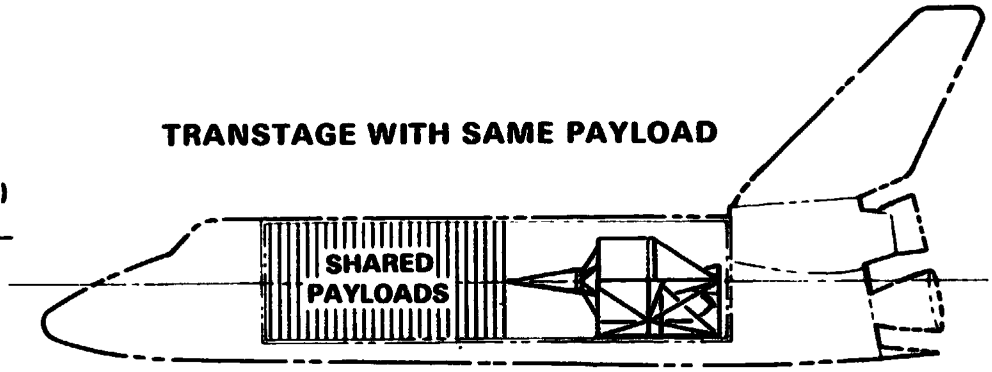
SHUTTLE: 1/6 (10$\overline{M}$)
BUS: 2 $\overline{M}$
$3.7 $\overline{M}$

MINIBUS WITH 1000 LB PAYLOAD



TRANSTAGE WITH SAME PAYLOAD

SHUTTLE: 1/2 (10$\overline{M}$)
BUS: 4 $\overline{M}$
$9 $\overline{M}$



NOTE: 1971 DOLLARS

FIGURE 74

MINIBUS SIZING/PERFORMANCE STUDY
MAXIMUM PAYLOAD CAPABILITY

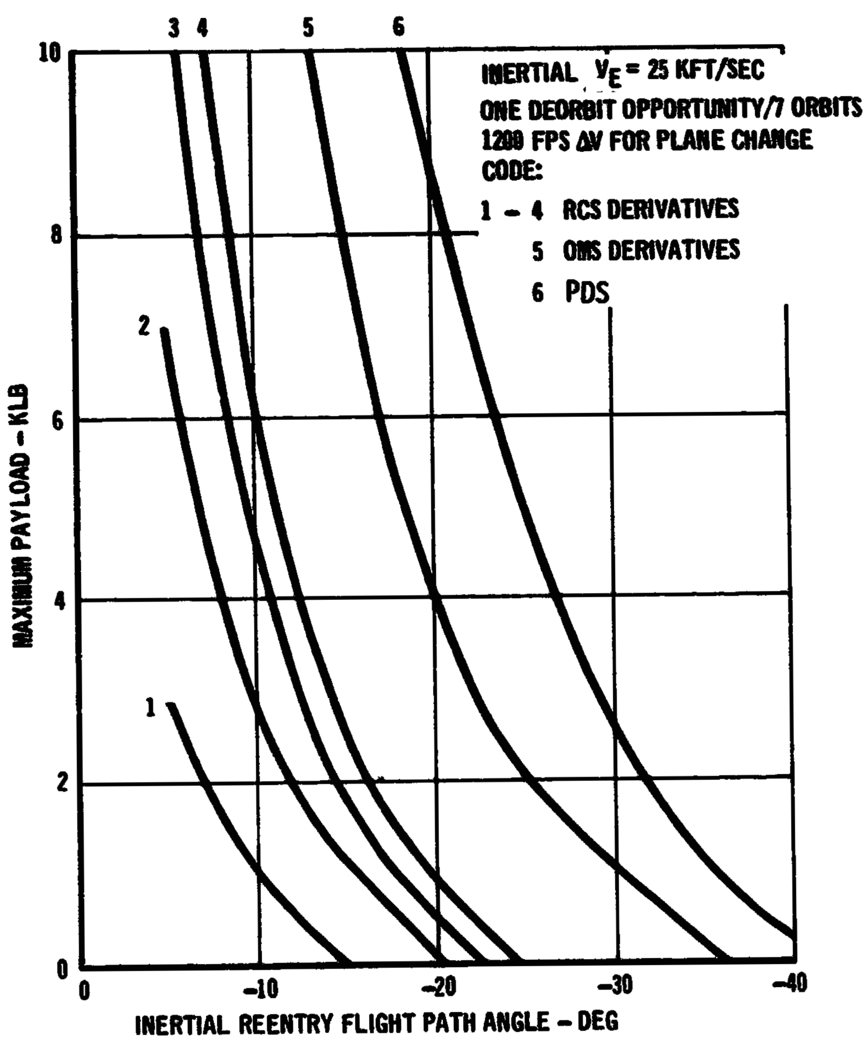


FIGURE 75

spin separation system with payload weight. As would be expected, the OMS derivative and IUS have a very high payload weight capability at moderate flight path angles. For angles shallower than -10 deg, concepts 3 and 4 also have payload capability greater than 10,000 lbs.

The conclusions of the Minibus-Transtage comparisons are:

1. The Transtage is significantly oversized for high velocity - shallow flight path angles.
2. RCS derivative packages can be configured which significantly increase shared payload bay length.

9.4 Finite Burn Analysis - All the analyses presented in this report are based upon impulsive ΔV assumptions, i.e., the ΔV is applied instantaneously at a point in the trajectory. This corresponds to a booster with very high thrust over a short time interval. For boosters such as the Transtage and Centaur maximum burn times are approximately 450 sec. Therefore, the purpose of this section is to compare impulsive ΔV results with finite burn analyses to define the range of validity of the impulsive ΔV analyses for Transtage and Centaur.

Finite burn analyses were received from Duane Dugan of Ames Research Center, Reference 9. The most severe limitation of the impulsive ΔV assumption occurs for the deorbit burn of Strategies A or B at steep flight path angles and large payloads. However, this conclusion is dependent upon the thrust vector pointing logic used. The angle between the velocity vector and the thrust vector in this analysis was set equal to the angle that an impulsive ΔV burn requires to achieve the desired velocity and flight path angle. For a long burn time low thrust applications the thrust vector becomes aligned with the velocity vector at the end of the burn. Different thrust vector orientation will provide different results than described here. As a consequence, each special case should be considered individually.

For Strategy A deorbits, the impulsive ΔV assumption is valid to approximately -20 deg flight path angle reentries for the velocities of interest. For steeper flight path angles, reworking of the guidance law for thrust vector aiming is required before an assessment of the applicability of the impulsive analysis can be made.

For Strategy C both Transtage and Centaur can provide the complete range of flight path angles and velocities of interest for payloads in the 1000 to 2000 lb class. Figures 76 and 77 present these data as a function of the orbit apogee radius. For a fixed flight path angle, more propellant is required to achieve the higher apogee or higher velocity conditions. If a particular Strategy A maneuver cannot

STRATEGY C FINITE BURN CAPABILITY OF TRANSTAGE

PAYLOAD = 1000 LB

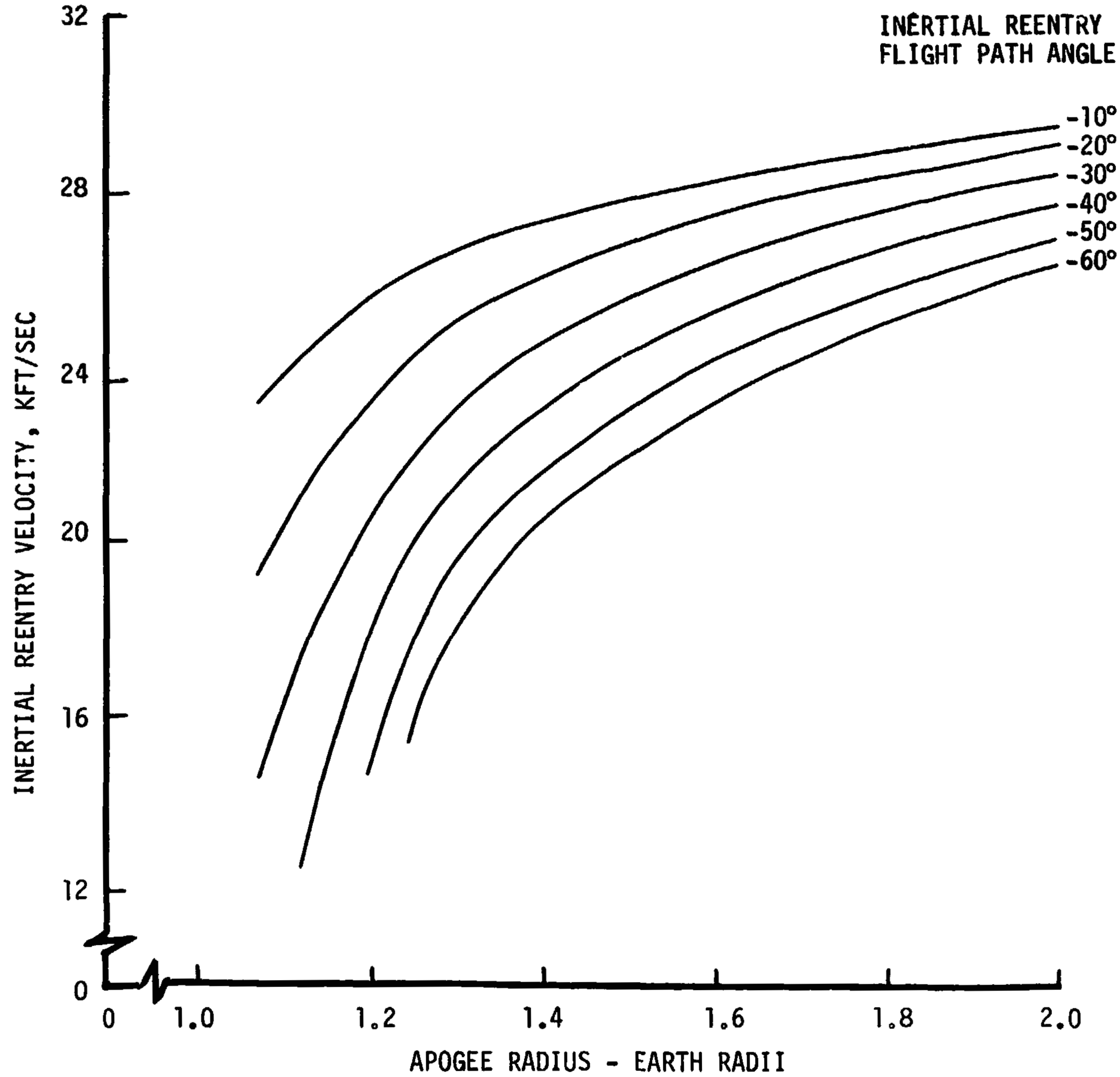


FIGURE 76

STRATEGY C FINITE BURN CAPABILITY OF CENTAUR

PAYLOAD = 2000 LB

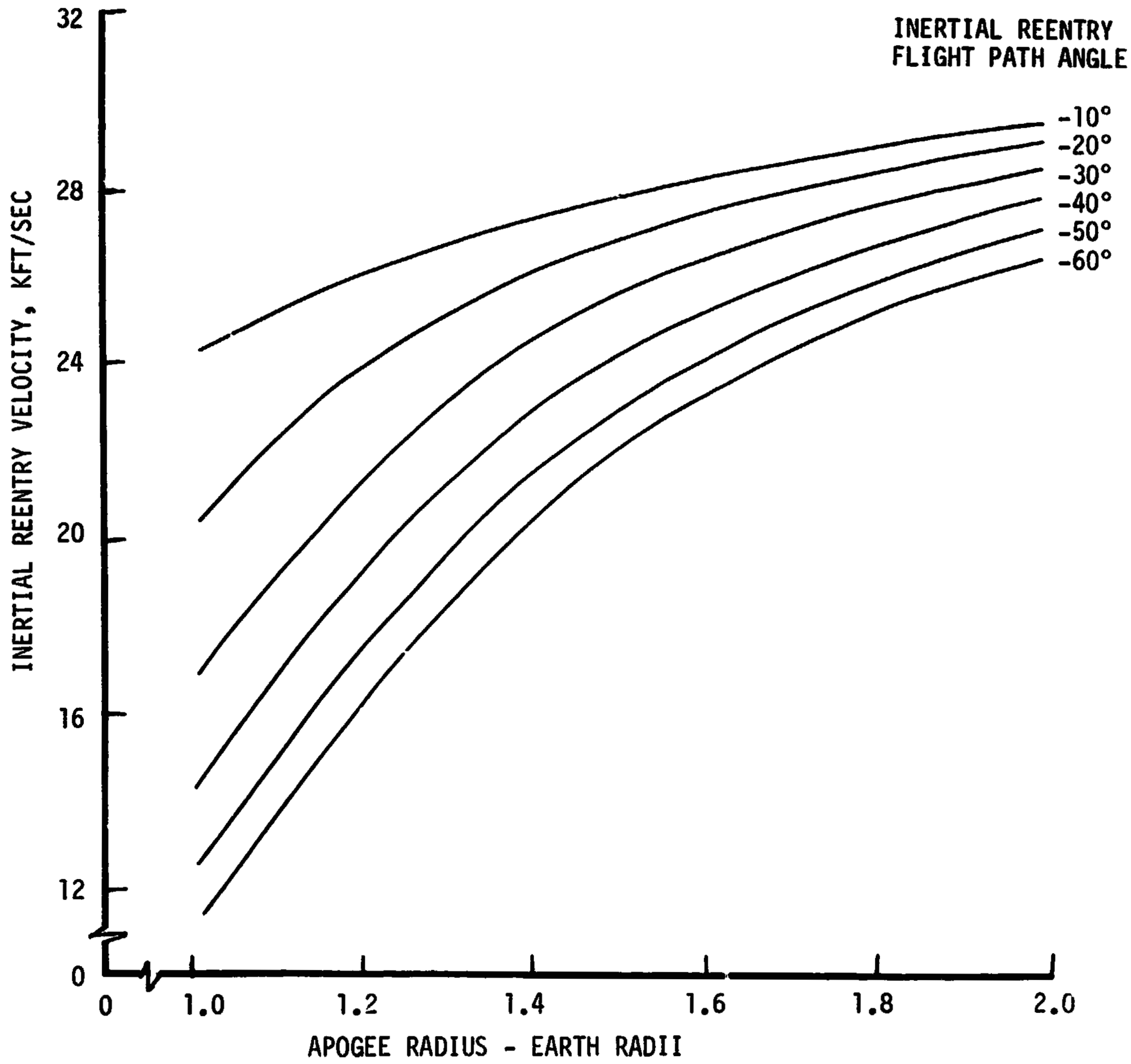


FIGURE 77

be achieved because of finite burn limitations, then a Strategy C deorbit would be required. Detailed data on the effects of finite burn are contained in Reference 9. In conclusion, finite burn analysis indicates a need for a high thrust booster to achieve Strategy A deorbit maneuvers for steep flight path angles. The alternate to this is a Strategy C maneuver which is easily performed by Transtage class boosters.

10. DELIVERY SYSTEM ACCURACY

10.1 Navigation Error Dispersions - The Mathematical Physics Branch of the NAS provided navigation error covariance matrices for use in the Shuttle payload dispersion studies. It was assumed that the Shuttle would be tracked, using S-band, by either the ground network (STDN) or a relay satellite (TDRSS).

Figure 78 contains the typical navigation error covariance matrices for two navigation configurations. These matrices are representative of the following cases:

(1) Good tracking by the ground network (i.e., tracking within two revs of the rendezvous sequence)

(2) Moderately good tracking by the ground network (i.e., tracking within 6-8 revs of the rendezvous sequence)

These data are applicable for Shuttle orbiting at altitudes of 150 NMI or above. The diagonal terms represent the square of the error component applicable to this navigation analysis.

Figure 79 shows the results of a preliminary navigation analysis for the first payload of a typical deorbit trajectory such as that shown in Figure 32 of Section 16. The two Shuttle navigation situations described above are represented.

A considerable difference in initial error occurs as a result of tracking conditions. (δX_o, δY_o, δZ_o correspond to downrange, crossrange, and altitude error, respectively. $\delta \dot{X}_o$, $\delta \dot{Y}_o$, $\delta \dot{Z}_o$ are the corresponding velocity error components.) The downrange error component, δX_o, at launch has a one to one correspondence with the error at the pierce point. The pierce point is merely displaced in range by the same amount as the position error at launch. As a consequence, for the tracking conditions shown, the pierce point dispersion due to this error source is small. The crossrange error component, δY_o, is large and results in appreciable errors at the pierce point for the moderate tracking situation. The sensitivity used in this case is a function of the total angular difference from payload deorbit burn to the pierce point. It has a maximum magnitude for orbits with 180 degrees between launch and pierce and zero for orbits with 90 degree angular displacement. The dispersion is approximately $\delta Y_o \cos \theta$ where θ is the angular displacement between launch and pierce. The altitude position error δZ_o affects the range error

NAVIGATION ERROR COVARIANCE MATRICES

(1) 1σ - COVARIANCE MATRIX FOR GOOD TRACKING BY GROUND NETWORK (I.E., WITHIN 2 REVS)
UVW ORBIT PLANE COORDINATES, FT AND FPS.

$$\begin{bmatrix} .275671+05 & -.802681+05 & .294703+04 & .932597+02 & -.177456+02 & -.126507+01 \\ & .799405+06 & -.197430+05 & -.795606+03 & .575508+02 & -.106974+03 \\ & & .116392+05 & .182926+02 & .128867+01 & -.334140+01 \\ & & & .803689+00 & -.680565-01 & .977056-01 \\ & & & & .240196-01 & .136106-02 \\ & & & & & .525264-01 \\ \text{SYMMETRIC} & & & & & \end{bmatrix}$$

3 RSS POSITION, VELOCITY - 2747 FT, 2.8 FPS

(2) 1σ - COVARIANCE MATRIX FOR MODERATELY GOOD TRACKING BY GROUND NETWORK (I.E., WITHIN 6-8 REVS)
UVW ORBIT PLANE COORDINATES, FT AND FPS

$$\begin{bmatrix} .76011313+06 & -.56621790+07 & -.34966330+06 & .68314555+04 & -.29091973+03 & -.21568041+02 \\ & .42285730+08 & .26182837+07 & -.51019620+05 & .21693150+04 & .16223562+05 \\ & & .17036212+06 & -.31603293+04 & .13369130+03 & .10912055+02 \\ & & & .61559055+02 & -.26171425+01 & -.19590073+00 \\ & & & & .11370485+00 & .81488845-02 \\ & & & & & .23023495-02 \\ \text{SYMMETRIC} & & & & & \end{bmatrix}$$

3 RSS POSITION, VELOCITY - 19722 FT, 23.6 FPS

FIGURE 78

PAYLOAD PIERCE POINT DISPERSIONS DUE TO TRACKING ERRORS

ERROR SOURCE	1σ ERROR		SENSITIVITY	DOWNRANGE NM - 1σ		SENSITIVITY	CROSSRANGE NM - 1σ	
	MOD. GOOD TRACKING (6-8 REVS)	GOOD TRACKING (2 REVS)		MOD. GOOD TRACKING	GOOD TRACKING		MOD. GOOD TRACKING	GOOD TRACKING
δX_0	872 FT	166 FT	1 FT/FT	.144	.0273	0	0	0
δY_0	6500 FT	894 FT	0	0	0	-.958 FT/FT	-1.025	-.141
δZ_0	412 FT	108 FT	.00557 NM/FT	2.3	.602	0	0	0
$\delta \dot{X}_0$	7.8 FT/SEC	.89 FT/SEC	6.4 NM/FT/SEC	50	5.7	0	0	0
$\delta \dot{Y}_0$	.337 FT/SEC	.155 FT/SEC	0	0	0	.0408 NM/FT/SEC	.0138	.00632
$\delta \dot{Z}_0$	.048 FT/SEC	.229 FT/SEC	.588 NM/FT/SEC	.0282	.135	0	0	0

FIGURE 79

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OF POOR QUALITY

at pierce by changing the orbital parameters of the trajectory. A derivation of the sensitivity of pierce point range to launch altitude error is provided in Appendix D. With moderate tracking, the downrange error is 2.3 NMI. Better tracking reduces this error component considerably. This is the most significant error component of the three position errors.

Of the three velocity tracking errors, $\delta\dot{X}_0$, $\delta\dot{Y}_0$, $\delta\dot{Z}_0$, the downrange component, $\delta\dot{X}_0$, shows the most sensitivity. This sensitivity is derived in Appendix D and is dominated by the sensitivity of trajectory range to initial velocity. Tracking within .15 ft/sec would be required to reduce this component to less than 1 NMI. The out of plane velocity error, $\delta\dot{Y}_0$, results in a heading angle error at launch. The resultant pierce point range dispersion is maximum 90 deg from launch and minimum 180 deg from launch. The error is approximately $(\delta\dot{Y}_0/V)\sin\theta$ where V is the ballistic trajectory velocity at the launch point and θ is the angular displacement between launch and pierce. For high velocity deorbit trajectories this component is small. The tracking error for vertical velocity has a moderate sensitivity which is derived in Appendix D. However, the dispersion at pierce is small because the tracking error at launch is small.

In conclusion, the navigation error which results in the most significant range dispersion at pierce is the downrange velocity error. The sensitivity of this error component varies strongly with the flight path angle desired at pierce as demonstrated in Figure 80 for a typical Poker Flat reentry at 25 kft/sec. The sensitivity decreases by an order of magnitude between -5 and -40 deg flight path angle. For instance, a 1 ft/sec tracking error for a -5 deg and -40 deg reentry results in a 5.7 and .50 NMI dispersion at pierce, respectively.

10.2 Navigation System Accuracy - Figure 81 summarizes the navigation system error sources and resultant pierce point dispersions. The payload navigation system considered was the Delco Carousel V IMU of Reference 10. Transfer alignment assumed alignment maneuvers of the Shuttle of 1 g_0 with acceleration matching. The downrange and crossrange dispersions due to alignment and drift are small with respect to the dispersions due to initial position and velocity errors described previously. For deorbit maneuvers requiring higher accelerations and PDS maneuver rates, the alignment errors will increase. However, the resulting dispersions will, in general, be small with respect to dispersions due to tracking errors.

For multiple payload deorbits, subsequent payloads deployed will suffer from the same navigation error sources. The deorbit burn uncertainty will be compensated

SENSITIVITY OF PIERCE POINT RANGE TO LAUNCH VELOCITY

- o POKER FLAT REENTRY
- o RELATIVE REENTRY VELOCITY = 25 KFT/SEC
- o RELATIVE REENTRY AZIMUTH = 60 DEG
- o DEORBIT BURN ALTITUDE = 160 NMI

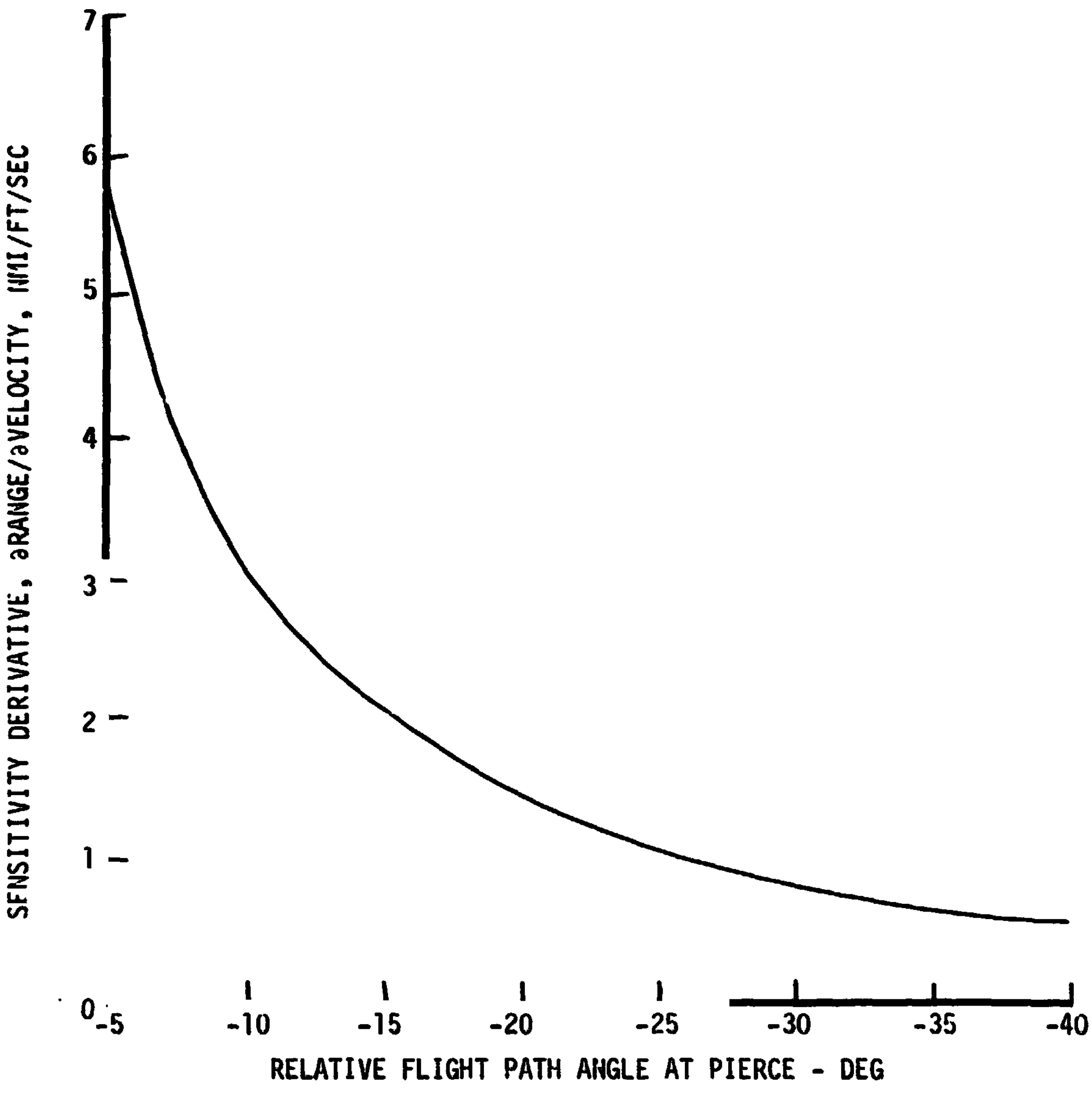


FIGURE 80

PAYLOAD PIERCE POINT DISPERSIONS DUE TO GUIDANCE SYSTEM ERRORS

INPLANE ERROR SOURCE	(1σ)	SENSITIVITY	DOWNRANGE (1σ) (NMI)
PLATFORM ALIGNMENT	97 μRAD	1450 NM/RAD	.141
TRANSFER* ALIGNMENT	86.7 μRAD	1450 NM/RAD	.126
GYRO DRIFT	6.83 μRAD	1450 NM/RAD	.0099

OUT OF PLANE ERROR SOURCE	(1σ)	SENSITIVITY	CROSSRANGE (1σ) (NMI)
PLATFORM ALIGNMENT	315 μRAD	85.2 NM/RAD	.0268
TRANSFER* ALIGNMENT	86.7 μRAD	85.2 NM/RAD	.00738
GYRO DRIFT	6.83 μRAD	85.2 NM/RAD	.000581

FIGURE 81

for by IMU acceleration measurements but the additional payloads will suffer from uncertainties in their own deployment ΔV's. They will also experience additional IMU drifts. It is conjectured that these will not be excessive. In summary, for reentries at shallow flight path angles and high speeds, the pierce point dispersions will be larger than 1 NMI for typical tracking errors. The navigation system errors will be small compared to the tracking errors.

11. BUS DESIGNS AND VEHICLE INTERFACES

The general requirements for the Shuttle Payload Deployment System, PDS, were defined in Reference (2). The PDS designs which were developed to satisfy these requirements are presented in Sections 11.1 through 11.3. The Shuttle payload interfaces are described in Section 11.4.

A brief summary of the PDS requirements is provided in Figure 82. The DoD philosophy implied by these requirements is one of maintaining operation approximating ground launch. A large number of payloads with a range of weights and sizes must be accommodated. This implies that a single PDS design may be oversized for many mission requirements. In addition, this requirement favors liquid propellant system because of the multiple burn requirements for a wide range of total impulses.

Access to the payloads for servicing must be provided prior to launch. During the Shuttle orbit, automatic checkout of the payloads will be necessary with data transmittal to ground stations for system readiness verification. No in orbit manual checkout is anticipated or desirable.

The 1 kw power and cooling requirements from Shuttle are for each payload. For eight payloads these requirements can become excessive. This is also true for the communications interface requiring 5 data links per payload. In general, however, only a few fully instrumented RV's are deployed on a given mission.

The attitude control and separation system requirements for the multiple RV's are analogous to ground launch systems. For a liquid propellant PDS, the attitude control can be maintained by the booster, e.g., Transtage. For solid rocket motor PDS additional chemical propellant attitude control systems are required. The separation of the RV's from the booster or bus can be accomplished using either spin tubes or tables analogous to those used on RVTO-2A, Reference 11. An alternate approach to spin stabilization using spin jets on each RV is also feasible but not used in this study. This approach could reduce PDS launch weight for the smaller RV's.

The PDS needs a propulsion system to deorbit from the Shuttle orbit, change the orbital plane, and provide spacing between multiple RV's. Because of the potential for a large number of significant ΔV burns a liquid propellant system provides more flexibility. However, for special cases where deorbit and plane change burns can be made simultaneously, single stage solid rocket motor propulsion is feasible.

Once the PDS separates from the Shuttle it must provide the orientating maneuvers to perform multiple burns and deploy payloads. As a consequence, it

RV CARRIER BUS STUDY GENERAL REQUIREMENTS

- o BUS CONFIGURATION - UP TO 8 RV'S
  - RV WEIGHTS UP TO 9000 LBS
  - RV LENGTHS - UP TO 185", DIA'S 45"
- o SERVICE (ONLY AT LAUNCH PAD)
  - GROUND, INDIVIDUAL SYSTEM CHECKOUT
  - ON ORBIT (IN SHUTTLE), SYSTEM CHECKOUT RELAY VIA SHUTTLE TO GROUND
- o POWER - ON SHUTTLE
  SHUTTLE SUPPLIES UP TO 1 KW AC
  - FREE FLIGHT BUS BATTERIES
- o COOLING - ON SHUTTLE 1 KW
  - FREE FLIGHT - BUS SUPPLIED COOLING
- o COMMUNICATIONS - RF LINKS (5) TO SHUTTLE/GROUND STATIONS
  BANDWIDTH 1 MHz
- o ATTITUDE CONTROL SYSTEM
  - CHEMICAL PROPELLANT ATTITUDE CONTROL SYSTEM
- o RELEASE SEPARATION SYSTEM
  - RELEASE/SEPARATION FROM BOOSTER
  - RELEASE/SEPARATION FOR UP TO 8 RV'S
- o PROPULSION SYSTEM FOR BUS
- o GUIDANCE/COMPUTER SYSTEM
  - ORIENTATING BUS FOR MULTI BURNS
  - ORIENTATING BUS FOR DEPLOYING PAYLOADS
- o INTERFACES
  - o STRUCTURAL WITH BOOSTER
  - o ELECTRICAL WITH SHUTTLE
  - o RF & HARDWIRE WITH SHUTTLE
  - o HARDWIRE/CHECKOUT WITH GROUND

FIGURE 82

needs its own guidance system.

Prior to deployment, the PDS must be compatible with the structural, electrical, communications, and thermal interfaces of Shuttle. In the event of an aborted mission, the PDS must survive the Shuttle landing environment.

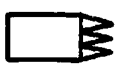
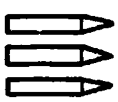
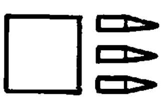
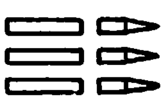
The designs presented in this section were developed to satisfy these requirements. In contrast to a ground launch system which at launch is at zero velocity and altitude, a Shuttle deployed payload has an initial velocity of approximately 25 kft/sec at an altitude of 160 NMI. The Shuttle, in effect, has supplied a little too much energy to the payload and put it into orbit. The PDS must get the payload out of orbit. The PDS was placed into orbit in a closed payload bay and, therefore, does not have to be designed for aerodynamic loads and heating. It must, however, be designed for the vibration levels associated with ground launch. In addition to providing the spacing and orientation burns of a typical ground launch payload bus, it must also provide the burn to establish the deorbit trajectory. This is an additional requirement which ground launch payload buses do not have. In general, the deorbit burns require large ΔV and, therefore, a significant amount of propellant.

As a first step in meeting these requirements, five options described in Figure 83, were explored for the PDS concept. Each option represents a method of interfacing propulsion stages with payloads to meet the DoD requirements on payload number and spacing. The first option looks very much like a ground launch bus except that the propulsion stage is typically of Transtage class but could be as large as a Centaur. It uses a single propulsion system for all the burns. Because the booster is also used for the spacing burn, a considerable amount of inert weight is being carried to do spacing burns. This, in effect, limits the payload capability of this configuration.

The second concept improves upon the payload capability by separating the booster, which provides the deorbit burn, at the completion of that burn. This eliminates a considerable amount of inert weight. The deployed bus provides the propulsion for the spacing burns – a function analogous to ground launch payload buses.

Options 3 through 5 are attempts to reduce total system weight by providing dedicated propulsion systems for each payload. These concepts result in packaging difficulties in the Shuttle payload bay and an increase in the system complexity. Multiple guidance systems are required and the number of reentry objects increases

OPTIONS EXPLORED FOR MULTIPLE-PAYLOAD DEORBIT

OPTION	CONCEPT	NUMBER OF BOOSTERS/SEPARABLE BUSES	PAYLOAD SPACING BURN	BURN TIME SPACING (SEC)	COMMENT
√ 1		1/0 (COMMON BUS)	BOOSTER	60	(+) LEAST COMPLEX (−) PAYLOAD LIMITED
√ 2		1/1 (DEPLOYED BUS)	BUS	60	(+) MAXIMIZES PAYLOAD (−) COMPLEX SYSTEM
3		N/0	BOOSTERS	0	(+) MINIMIZES ΔV REQUIREMENTS (−) SHUTTLE PACKAGING DIFFICULT
4		1/N	BUSES	0	(+) SHUTTLE PACKAGING FEASIBLE (−) COMPLEX SYSTEM
5		N/N	BUSES	0	(+) MAXIMIZES PAYLOAD (−) COMPLEX & SHUTTLE PACKAGING DIFFICULT

√ OPTIONS STUDIED IN DETAIL

FIGURE 83

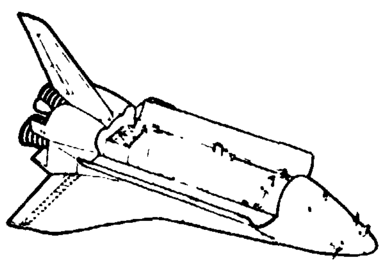
significantly. The advantage of a system such as option 3 is shown in Figure 84 where 3 RV's with Burner II boosters can be carried in the Shuttle payload bay. These payloads could be deployed simultaneously or on consecutive orbits to provide multiple reentry at various impact points.

The disadvantages of options 3 through 5 far outweight the advantages. As a result, these were not studied in further detail. Option 1, however, did have many advantages and received the most detailed study. For instance, reentry conditions at high velocity and shallow flight path angle can be achieved with only small deorbit burns from the Shuttle orbit. A booster such as Transtage can meet the propulsion requirements for such missions. In addition, if the number of payloads is two or three or if the spacing requirement for a large number of payloads is only a few seconds, then the spacing burn requirements are not severe. Again, a Transtage class booster can perform most missions. In fact even a Transtage class booster is overdesigned for some of these missions. Option 2 concepts are only required in the extreme cases of large number of heavy weight RV's with large spacing requirements. This option was only considered to fulfill these special requirements. The next section describes the option 1 and 2 design concepts.

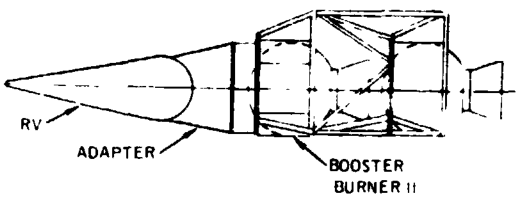
11.1 PDS Designs - Each PDS design for option 1 consists of a booster, spin separation system, and RV's. The designs presented in this section are intended to represent typical configurations. In Section 11.2, specific designs are presented which meet the mission requirements of the six example cases described in Section 3.2. Figure 85 shows a design for a single RV of the 2500 lb class. The booster shown is the Transtage for comparison purposes. Its characteristics are described in detail in Section 8. Other boosters can be substituted for the Transtage. The method of attachment of the tubular support structure to the booster may change, however, for different boosters.

The spin separation system design is based upon that developed for RVTO-2A, Reference 11,with the following exceptions. Since the RVTO-2A system was a ground launch system, the tubular, aluminum support structure had to be redesigned to attach to the Transtage interface ring. In addition, the system was redesigned to accommodate the aborted landing loads experienced in the Shuttle payload bay. Otherwise, the system is a scaled up version of the system used for the large vehicles of the RVTO-2A flights. The spin table consists of a helical, prewound spring which is released by pyrotechnic bolts. The RV is then spun-up to the desired RPM, usually of the order of 60 RPM, and released. The booster performs all the guidance, propulsion, and orientation functions of a ground launch bus.

ıYPICAL SHUTTLE/PAYLOAD CONFIGURATION



TYPICAL RV BOOSTER ARRANGEMENT



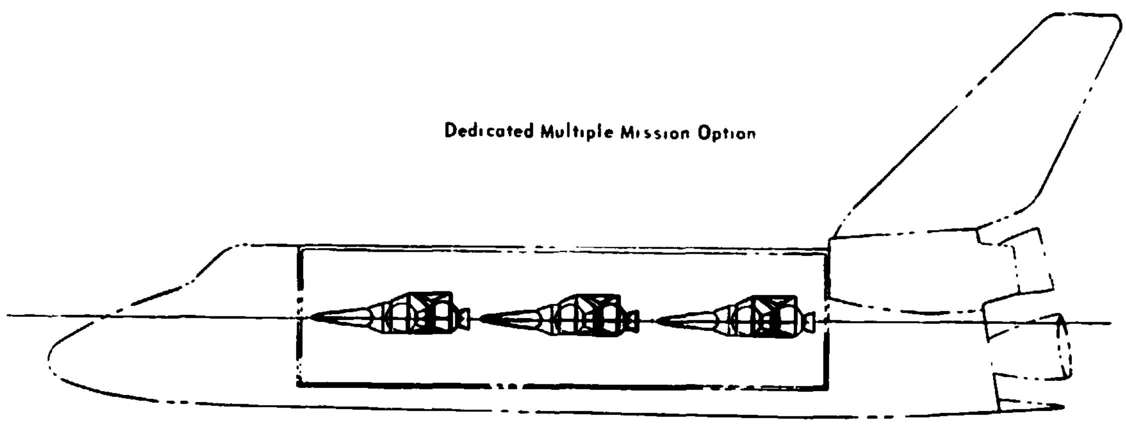


FIGURE 84

VEHICLE CONFIGURATION FOR TESTING A SINGLE 2500 LB RV

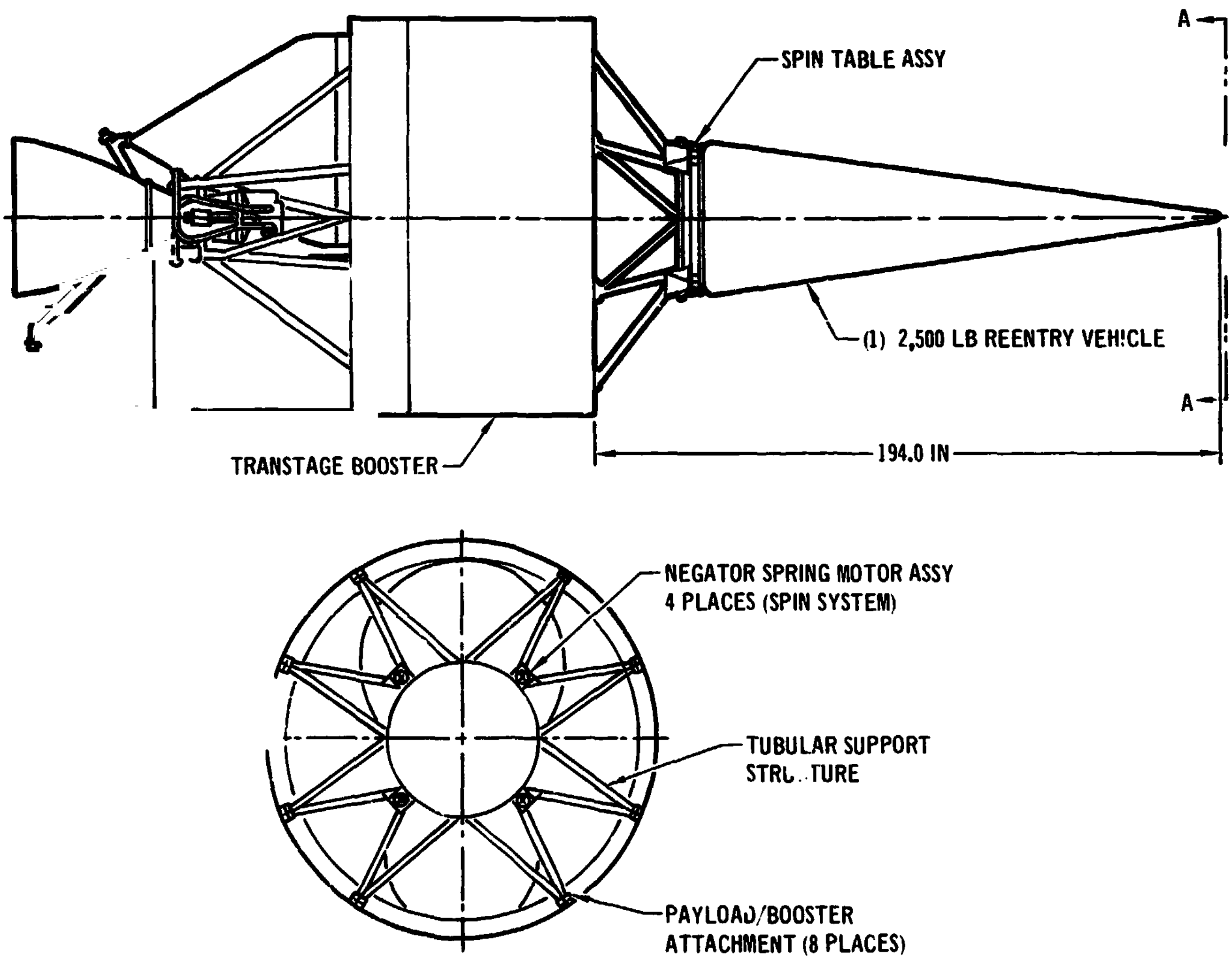


FIGURE 85

After payload deployment, the booster with attached spin separation system perform an orientation maneuver and spacing burn to reenter at the desired pierce point.

The PDS system consists of proven subsystems, i.e., Transtage booster and RVTO-2A type spin separation system. This is expected to minimize development cost and risk. In addition, this basic PDS configuration with changes to the spin separation system to accommodate different payloads, is applicable to a wide range of DoD mission requirements.

For instance, Figure 86 demonstrates the PDS configuration for a small 350 lb RV. In this case a spin tube replaces the spin table. The spin tube allows for more efficient packaging of the smaller RV's and was used on the small vehicles of the RVTO-2A flight. Note that the nose of the RV is almost touching the booster front face. This reduces the total PDS length and frees more of the Shuttle payload bay for shared payloads.

Figure 87 details some design concepts for deploying 4, 6, or 8 RV's of various weights and sizes. Two large vehicles of the 3500 lb class are shown mounted side by side on spin tables. These are nearly the maximum size for geometric fit of two RV's on the 10 ft diameter of a Transtage booster. The attachment to the booster is provided by a structural ring with cross beams to transfer the RV inertial loading to the ring. This structural design can accommodate the many arrangements shown. The 3-1500 lb vehicles of the second sketch are shown in spin tubes. More detailed design may prove that a spin table is more desirable. In either case, the 3 large vehicles can be accommodated. The particular orientation of the RV on the spin separation assembly is not important for payload deployment purposes. The PDS must maneuver to a predefined orientation for any packaging of the payload. The particular orientation is a function of the payload packaging. Therefore, nose first, base first or sideways packaging is permissable and is dictated only by efficient use of the available space.

The eight 200-lb RV case shown in the bottom sketch of Figure 87 may require more booster capability than Transtage can provide. This is especially true if a spacing requirement of 90 seconds at pierce is required between each payload. Figure 88 presents an option 2 design which can better meet these more severe requirements. The booster remains the Transtage but a systems module is added which performs the guidance and propulsion requirements analogous to a ground launch payload bus. The systems module and RV package are in fact analogous to the ground launch bus. Only the initial deorbit burn is provided by the booster. It separates at the completion of that burn and the systems module provides the remaining burns.

VEHICLE CONFIGURATION FOR TESTING A SINGLE 350 LB RV

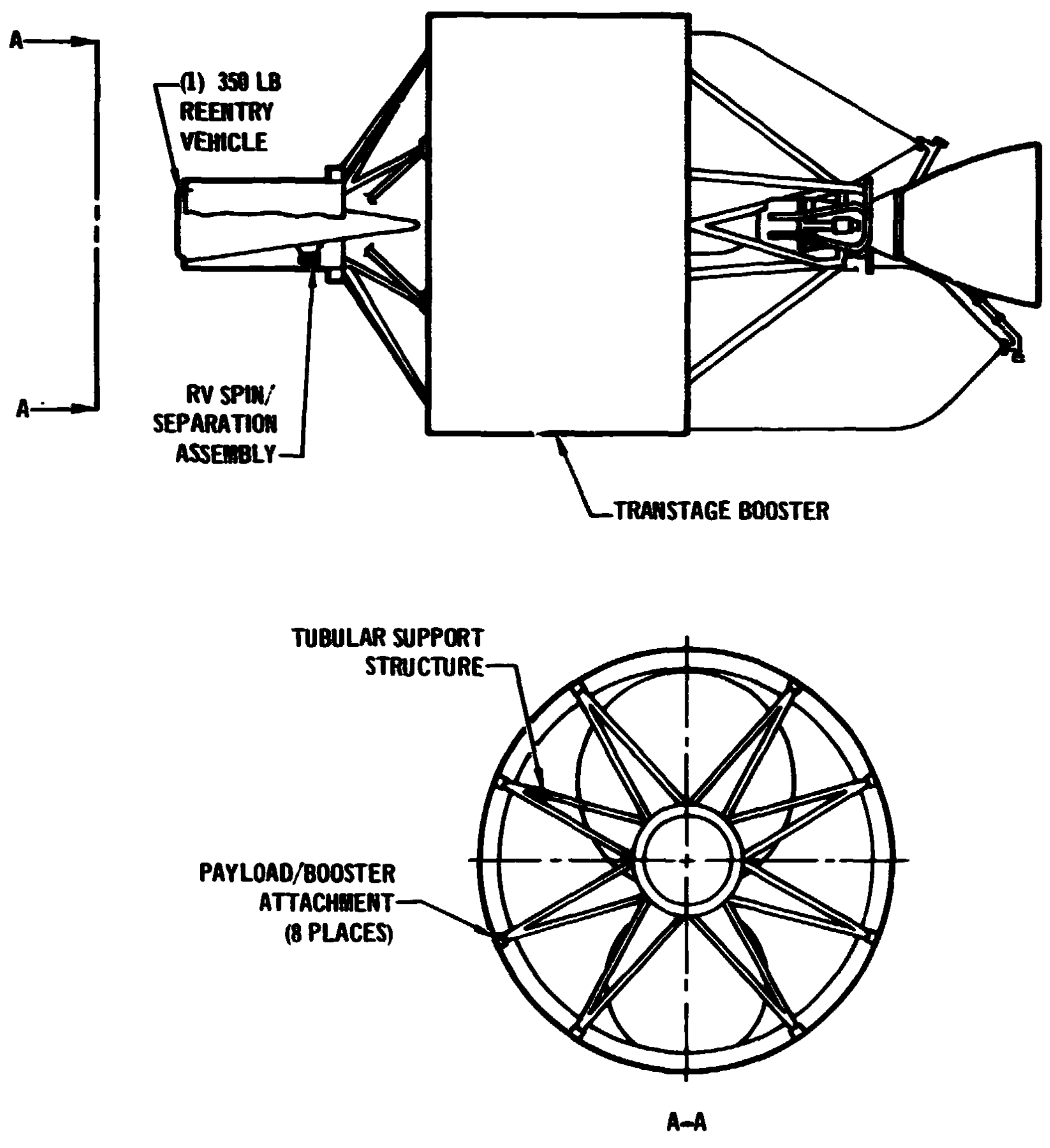


FIGURE 86

MULTIPLE PAYLOAD PDS CONCEPTS

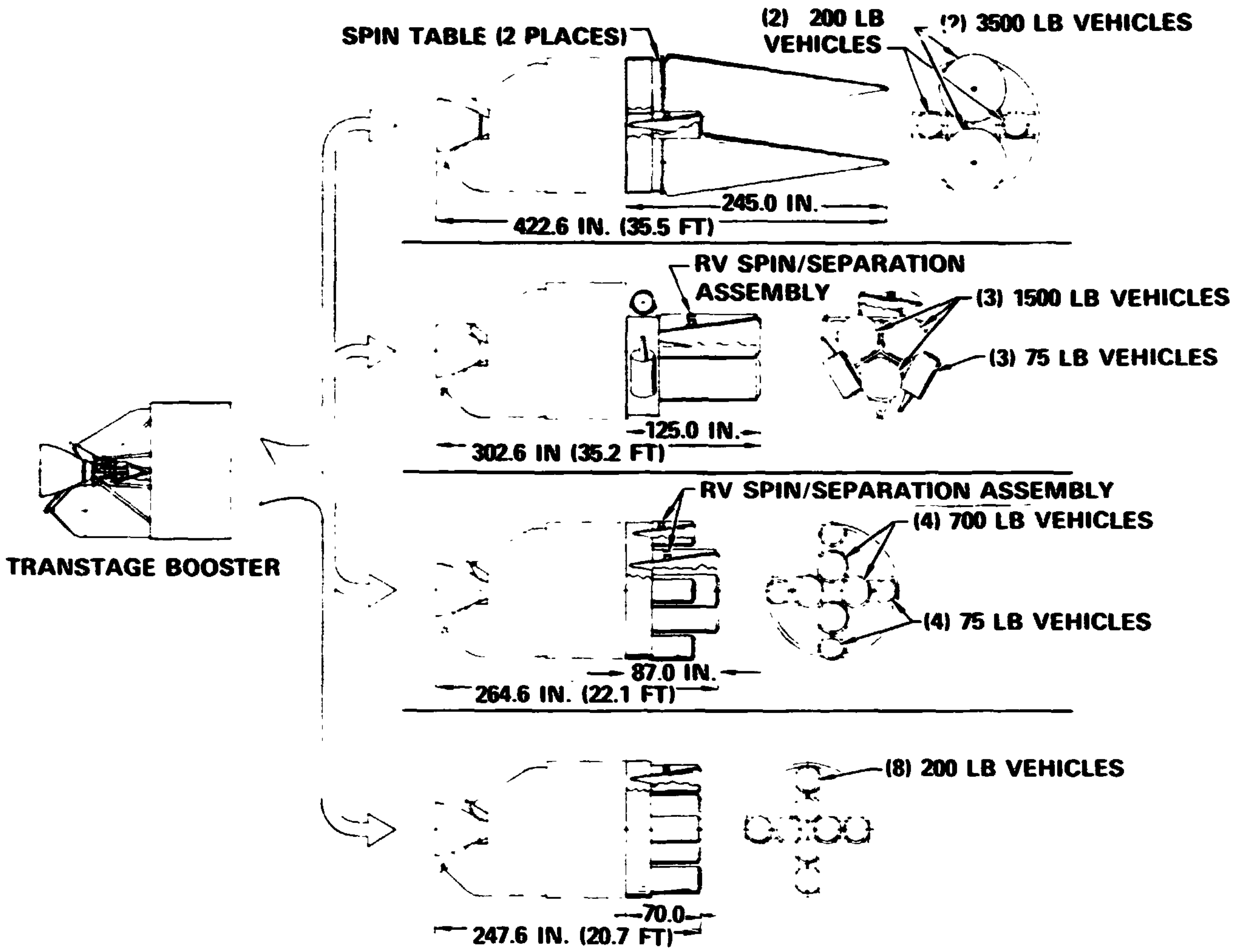


FIGURE 87

DEPLOYABLE BUS CONCEPT FOR (8) 200 LB RV'S

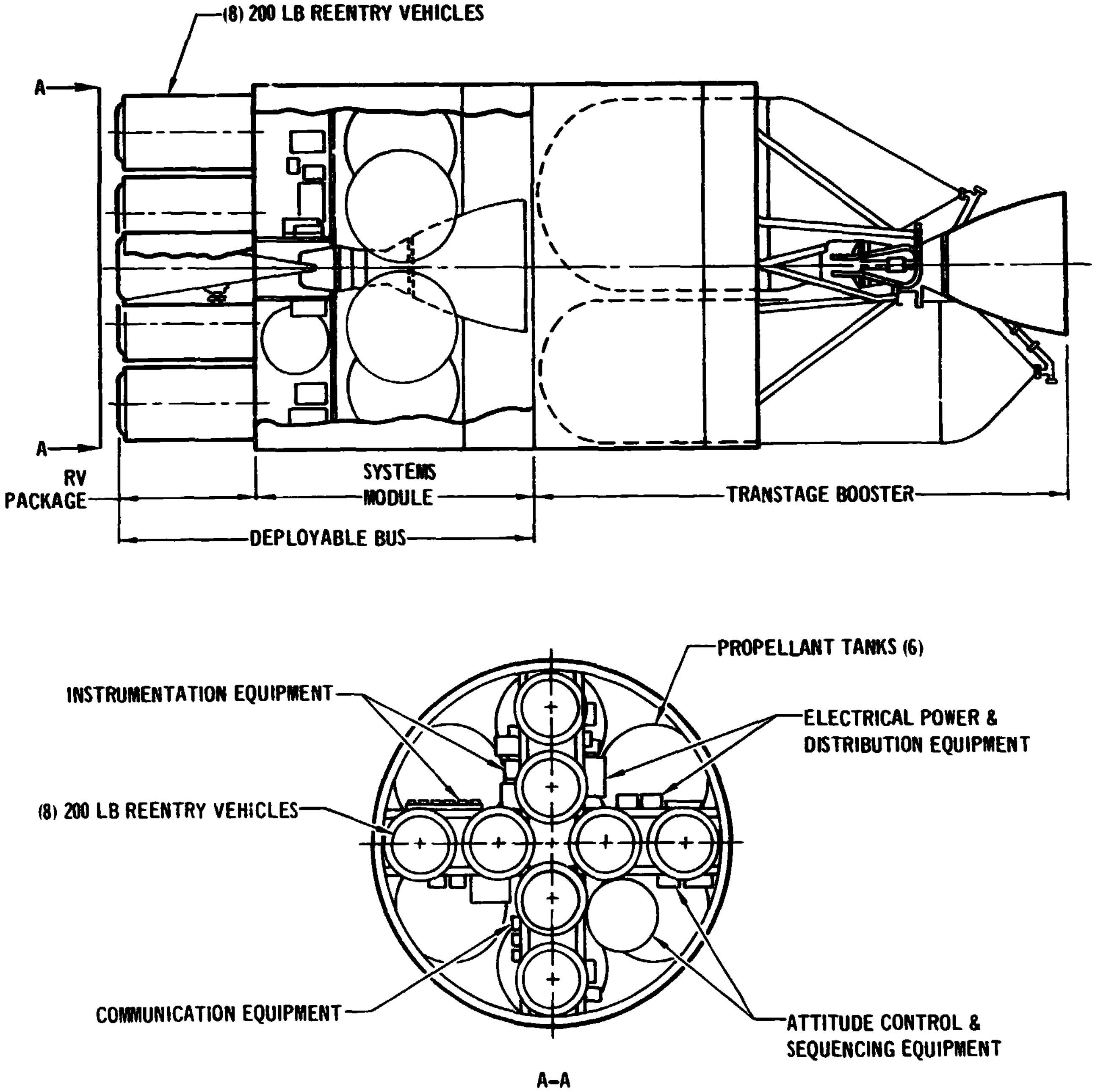


FIGURE 88

As shown in the cross section view it has all the equipment required to be an autonomous system. Because of the large number of maneuver and spacing burns required, it is a liquid propellant system. This type of option 2 PDS would be required for very steep flight path angle missions with multiple payloads spaced far apart at pierce. As is evident from the figure, the system is complex. Both the booster and the system module have their own instrumentation, propulsion, electrical, attitude control and communication equipment.

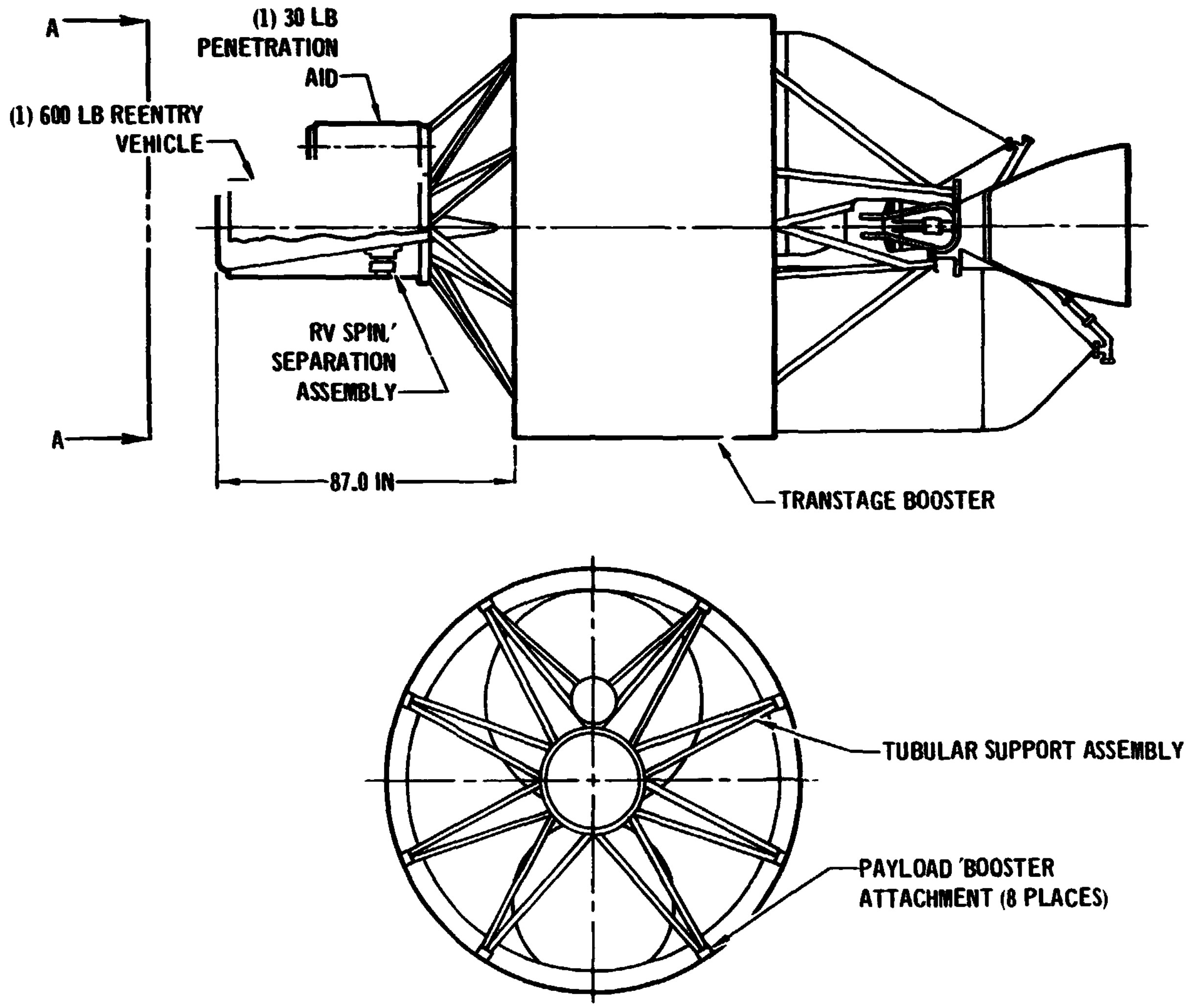
In conclusion, the simplest, least risk PDS for Shuttle consists of a booster, spin separation system and RV's - the option 1 concept. The booster preferably is a liquid propellant system to supply multiple burns. Any additional system modules complicate the PDS and involve additional development cost and risk.

The next sections describes the designs which meet the mission requirements of the example cases identified in Section 3.2.

11.2 Example Cases PDS Designs - The requirements for the example cases are described in Section 3.2 and the mission profiles and performance are described in Section 16. The PDS design and mass properties for example case 1 are shown in Figure 89. The 600-lb RV is centerline mounted in a spin tube. A 30-lb pen-aid is mounted off centerline. The Transtage booster is described in Section 8 and the spin separation system in Section 11.1. The mass properties, i.e., mass, center of gravity, and moment of inertia of the structure and RV's forward of the booster interface are given in the table. Booster masses are given in Section 8. The X coordinate is along the centerline measured from the booster interface, the Y-coordinate is measured perpendicular to the centerline and to the right in section AA; the Z-coordinate is also perpendicular to the center line and up in Section AA. Note that there is a slight c.g. offset due to the off centerline positioning of the pen-aid. The total weights given in the table will differ slightly from those given in Section 16 because these have been refined to reflect design changes and more optimum packaging.

Figure 90 presents the example cases 2 and 4 PDS design for a 1000 lb RV. The RV is mounted on a spin table analogous to that described in Section 11.1. The spin system and RV mounting are symmetric as indicated by the mass properties. This same RV could easily be mounted atop the solid rocket motor IUS as shown in Figure 91. There is little difference in size or interface requirements between the Transtage and SRB. Consequently, the PDS designs provided could be shown with an SRB. However, only in a few special cases can the SRB perform the DoD missions without an additional stage. Example cases 1 and 6 show potential for use of the

VEHICLE CONFIGURATION FOR EXAMPLE CASE I
(ONE 600 LB RV & ONE 30 LB PEN AID)



A–A

MASS PROPERTIES

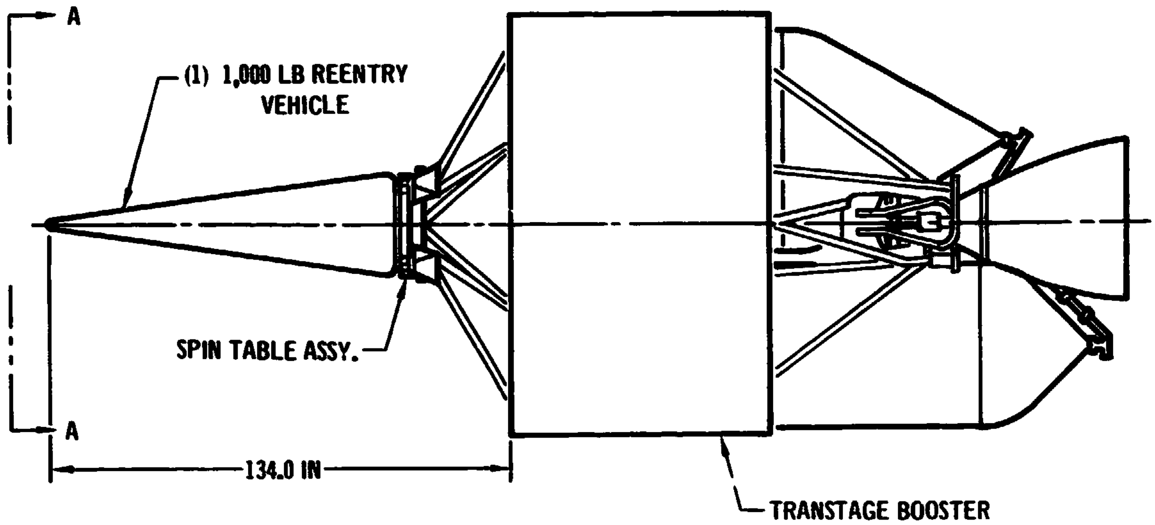
WEIGHT LESS BOOSTER = 860 LB_M
- SINGLE 600 LB RV,
- SINGLE 30 LB PEN AID
- SPIN TABLE & TUBULAR SUPPORT STRUCTURE

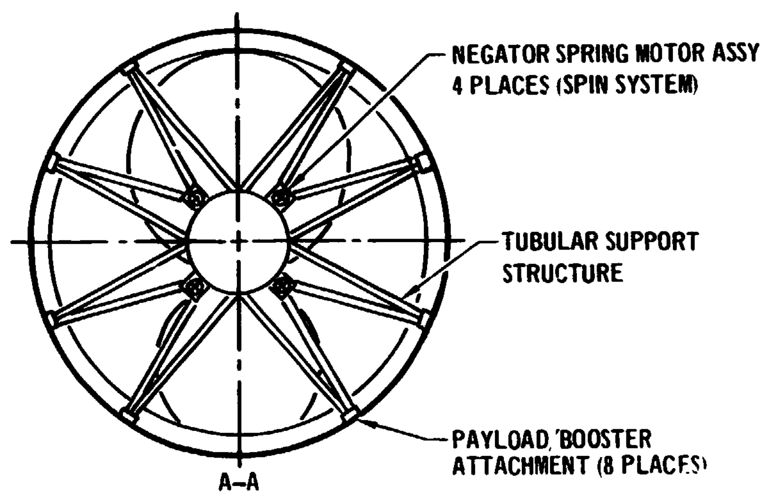
CENTER OF GRAVITY
- X (FWD OF BOOSTER INTERFACE) = 50.0 INCH
- Y (FROM ℄) = 0
- Z (FROM ℄) = 1.6 INCH

MOMENT OF INERTIA
- ABOUT X AXIS = 36.8 SLUG FT^2
- ABOUT Y AXIS = 129.1 SLUG FT^2
- ABOUT Z AXIS = 122.6 SLUG FT^2

FIGURE 89

VEHICLE CONFIGURATION FOR EXAMPLE CASES 2 & 4 (SINGLE 1000 LB RV)





MASS PROPERTIES

WEIGHT LESS BOOSTER = 1270 LB_M
SINGLE 1000 LB RV
SPIN TABLE & TUBULAR
SUPPORT STRUCTURE

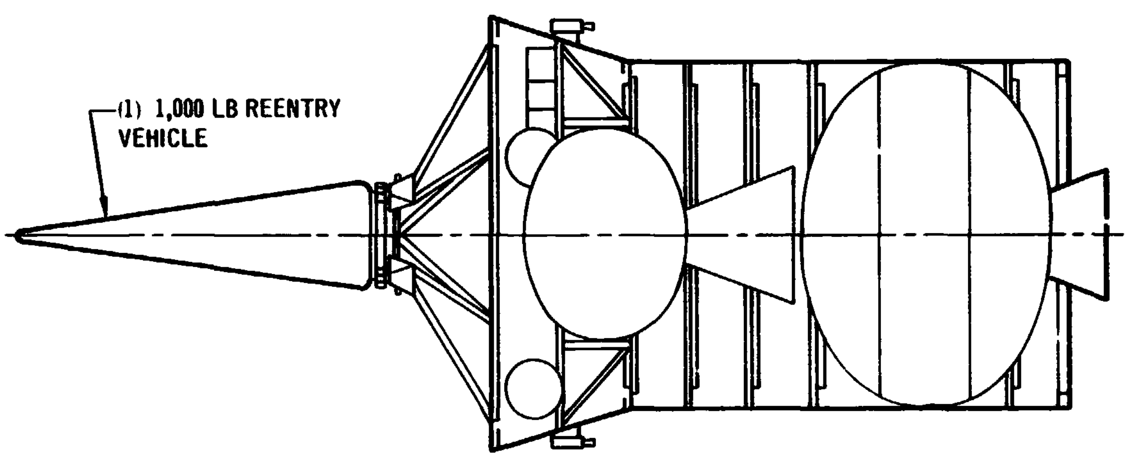
CENTER OF GRAVITY
X (FWD OF BOOSTER = 61.3 INCH
INTERFACE)
Y (FROM ℄) = 0
Z (FROM ℄) = 0

MOMENT OF INERTIA
ABOUT X AXIS = 39.7 SLUG FT^2
ABOUT Y AXIS = 305.1 SLUG FT^2
ABOUT Z AXIS = 305.1 SLUG FT^2

FIGURE 90

COMPATIBILITY OF RV INSTALLATION WITH SRM IUS & TRANSTAGE

SOLID ROCKET MOTOR (SRM) IUS



TRANSTAGE

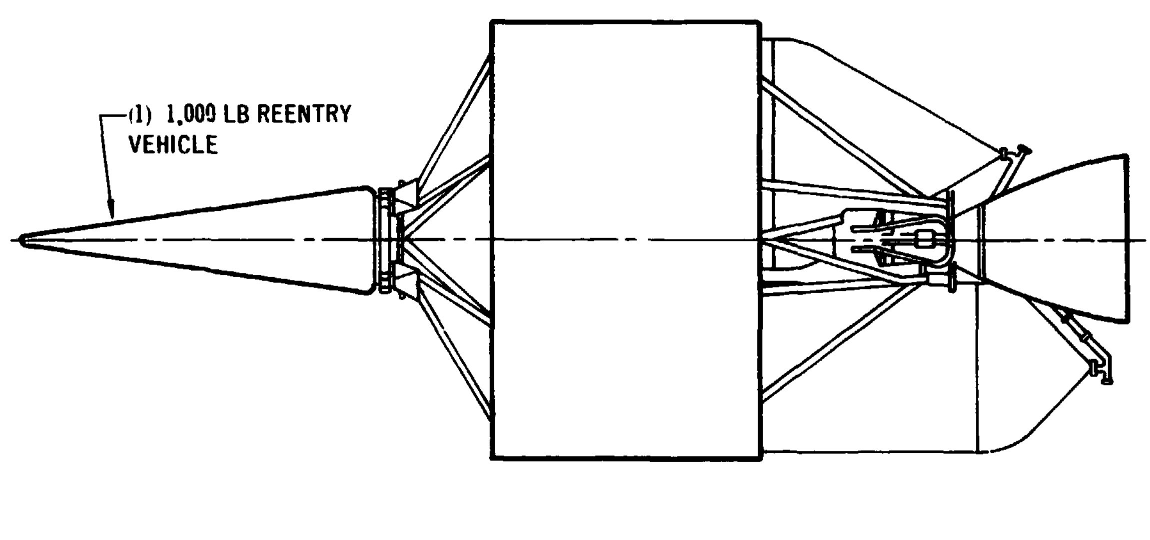


FIGURE 91

SRB as the booster because the plane change and deorbit burn can be combined.

The PDS design for example cases 3 and 5, Figure 92, results in the heaviest and most forward c.g. design of the six example cases. The RV's are mounted side by side on spin tables which results in a c.g. 65.7 inches forward of the booster interface. This c.g. and the 2273 lb_m load is well within the payload interface requirements of Transtage. The mounting is symmetric and no c.g. offset results.

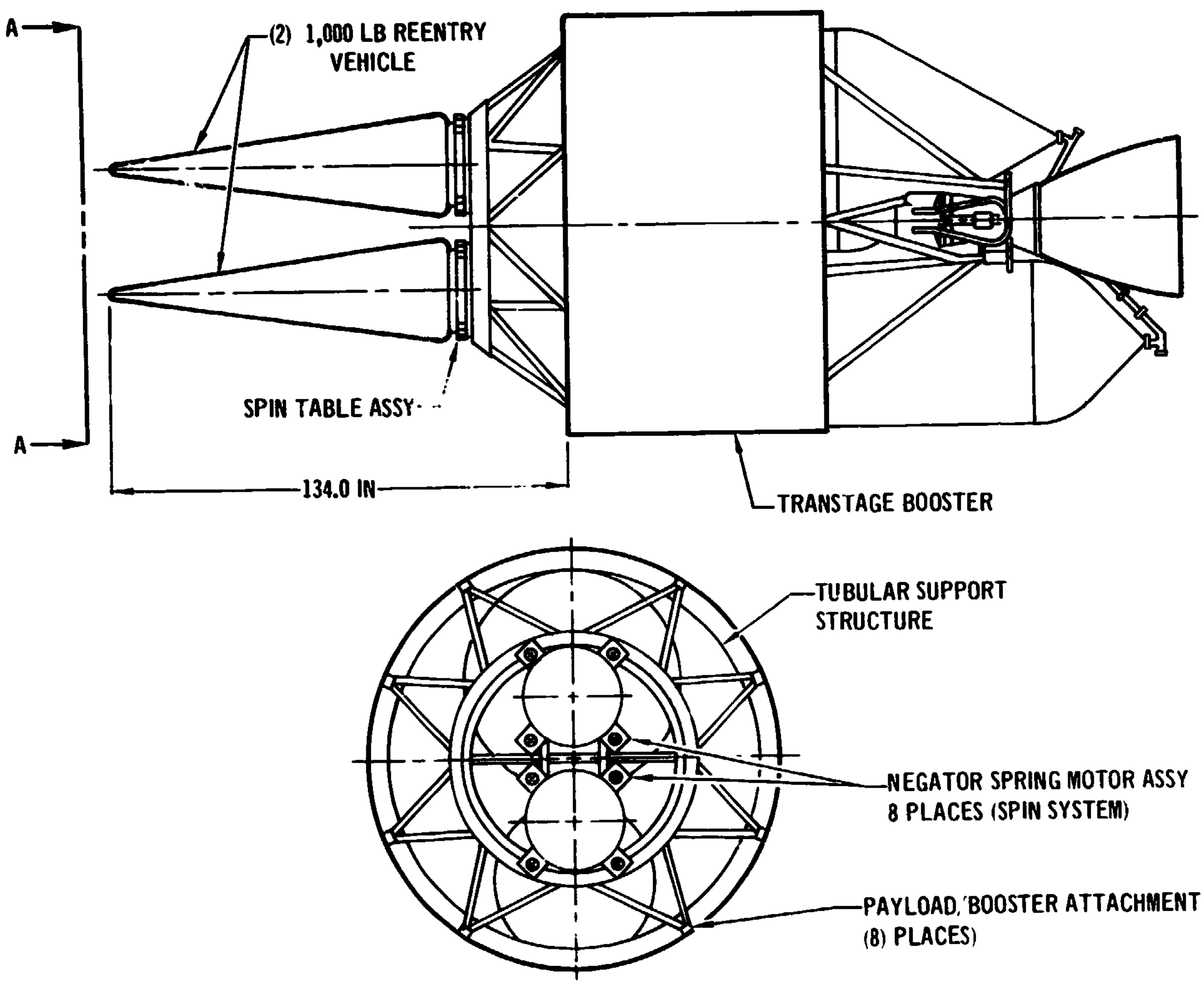
The case 6 PDS design, Figure 93, packages 2-350 lb RV's and 4-30 lb pen aids in spin tubes. The resulting assembly weighs 1110 lb and has a c.g. 45.2 in forward of the interface. This is a less severe payload configuration than the previous case.

In summary, example cases 1 through 6 can be accommodated by the option 1 booster, spin separation system, RV's concept. However, in the particular case of high velocity, shallow flight path angle reentry even Transtage class boosters are oversized. In this case, additional development of Minibus concepts may be warranted.

11.3 Minibus Designs - The Minibus concept was studied as an alternate to using the heavier Transtage for deploying the lighter, low ΔV reentry experiments. The Minibus evolved from a Velocity Package, developed in Reference 12, for use with Shuttle. Figure 94 shows the Velocity Package configurations. The configurations are made up of modules configured from the Shuttle Reaction Control System (RCS) components including the propellant pressurization and engine system. By combining several of these modules, higher total impulse is obtained. Components of the Shuttle Orbital Maneuvering System (OMS) could also be used. The OMS and RCS components were selected because (1) they will be fully qualified to all Shuttle requirements and hence have no development risk, (2) they will be fully compatible with all Shuttle flight, ground and maintenance requirements and equipment, and (3) they will be the newest and hence latest state of the art propulsion components from a performance standpoint. Details of this system design are given in Reference 12.

Figure 95 presents the Minibus concept which uses the two module RCS package of Figure 94. This provides a PDS of 170 inches in length and 110 inches in diameter which fits easily into the Shuttle payload bay and can perform the missions described in Section 9.3. Both RCS derivative 1 or 2, which use 1 or 2 RCS modules, have the payload mounted on a spin table atop the bus. In addition, the Minuteman guidance system is assumed to be the guidance system.

VEHICLE CONFIGURATION FOR EXAMPLE CASES 3 & 5
(TWO 1000 LB RV)

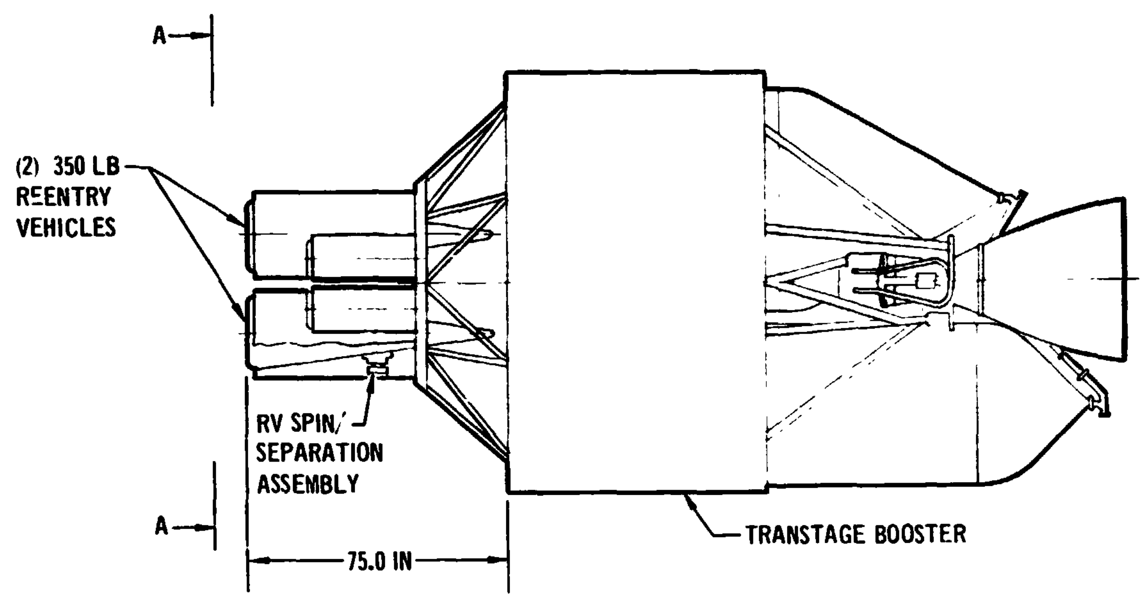


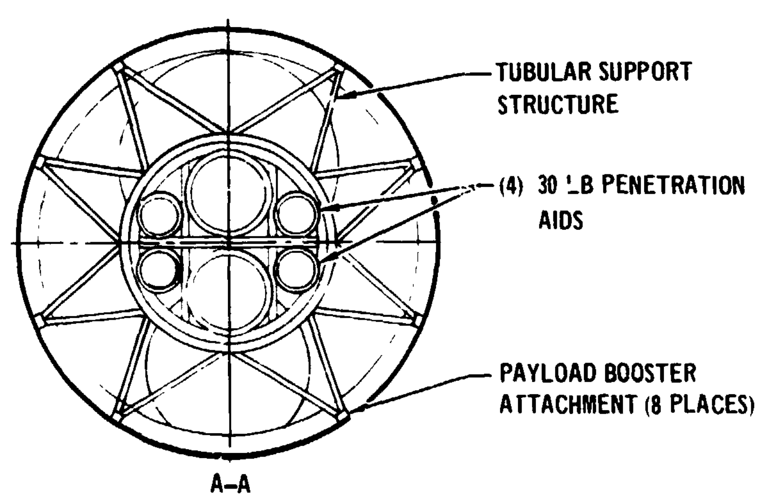
MASS PROPERTIES

WEIGHT LESS BOOSTER = 2273 LB_M	
TWO 1000 LB RV'S	
SPIN TABLE & TUBULAR SUPPORT STRUCTURE	
CENTER OF GRAVITY	
X (FWD OF BOOSTER INTERFACE)	= 65.7 INCH
Y (FROM ℄)	= 0
Z (FROM ℄)	= 0
MOMENT OF INERTIA	
ABOUT X AXIS = 235.1 SLUG FT^2	
ABOUT Y AXIS = 666.2 SLUG FT^2	
ABOUT Z AXIS = 520.8 SLUG FT^2	

FIGURE 92

VEHICLE CONFIGURATION FOR EXAMPLE CASE 6
(TWO 350 LB RV'S & FOUR 30 LB PEN AIDS)





MASS PROPERTIES

WEIGHT LESS BOOSTER = 1110 LB_M
- TWO 350 LB RV
- FOUR 30 LB PENT. AIDS
- SPIN TABLE & TUBULAR SUPPORT STRUCTURE

CENTER OF GRAVITY
- X (FWD OF BOOSTER INTERFACE) = 45.2 INCH
- Y (FROM ℄) = 0
- Z (FROM ℄) = 0

MOMENT OF INERTIA
- ABOUT X AXIS = 91.4 SLUG FT^2
- ABOUT Y AXIS = 145.6 SLUG FT^2
- ABOUT Z AXIS = 118.2 SLUG FT^2

FIGURE 93

VELOCITY PACKAGE DESIGN OPTIONS

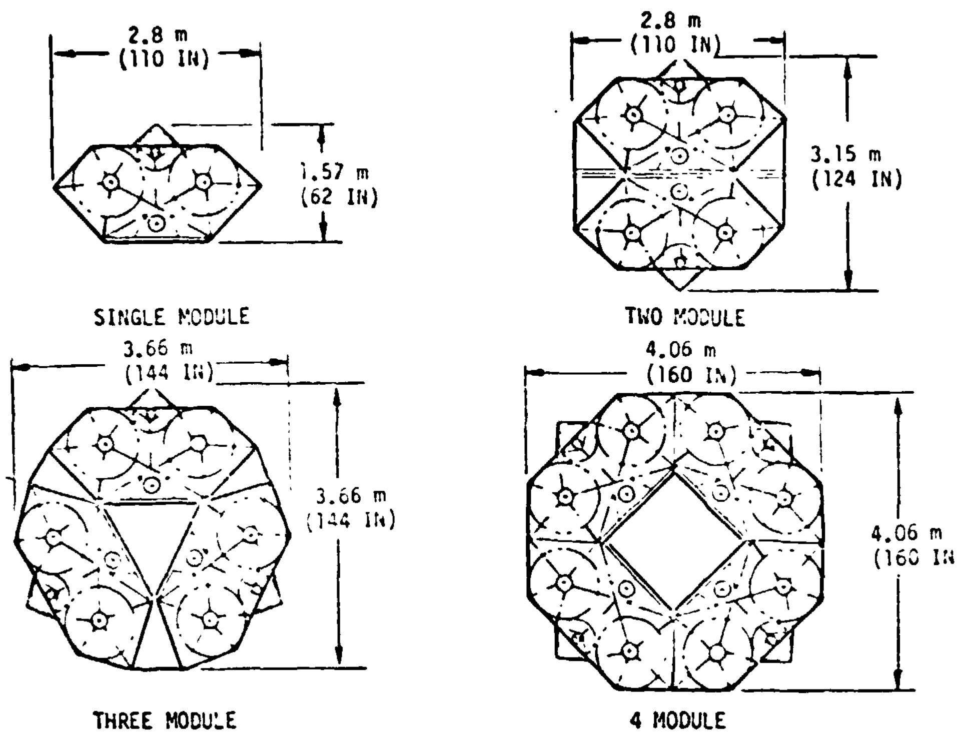


FIGURE 94

MINIBUS CONCEPT
(RCS DERIVATIVE 2)

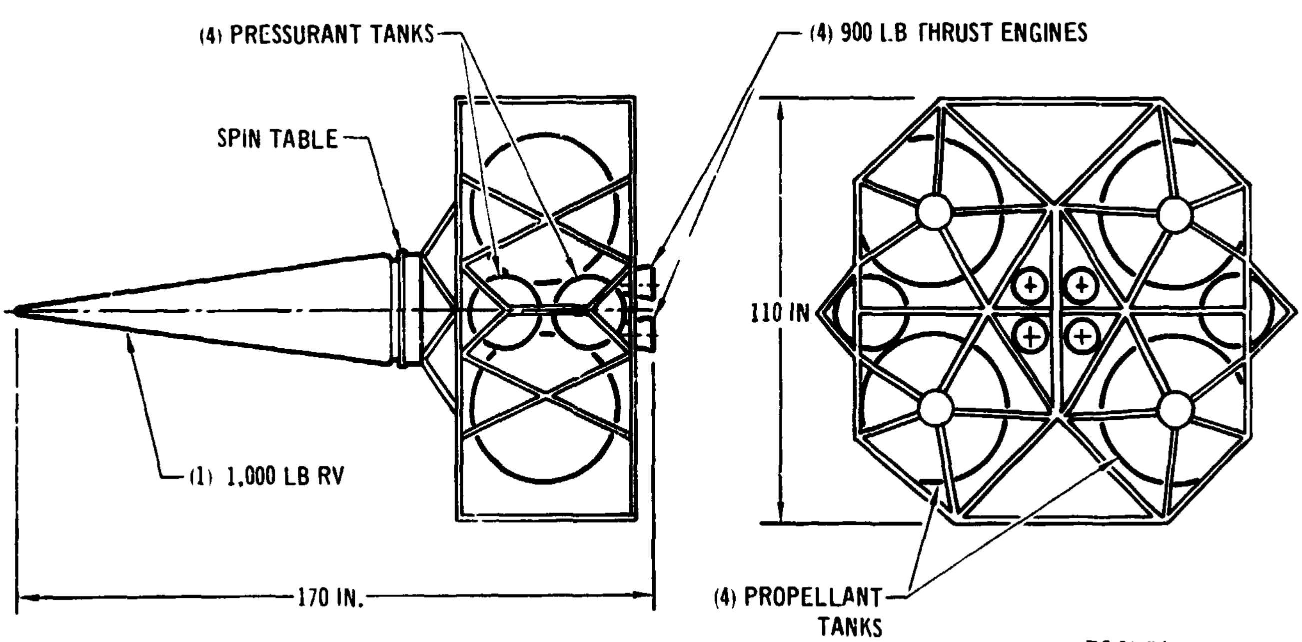


FIGURE 95

By using 3 or 4 RCS modules in RCS derivatives 3 and 4, the payload is semi-submerged in the center of the configuration as shown in Figure 96 for a three module system. This arrangement minimizes the length of the resultant launch configuration. It is only 9.5 ft long for a 1000 lb RV. A fifth concept, a derivative of the Shuttle Orbital Maneuvering System (OMS), is shown in Figure 97. It uses the propellant, pressurization, and engine system of the OMS system. The RV is mounted on a spin table atop the bus. For all concepts a Minuteman type guidance system was assumed.

The physical and propulsion characteristics for the fives concepts and the Transtage are contained in Figure 98. The concepts are shown in order of increasing launch weight and propulsion stage length. The RCS derivatives 3 or 4 seem the most attractive because they minimize length, have a moderately high thrust level, and weigh between 1/3 to 1/2 of the Transtage fully loaded. In addition, they need not be offloaded for the shallow flight path angle high velocity reentries described in Section 9.3. These Minibus concepts are a strong candidate for a PDS in that flight regime.

11.4 Payload Interfaces - The interfaces of the PDS in the payload bay with Shuttle were identified for communications, environmental, electrical power, structural, and thermal control. In general, these interfaces with Shuttle are compatible or can be made compatible with the payload requirements.

The Shuttle/payload communications interfaces are identified in Figure 99. These do not include the checkout console described in Section 12. Prior to deployment the bus PCM data from the booster and RV's must be monitored. These data provide mission critical parameter and safety data for launch commit decisions. The Shuttle payload data interleaver provides the interface unit to receive these data and incorporate it into the Shuttle communication system. If the booster or RV data rate is greater than 64 kbps, the payload must provide only 64 kbps to the interleaver.

Once the PDS is deployed an RF data link between the PDS and Shuttle must be provided through the payload interrogator to monitor safety data. The PDS must provide these data at 16 kbps and may need an additional antenna for data transmission to Shuttle.

A hardwired command link is provided through the Shuttle modulator-demodulator. Commands and PDS guidance system updates prior to launch are provided by this link. Once the payload is deployed an RF command link must be maintained through the pay-

MINI BUS CONCEPT (RCS DERIVATIVE 3)

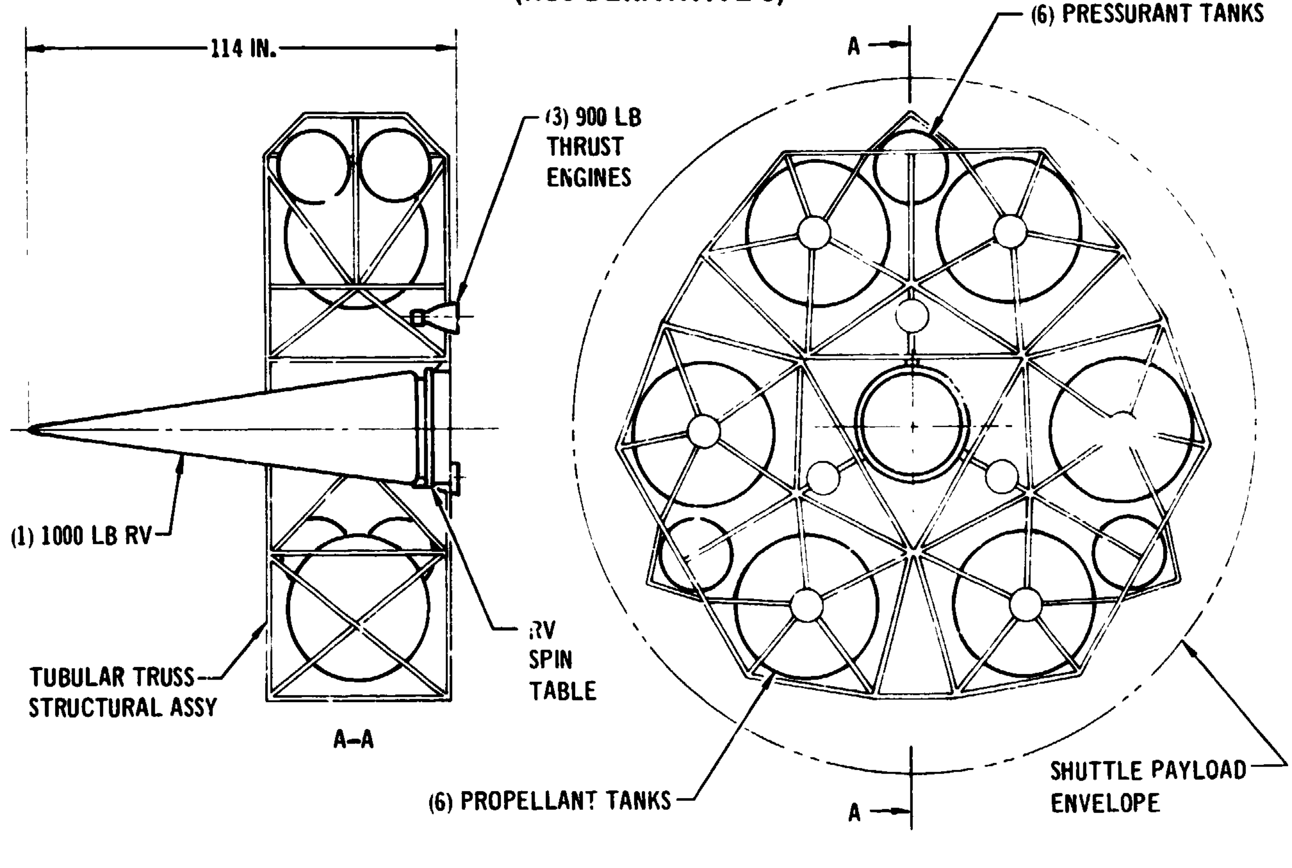


FIGURE 96

MINI BUS CONCEPT
(SHUTTLE OMS DERIVATIVE)

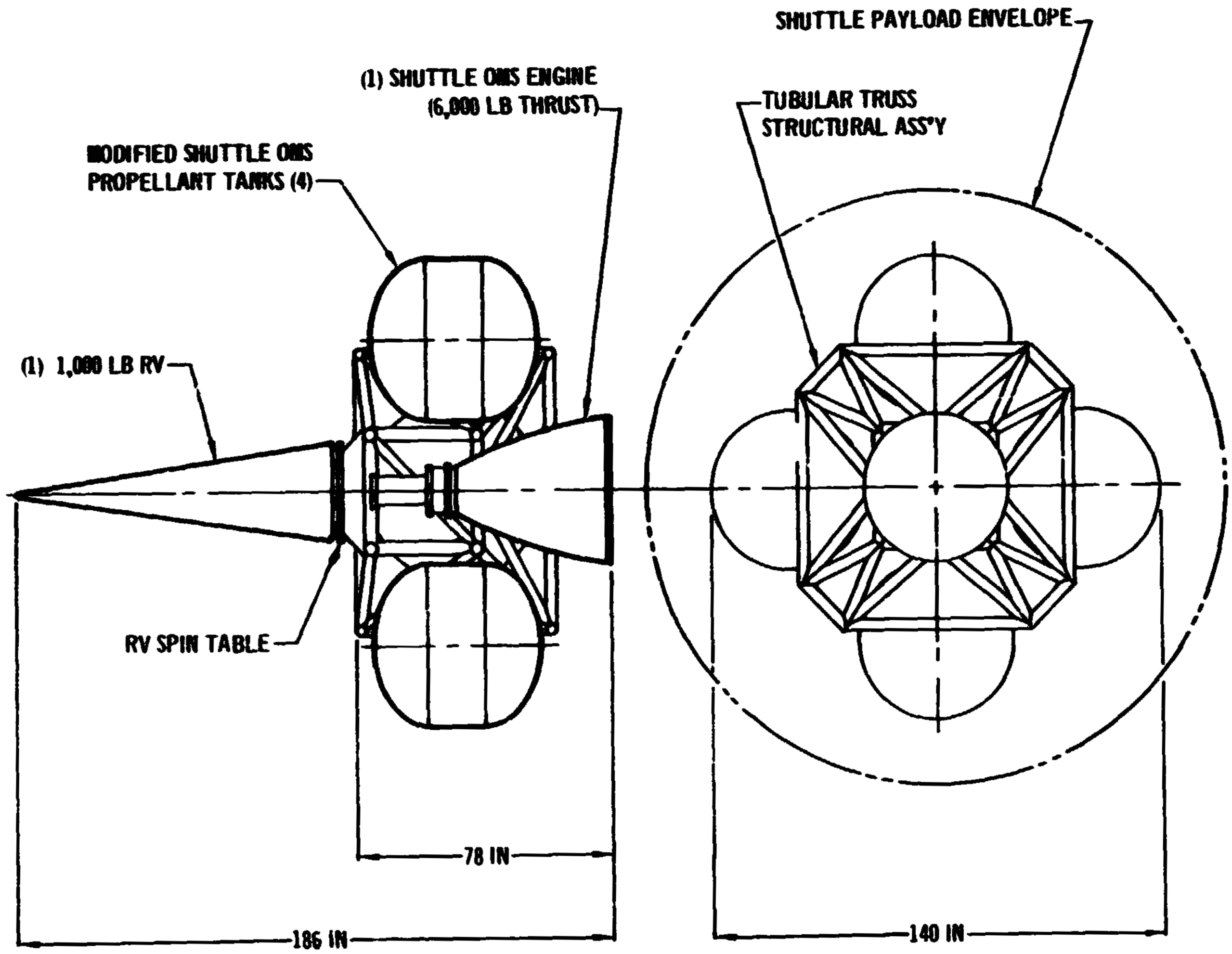


FIGURE 97

COMPARISON OF MINIBUS & TRANSTAGE PROPULSION CHARACTERISTICS

	①	②	③	④	⑤	⑥
TOTAL WEIGHT (LB)	3192	6085	8977	11869	14418	26783
PROPELLANT WEIGHT (LB)	2167	4334	6500	8667	12000	23032
BURNOUT WEIGHT (LB)	1025*	1751*	2477*	3202*	2418*	3751**
SPECIFIC IMPULSE (SEC)	289	289	289	289	313	301.3
THRUST LEVEL (LB_f)	900	1800	2700	3500	6000	15733
LENGTH (FT)	4	4	4	4	6	14

* INCLUDES MINUTEMAN GUIDANCE SYSTEM

** INCLUDES TRANSTAGE SYSTEM

CODE:

1 - 4 RCS DERIVATIVES

5 OMS DERIVATIVES

6 TRANSTAGE

FIGURE 98

SHUTTLE/PAYLOAD COMMUNICATION INTERFACES

WITHOUT OPERATION & CHECKOUT CONSOLE

SIGNAL TYPE	CONCEPT	SHUTTLE INTERFACE UNIT	PAYLOAD REQUIREMENT
BUS PCM DATA	MISSION CRITICAL & SAFETY DATA TO BE MONITORED VIA ENGRG. DATA INTERFACE	PAYLOAD DATA INTERLEAVER	PROVIDE 64 KBPS SUB-FRAME OUTPUT FROM PCM SYSTEM IF TOTAL BUS DATA RATE EXCEEDS 64 KBPS
RV PCM DATA	SAME AS ABOVE USING SEPARATE INTERFACE(S)	PAYLOAD DATA INTERLEAVER	SAME AS ABOVE FOR RV
BUS/RV PCM DATA (DEPLOYED)	SAFETY DATA TO BE MONITORED VIA RF LINK	PAYLOAD INTERROGATOR	PROVIDE 16 KBPS SUB-FRAME FROM BUS PCM SYSTEM THAT INCLUDES RV DATA. MAY REQUIRE SEPARATE TRANSMITTER & ANTENNA.
COMMANDS	BUS & RV DISCRETE COMMANDS & GUIDANCE SYSTEM UPDATES HARDWIRED DIRECT TO BUS.	MODULATOR-DEMODULATOR	SWITCHING RELAYS & STORAGE REGISTERS. BUS TO RV INTERFACE.
COMMANDS (DEPLOYED)	CODED COMMANDS TO BE SENT VIA RF LINK	PAYLOAD INTERROGATOR	COMMAND RECEIVER & DECODER.
CAUTION & WARNING DATA (PRIMARY)	BUS & RV FUNCTIONS HARDWIRED DIRECTLY TO SHUTTLE	C&W ELECT. UNIT	SENSORS, WIRING & SIGNAL CONDITIONING.
CAUTION & WARNING DATA (BACK-UP)	DATA INCLUDED IN PCM SIGNALS DEFINED ABOVE.	PAYLOAD DATA INTERLEAVER	NO ADDITIONAL REQUIREMENT.

FIGURE 99

load interrogator. The PDS must therefore have switching relays and storage registers to accommodate the commands and a receiver and decoder to process the RF link commands.

Caution and warning data provides indications of PDS malfunctions which could result in an aborted mission. These can be handled through the Shuttle caution and warning electrical system in the primary mode and the payload data interleaver in the back-up mode. For the primary mode, payload sensors and electrical systems would be required to monitor and transmit malfunction information. The impact of these communication interface requirements on the PDS design are detailed in Section 12.

The electrical power interface is provided at two locations within the Shuttle payload bay as shown in Figure 100. Two panels are located on the aft bulkhead and provide the average and peak power indicated from the Shuttle bus B & C. Near the front of the payload bay, two more panels exist. One provides power from a payload dedicated fuel cell and the other from the Shuttle main bus. If all four panels were to be used an average power level of 15 kw could be maintained. Correspondingly, a peak power requirement of 24 kw can be met. The DoD power requirement of 1 kw per payload can easily be met by the electrical interfaces.

Figure 101 describes the structural interfaces in the Shuttle payload bay. The retention points are points for attachment of the pallet to the payload bay load support points. There are 12 along the bottom centerline beam and 13 on each longeron for a total of 36 attachment points. The stations are indicated in the figure. The DoD payloads examined are compatible with pallets which would be secured at the retention points.

The coolant interface is shown in Figure 102 and consists of a closed liquid coolant piping system with a payload heat exchanger. A liquid coolant heat exchanger is required because the payload bay has no atmosphere and radiation cooling of the payloads is not adequate. The couplings are located at the front of the payload bay. Sections 13 and 14 discuss the coolant system performance in detail.

Another interface of importance is the environmental interface within the payload bay. During Shuttle launch, acoustic levels greater than 135 db are experienced in the Shuttle payload bay. Therefore, payloads with lower acoustic limits will require a noise attenuation shroud.

This interface definition as well as the design concepts presented in this section provide the basis for the analyses presented in the following section. In general, DoD payloads do not impose severe requirements on Shuttle nor does Shuttle

ELECTRICAL POWER INTERFACE

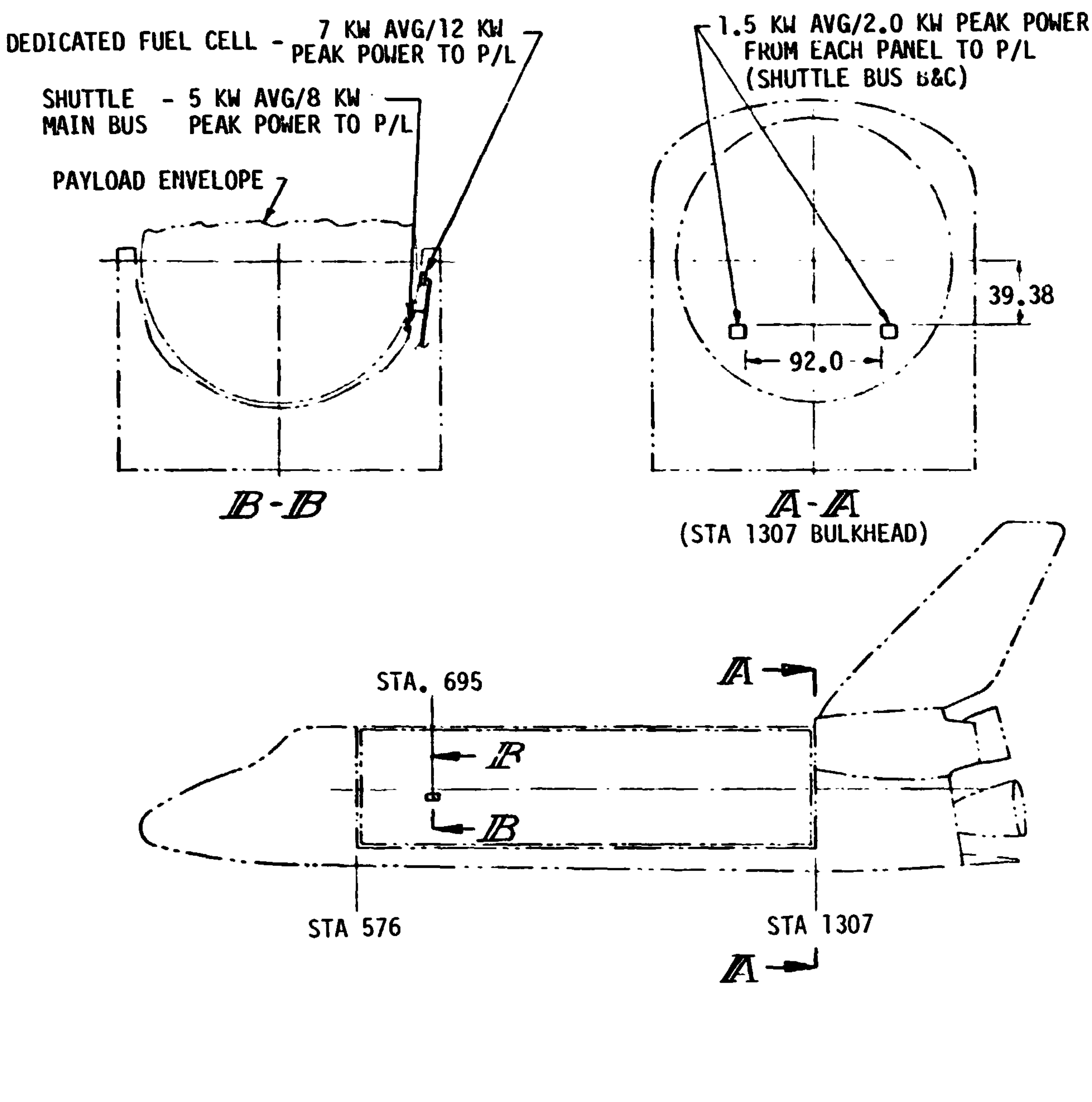


FIGURE 100

STRUCTURAL INTERFACE

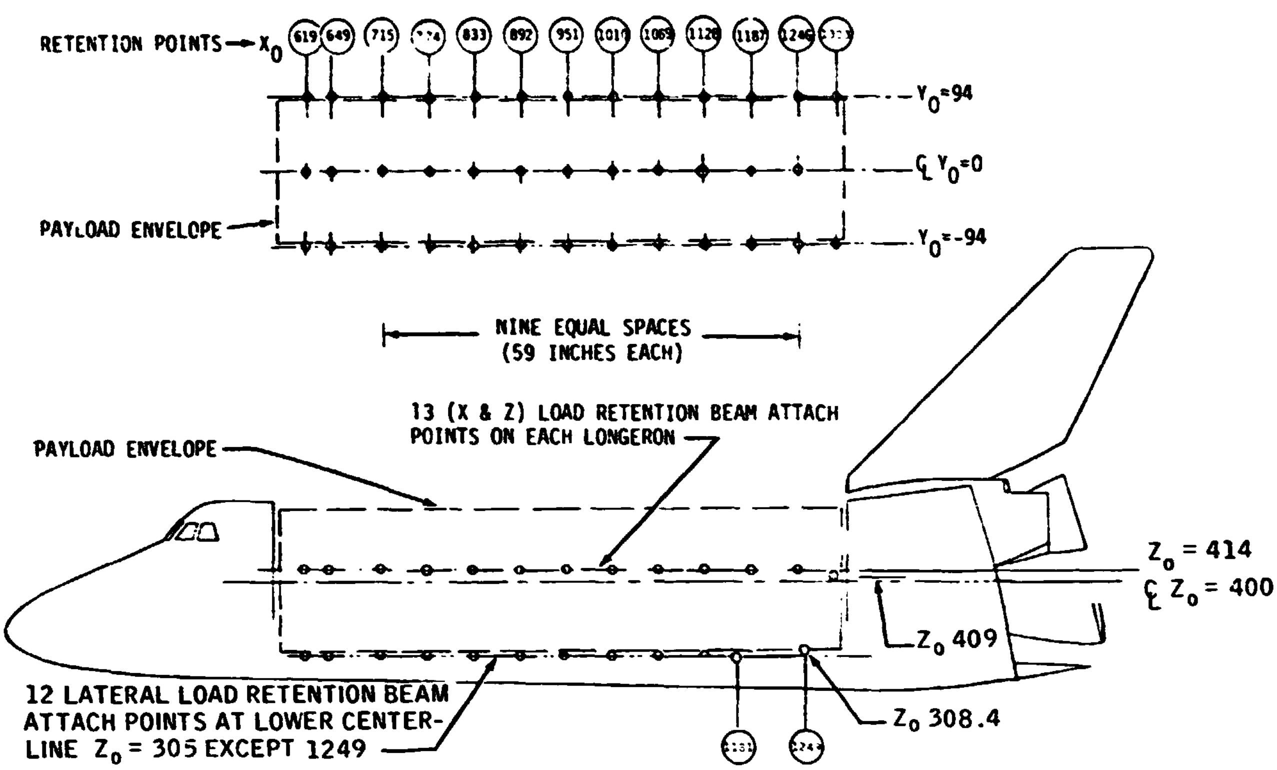


FIGURE 101

PAYLOAD HEAT EXCHANGER INTERFACE

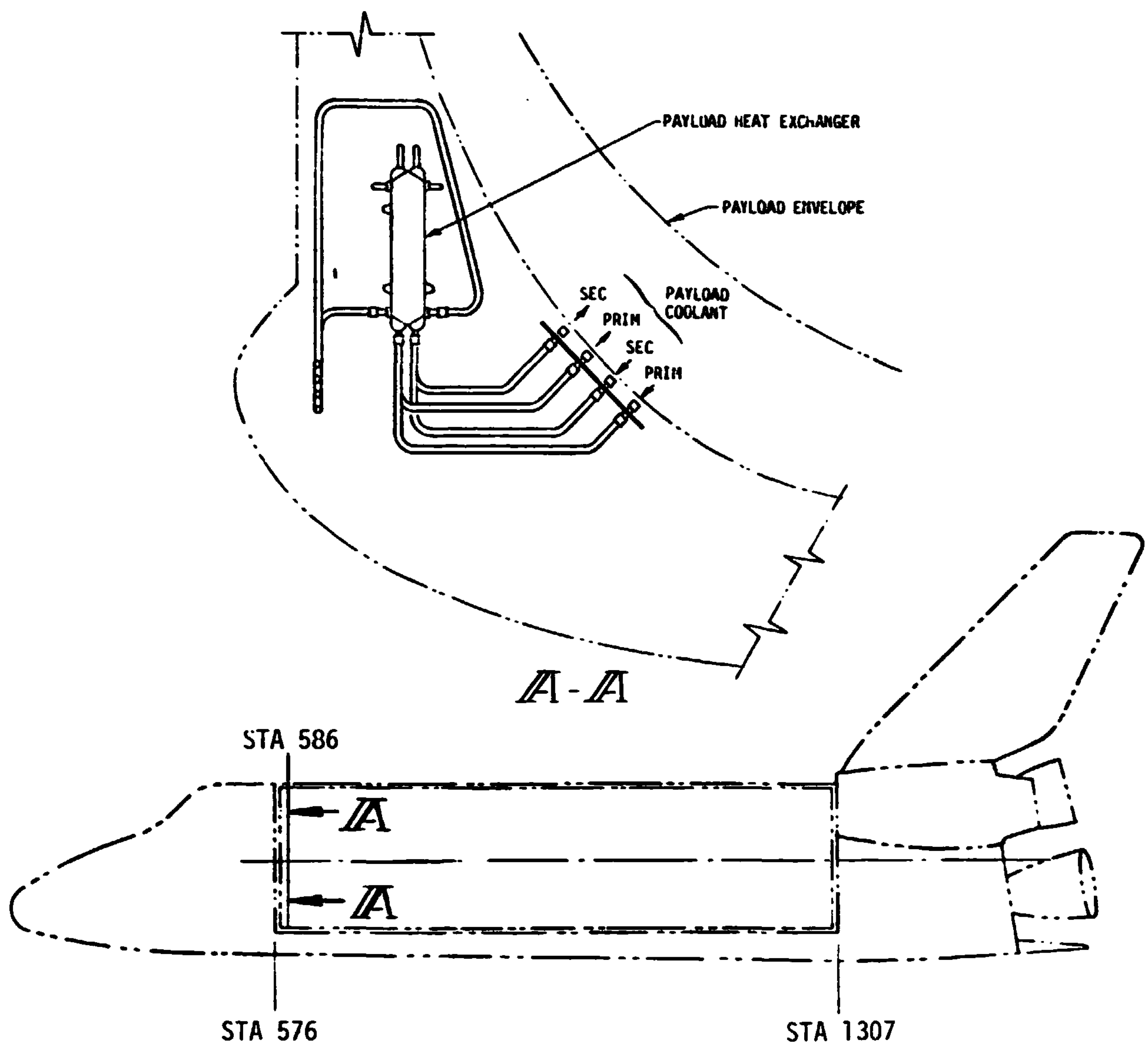


FIGURE 102

greatly change the mode of operation or design for DoD payloads. Perhaps, the most significant impact is on communications which is discussed in the next section.

12.0 COMMUNICATIONS

12.1 Introduction - Communications includes telemetry downlinks, command uplinks and radar tracking for the PDS and one or more attached payloads. The several available communication links between payload, Shuttle, ground and satellites have been analyzed in Section 12.2. The interfaces between Shuttle systems and payloads are defined and their applicability discussed.

All of these interfaces are involved with mission operation discussed in Section 12.3. Particular emphasis has been placed on mission control during the PDS prelaunch checkout and involves the Shuttle to ground link. Shuttle and payload communications with ground stations for the example cases of Section 16 are discussed in Section 15.

Included throughout are recommendations designed specifically to more fully accomplish communication goals for DoD reentry experiments for the 1980's. These recommendations are summarized in Section 19.

(A special list of acronyms and abbreviations used in this section is given in Figure 103 as a convenient reference.)

12.2 Communication Interface Analysis - Much of this analysis involves interpreting available documentation (References 4, 10 and 13 through 17) about the Shuttle and other vehicle systems as they apply to performing DoD experiments. The multiple telemetry system interfaces, including several options, have been investigated and the results are reported herein. These interfaces include:

a. Shuttle to ground
b. Payload to Shuttle
c. Payload to ground
d. Payload to satellite

It is assumed that the PDS PCM downlinks and command uplink operate at S-band frequencies and are compatible with the Space-Ground Link System (SGLS) equipment used by ground stations in the Satellite Control Facility (SCF) network. The command uplink operates at a standard 2K bit rate. The telemetry downlinks can operate over a wide range of data rates. However, the data rate has a major impact on the payload to Shuttle interface. Before analyzing this interface, the Shuttle to ground communication link must be described.

12.2.1 Shuttle to Ground Communications Link - This interface is shown pictorially in Figure 104. An indirect ground link via satellite and deployed payload to Shuttle links is also shown. These links are discussed in subsequent paragraphs.

NOMENCLATURE USED IN COMMUNICATION ANALYSIS

C&W	Caution and Warning
C-Band	3900-6200 MHZ
FM	Frequency Modulated
GDS, MIL, MAD, ROS, ORR, ULA	NASA Tracking Stations
IOS, HTS, GTS, VTS, NHS	USAF Tracking Stations
KBPS	Kilo bits/sec
KMR	Kwajalein Missile Range
Ku-Band	12,000 - 15,000 MHZ
MA	Multiple Access
MBPS	Mega bits/sec
MHZ	Mega Hertz
PAM	Pulse Amplitude Modulation
PCM	Pulse Code Modulation
PDM	Pulse Duration Modulation
PM	Phase Modulated
PRN	Pseudo-Random Noise
SA	Single Access
SCF	Satellite Control Facility
SGLS	Space Ground Link System (Air Force)
STDN	Spaceflight Tracking and Data Network
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
USB	Unified S-band (NASA system)
VAFB	Vandenberg Air Force Base

FIGURE 103

CURRENT ON-ORBIT SHUTTLE COMMUNICATION CAPABILITY

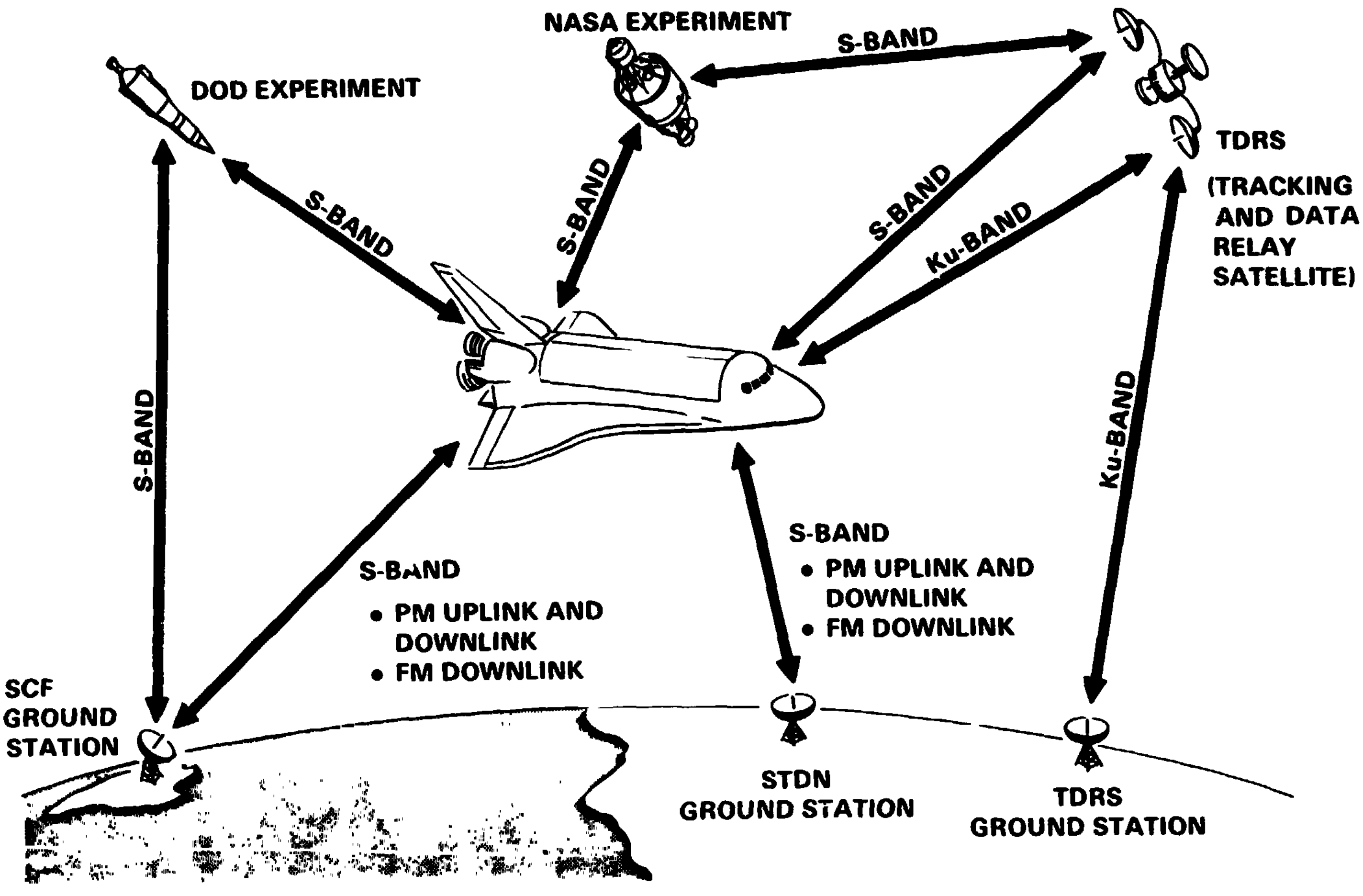


FIGURE 104

Shuttle has two S-band links and one Ku-band link. The S-band Phase Modulated (PM) system is a two-way system providing voice communications, ranging data, uplink commands and downlink telemetry data. This link is limited to a data rate of 128 Kilobits/second (KBPS). The other S-band link is a Frequency Modulated (FM) system used as a one-way system providing only downlink telemetry or video data. Because it provides only a single service, it can operate at a relatively high data rate of 5 Megabits/second (MBPS). The Ku-band system, having much greater bandwidth than the S-band, provides the same services as the S-band PM system plus allowing various combinations of high data rates (up to 50 MBPS) to be transmitted.

The S-band PM system is similar to that used on Apollo and is compatible with NASA's Unified S-band (USB) system installed at the Spaceflight Tracking and Data Network (STDN) stations. This system uses one subcarrier at 1.7 MHz to carry a ranging signal when required. The 128 KBPS telemetry signal is combined with digitized voice to produce a 192 KBPS bi-level signal which modulates the carrier directly. In contrast, the Air Force SGLS equipment of the SCF stations uses the carrier for ranging and puts data on 1 or 2 subcarriers. Therefore, to receive the Shuttle signal, the Air Force can either modify their SCF stations to be compatible or, if ranging is not used, they can receive the PCM and voice signal on standard receivers. In either case, a means for extracting voice from the combined signal and converting it to intelligible speech will be required. A similar problem exists if encrypted payload data is included in the Shuttle data. The telemetry ground station must be configured to separate out the encrypted data, reformat it, and then process it separately through a decrypter.

A simplified diagram of the Shuttle communication system is shown in Figure 105. Emphasis has been placed on those items that relate to the payload interface. As can be seen there are nine data downlink interfaces and one command uplink. The data rates available are 16 and 64 (5 sources) KBPS and 1.024, 5, and 50 MBPS. The choice of data interface with Shuttle payloads depends on the data rate coming from a single or multiple payloads. If this rate exceeds 64 KBPS, one of the three high rate links is required and the data can only be transmitted to ground stations, i.e., the crew does not have direct access to payload information. To provide data to the crew, a high data rate payload must have a dual output so that critical data can be released to the crew at a rate of 64 KBPS or less. Two of the high rate links provide for only real time data to the ground - the 5 MBPS FM link direct to ground stations and the 50 MBPS Ku-band satellite link. The third high rate link (1.024 MBPS) records data on tape and then retransmits it via satellite when convenient.

SHUTTLE COMMUNICATION SYSTEM

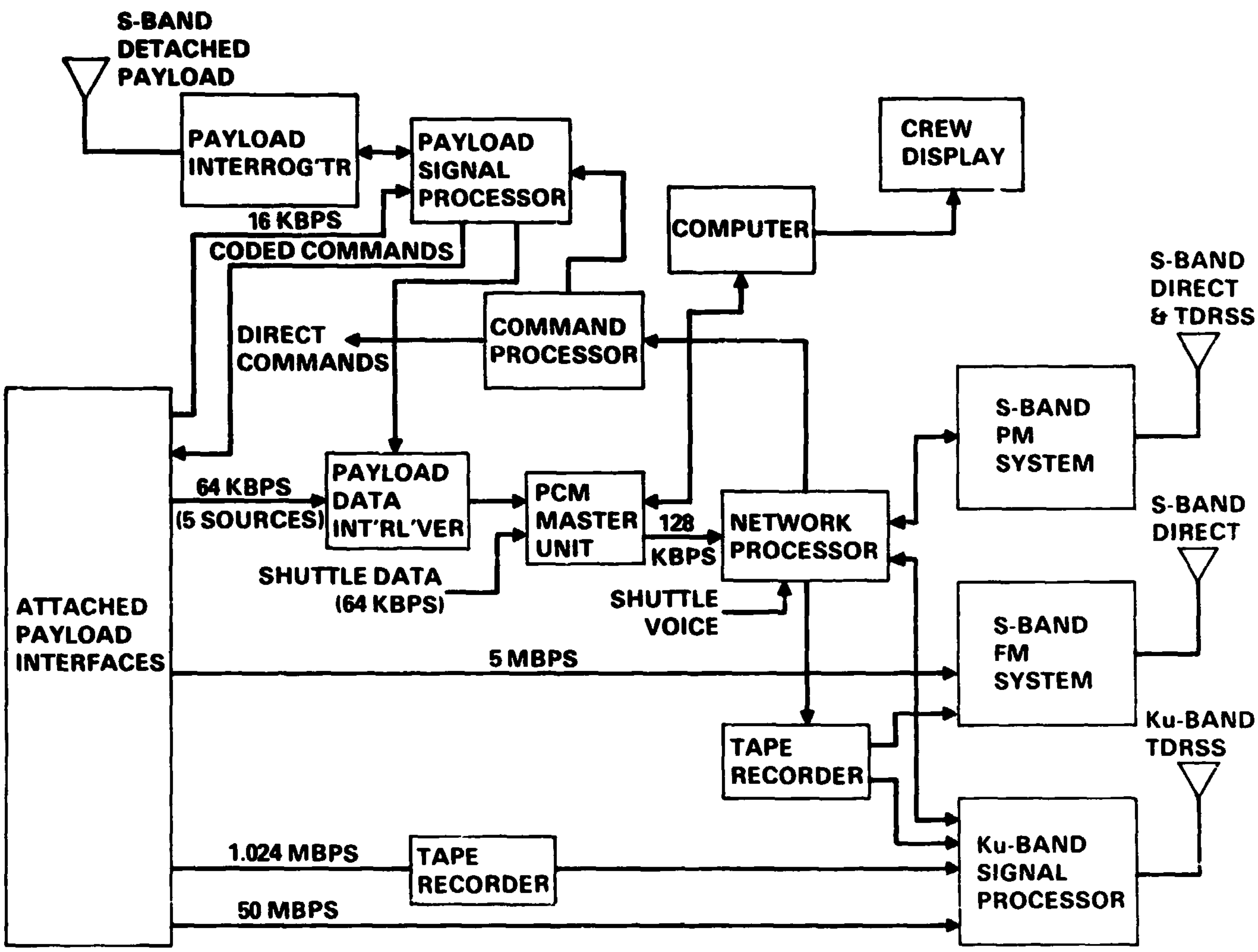


FIGURE 105

Low rate payload data (64 KBPS or less) is available to the Shuttle crew through the Shuttle PCM system. The Payload Data Interleaver will accept 64 KBPS PCM data from 5 separate sources and store these data in a memory. The control program stored in the PCM Master Unit selects the data to be transmitted by the Shuttle via the Network Processor. This same data is also available to the computer where it is processed to premission selected criteria and the results made available to the crew's CRT display. In the event of an anomaly, the signal is also processed by the Shuttle Caution and Warning System. It should be noted that the Shuttle PCM system accepts data only at the 64 KBPS rate; therefore, when more than one source is involved, either the sources must be looked at sequentially or critical data can be selected from each source when the Shuttle PCM control program is prepared.

After the payload is separated from the Shuttle, one two-way RF link at 16 KBPS is available. This interface is through the Payload Antenna and the Payload Interrogator. The Payload Interrogator can also send commands to the separated payload. The 16 KBPS data is stored in the Payload Data Interleaver for transmission and processing by the Shuttle PCM system. This same 16 KBPS data can be sent via hardline to the Payload Signal Processor prior to separation. The Payload Signal Processor also provides for voice communication with a payload.

In summary, telemetry data transmitted from Shuttle directly to ground with the S-band PM system will conform to the Unified S-Band system installed at the NASA ground stations. This PM system is not compatible with the Air Force SGLS equipment at the Air Force ground stations. To receive these data either equipment must be changed or ranging information eliminated from the signal. In addition, the Air Force facilities must install the equipment to extract speech and to decrypt data from the telemetry. Data transmission to the ground for checkout may be required because only 64 KBPS data rate is available for onboard monitoring of payloads.

12.2.2 Payload to Shuttle Communications Link - This link is identified (Reference 4) as consisting of 32 KBPS of digitized voice and 16 KBPS of telemetry data described above. The Shuttle can send 32 KBPS voice and 8 KBPS command data to the payload. (Command data is encoded. The basic, uncoded command rate is 2.4 KBPS.) The payload-dedicated antenna and receiver system on the Shuttle requires a minimum signal strength of -93.8 dbm at the antenna to provide a bit error rate of $10^{-?}$.

In the Shuttle, the payload interrogator serves as a transmitter/receiver connected to the payload-dedicated antenna. This antenna is mounted on the upper fuselage of the Shuttle just forward of the payload compartment hatch. The payload interrogator interfaces with the payload signal processor. In the payload signal processor analog voice is digitized and combined with commands in the transmit mode and digitized voice is separated from data in the receive mode.

The primary need for this link will be to communicate with the PDS booster after deployment. This link was analyzed in a recently completed Delta/IUS study (Reference 10). The results of this analysis are shown in Figure 106 for a NASA (USB compatible) downlink. All the data used in the link analysis are given in the figure for reference. This is true of all the link analyses given in this section. Of course, the important output of the link analysis is the margin given at the bottom of the figure. The study used nominal performance values with favorable and adverse tolerances to show probable, optimistic and pessimistic signal margin. In Figure 106, nominal values have been adjusted to be consistent with the Receive Sensitivity Threshold EIRP of -103.8 dbm as given in Reference 4. The calculated data signal margin of 7.1 db depends heavily on antenna positions of both the Shuttle and the IUS. With unfavorable positions, the Shuttle antenna gain becomes -5 db and the IUS antenna gain is -16 db. This causes an overall negative margin of -13.9 db. Thus, it is obvious that even at the short range of 20 NMI, good communication can not be assured unless the relative positions of the vehicles are constrained.

The IUS downlink considered in the Delta/IUS study (Reference 10) was designed to be compatible with SGLS or USB when transmitting directly to ground stations. In the NASA (USB) link, 16 KBPS data was used to directly modulate the RF carrier and 70 KBPS data was put on a subcarrier. In the DoD (SGLS) link, both signals are on a subcarrier. The NASA configuration is compatible with the payload to Shuttle link but the DoD configuration is not. Improved IUS to Shuttle link performance can be achieved by using direct transmitter modulation, i.e., no subcarriers. A margin calculation for this mode is shown in Figure 107. Note that the margin has been improved from 7.1 db given in Figure 106 to 14.4 db. In addition, direct modulation is compatible with STDN, SGLS and the Shuttle. However, the data rate is limited to 16 KBPS by the Payload Signal Processor. Most payloads require higher data rates. One solution is to provide two outputs from the payload PCM system. One output would provide only critical data at a rate of 16 KBPS or less to Shuttle during initial deployment. The other output would include all data at a higher rate for transmission to ground. This would require a switch in the payload, so that after 20 NMI separation from Shuttle, the higher data rate is applied to the telemetry transmitter for direct ground transmission.

Another solution is an improved Payload Interrogator which performs the function of transmitter/receiver for voice (32 KBPS), telemetry (16 KBPS), and commands

ANALYSES OF A TYPICAL LINK BETWEEN A PDS AND ORBITER

PDS Transmitter Power (20 watts)	13.0 dbw
PDS Transmitter Line Loss	- 4.0
PDS Transmitter Antenna Gain	0.0
Space Loss 20 NMI	-131.0
Polarization Loss	0.5
Orbiter Receive Antenna Gain	0.0
Orbiter Receive Line Loss	-6.5
Received Carrier Power (C)	-129.0 dbw
Receiver Noise Temp (3207°R)	32.6
Boltzman's Constant	-228.6 dbw/°K-Hz
Orbiter Noise Spectral Density (N_o)	-196.0 dbw/Hz
Received C/N_o	67.0
Carrier Loop Performance	
Carrier Mod Loss (β = 1.0, β_1 = 1.5)	-11.2
C/N_o	55.8
Carrier PLL BW (1 KHz)	-30.0
Received SNR	25.8
Required SNR	- 6.0
Margin	19.8 db
Data Performance	
Suppressed Carrier Mod Loss (β = 1.0)	-7.3
Suppressed Carrier C/N_o	59.7
Bit Rate Bandwidth (16 KBPS)	-42.0
Received E_b/N_o	17.7
Required E_b/N_o (BER = 10^{-6})	-10.6
Margin	7.1 db

FIGURE 106

PDS TO SHUTTLE LINK ANALYSIS (DIRECT MODULATION)

Received C/N_0 (from Figure 106)	67.0 db
Bit Rate BW (16 KBPS)	42.0
Received E_b/N_0	25.0
Required E_b/N_0 (BER = 10^{-6})	10.6
Margin	14.4 db
Worst Case Margin (no antenna constraints)	-6.6 db

FIGURE 107

PDS TO SHUTTLE LINK ANALYSIS (JSC 07700)

Received C/N_0 (from Figure 106)	67.0 db
Bit Rate BW (48 KBPS)	46.8
Received E_b/N_0	20.2
Required E_b/N_0 (BER = 10^{-4})	8.4
Margin	11.8 db

FIGURE 108

(8 KBPS). Since DoD reentry vehicles do not require voice transmission capability, a modified Payload Interrogator on the Shuttle should be considered. The Shuttle Payload Interrogator is designed to receive a total of 48 KBPS; if this consisted entirely of data, the link margin would be 11.8 db as shown in Figure 108. A lower bit error rate is used to be consistent with Reference 4. The 11.8 db margin is 1.8 db more than the 10 db recommended in the reference.

Changes to the PDS telemetry system also will increase link margin. If a directional antenna, such as a 1 ft dia parabolic, can be used on the PDS instead of an omni-antenna, the additional 14.4 db gain increases the range to about 112 NMI. A transmitter power of 50 watts boosts the range to 34 NMI and doing both provides a range of 178 nm. Use of a directional antenna on the Shuttle rather than on the PDS also deserves consideration.

Another possibility is a phased array antenna. NASA is developing an Airborne Electronically Steerable Microwave Phased Array (AESPA) (Reference 12) for use on the Space Tug. This system includes an antenna array, transmit and receive capability and pointing and control logic. If it performs as planned, it will transmit 96 watts in a 120° cone, have a transmission gain of 25 db and a receive gain of 30 db. Installing the system on the PDS would provide a range of 790 NMI. Whereas installing the AESPA on Shuttle gives a receiving range of 680 NMI.

As noted, there is a payoff for improving the Payload Interrogator on Shuttle to increase its useful range. This would require a steerable antenna on the Shuttle and higher sensitivity in the receiver. Also, it would be desirable to increase the bit rate capability from 16 KBPS to 64 KBPS, a rate that should be adequate for most boosters and RV's. A 48 KBPS data rate could be realized if the voice capability were removed. Design information on the present Interrogator and the antenna system is not sufficient to identify the specific improvements required. However, if the receiver noise temperature could be reduced to 1607°R(3 db) and receive line losses reduced 3 db, the 6 db improvement would provide a range of 40 NMI at a 48 KBPS data rate at the same 7 db margin shown in Figure 106.

The command link from Shuttle to the PDS is range limited as in the telemetry case. Reference 4 specifies a transmitter EIRP of 1 dbw. Assuming a PDS receive system as defined in the Delta study, a typical command link margin is shown in Figure 109. The command margin of 11.3 db is comparable to the receive margins of Figures 106 through 108. Improvements in the command link are also warranted; e.g., more Shuttle transmitter power, steerable antennas, more sensitive PDS receivers.

SHUTTLE TO PDS COMMAND LINK ANALYSIS

Orbiter Transmit Power (EIRP)	1.0 dbw
Space Loss (2.1 GHz) 20 NMI	-130.2
Polarization Loss	- 0.5
PDS Receive Antenna Gain	0.0
PDS Receive Line Loss	- 4.0
Total Received Power (C)	-133.7 dbw
PDS System Noise Temp	32.5
Boltzmans Constant	-228.6 dbw/°k-Hz
PDS Noise Spectral Density (N_0)	-196.1 dbw/Hz
Received C/N_0	62.4
Data Performance	
Carrier Mod Loss (β = 1.0)	- 1.5
Bit Rate Bandwidth (8 KBPS)	39.0
Received E_b/N_0	21.9
Required E_b/N_0 (BER = 10^{-6})	10.6
Margin	11.3 db

FIGURE 109

In conclusion, improvements to the payload to Shuttle data link are required to increase transmission range and data rate. Range can be improved by using a directional antenna and 50 watt transmitter on the PDS. More significant improvement is realized by using a 96 watt steerable phased array antenna. In addition lowering receiver noise temperature and reducing line losses of the Shuttle payload Interrogator will improve range. Replacing the voice capability with data improves the Payload Interrogator data rate from 16 to 48 KBPS.

12.2.3 Payload to Ground Communications Link - Two payload configurations and three ground receiving systems have been analyzed to demonstrate typical communication link performance. The payloads are a typical booster/RV PDS and a typical RV. The receiving systems are a typical NASA STDN site, a typical USAF SCF site, and the KMR site. A 20 watt transmitter is assumed for the booster and an 8 watt transmitter for the RV. Omnidirectional antennas and a 64 KBPS data rate are assumed for both. This corresponds to a typical IUS configuration. The NASA USB system and the USAF SGLS system are similar in that both have a 1.024 subcarrier oscillator intended for digital data. In the SGLS system, the carrier is modulated with a PRN ranging code that provides range, range rate and angle of arrival tracking data to the ground receiving site. SGLS provides a second data subcarrier. In the USB system, the second subcarrier is intended for voice and the carrier can be modulated with either high rate data or a ranging code.

The link analysis shown in Figure 110 is for a PDS to SCF site and is based on use of a SCF 46 ft antenna of the type located at Oahu, Hawaii. These antennas have a gain of 47.5 db and the receiving system noise temperature is 396°R. A range of 2,700 NMI has been used as a probable maximum line of sight distance. The calculated data margin of 22.9 db is more than adequate and shows that for direct ground communication either higher bit rates or lower transmitter power could easily be tolerated.

A second link analysis from the PDS to a STDN site is shown in Figure 111. This analysis assumes an 85 ft receiving antenna such as the type located at Orroral, Australia. The system noise temperature is 360°R and again a range of 2,700 NMI is used. The data margin of 30.8 db is well above any probable requirement. Again, direct ground communication poses no problem, even for sites with smaller antennas. For example, a KC-135 Advanced Range Instrumentation Aircraft has a 7 ft dia antenna and a system noise temperature of 705°R. This is a combined capability reduction of 23 db which reduces the margin to 7.8 db.

PDS TO GROUND TELEMETRY LINK ANALYSIS (SCF)

PDS Transmitter Power (20 watts)	13.0 dbW
PDS Transmitter Line Loss	-4.0
PDS Transmitter Antenna Gain	0.0
Space Loss (2,700 NMI at 2.25 GHz)	-173.5
Polarization Loss	0.5
SCF Receive Antenna Gain (46 ft)	47.7
SCF Receive Line Loss	0.3
Total Received Power	-117.3 dbW
Receiver System Noise Temp (396°R)	23.4
Boltzman's Constant	-228.6 dbW/°K-Hz
SCF Noise Spectral Density (N_0)	-205.2 dbW/Hz
Received C/N_0	87.9
Carrier Loop Performance	
Carrier Mod Loss (β =0.1, β_1=1.84)	-10.0
C/N_0	77.9
Carrier PLL BW (1000 Hz)	33.0
Received SNR	44.9
Required SNR	6.0
Margin	38.9 db
Data Performance	
Subcarrier Mod Loss (β_1 = 1.84)	1.7
Subcarrier C/N_0	86.2
Bit Rate Bandwidth (64 KBPS)	48.1
Received E_b/N_0	38.1
Theoretical E_b/N_0 (BER = 10^{-6})	10.6
Hardware Degradation	4.6
Required E_b/N_0	15.2
Margin	22.9 db

FIGURE 110

PDS TO GROUND TELEMETRY LINK ANALYSIS (STDN)

PDS Transmitter Power (20 watts)	13.0 dbW
PDS Transmitter Line Loss	-4.0
PDS Transmitter Antenna Gain	0.0
Space Loss (2,700 NMI at 2.25 GHz)	-173.5
Polarization Loss	0.5
STDN Receive Antenna Gain (85 ft)	53.0
STDN Receive Line Loss	
Total Received Power	-111.9 dbW
Receiver System Noise Temp (360°R)	23.0
Boltzman's Constant	-228.6 dbW/°K-Hz
STDN Noise Spectral Density (N_0)	-205.6 dbW/Hz
Received C/N_0	93.7
Carrier Loop Performance	
Carrier Mod Loss (δ_1 = 1.84)	-10.0
C/N_0	83.7
Carrier PLL BW (1000 Hz)	33.0
Received SNR	50.7
Required SNR	6.0
Margin	44.7 db
Data Performance	
Subcarrier Mod Loss (δ_1 = 1.84)	-1.7
Subcarrier C/N_0	92.0
Bit Rate Bandwidth (64 KBPS)	48.1
Received E_b/N_0	43.9
Theoretical E_b/N_0 (BER = 10^{-6})	10.6
Hardware Degradations	2.5
Required E_b/N_0	13.1
Margin	30.8 db

FIGURE 111

The third link is analyzed in Figure 112 and is for an RV transmitting to Kwajalein (KMR). In this link, no subcarriers are used and direct PSK modulation of the carrier is assumed. The calculated margin of 15.5 db is sufficient and the link could tolerate lower transmitter power (5 watts reduces margin ? db) or a higher bit rate (128 KBPS reduces margin 3 db).

In summary, communication downlinks from either the booster or RV to ground stations present no problems and can be implemented with standard hardware. Should a tape recorder be required, data playback can be on the 1.7 MHz subcarrier in the SGLS application or through modulation of the carrier in USB applications. Video could also be applied to the USB carrier. These additional capabilities would not significantly affect the communications link.

12.2.4 Payload to Satellite Communications Link - The NASA Tracking and Data Relay Satellite System (TDRSS) concept consists of two geosynchronous relay satellites, 130° apart in longitude. TDRS east will be at 319° and TDRS west at 189°. They will provide both uplink (voice, tracking, command) and downlink (voice, tracking, telemetry) communications between earth-orbiting spacecraft and the TDRSS ground terminal. Two types of service are provided: Multiple Access (MA) and Single Access (SA). The MA link is intended for downlink data rates up to 50 KBPS, and is expected to provide continuous line of sight coverage for 85% or more of a low earth orbit. The SA link is for data rates up to 6 MBPS on S-band and 300 MBPS on K-band. This link is shared with other users on a priority basis. Since the DoD reentry flights are relatively short duration, priority should be available for use of an SA link.

From data given in the TDRSS Users Guide (Reference 14), Table C-4, we can estimate telemetry requirements for the DoD missions. This table assumes a satellite in a 1,080 NMI circular orbit with an average distance to the TDRS of 23,270 NMI for a space loss of -192 db. Convolution coding of data is required for MA downlinks and recommended for SA downlinks. Convolution coding gives 5 db improvement in reduced bit errors but requires 2 bits per data bit, i.e., a 2 times higher channel symbol rate. For the same 20 watt transmitter system as planned for the Delta IUS and as used in the previous link margin calculations for the PDS, calculated link margins for a S-Band SA satellite link are given in Figure 113. As can be seen, this system produces negative margins. The calculation assumed a single data signal on the link. If subcarriers are used because of a ranging signal or additional data such as tape recorder playback, there would be additional

RV TO GROUND TELEMETRY LINK ANALYSIS (KMR)

RV Transmitter Power (8 watts)	9.0 dbW
RV Transmitter Line Loss	-2.0
RV Transmitter Antenna Gain	0.0
Space Loss (2,700 NMI at 2.25 GHz)	-173.5
Polarization Loss	-0.5
KMR Receive Antenna Gain (28 ft)	43.0
KMR Receive Line Loss	
Total Received Power (C)	-123.5
Receiver System Noise Temp (804°R)	26.5
Boltzman's Constant	-228.6 dbW/°K-Hz
KMR Noise Spectral Density (N_0)	-202.1 dbW/Hz
Received C/N_0	78.6 db
Data Performance	
Bit Rate Bandwith (64 KBPS)	48.1
Received E_b/N_0	30.5
Theoretical E_b/N_0 (BER = 10^{-6})	10.6
Hardware Degradations	4.4
Required E_b/N_0	15.0
Margin	15.5 db

FIGURE 112

PDS TO TDRS TELEMETRY LINK ANALYSIS

PDS Transmitter Power (20 watts)	13.0 dbW
PDS Transmitter Line Loss	-4.0
PDS Transmitter Antenna Gain	0.0
Space Loss (23,270 NMI)	-192.2
Pointing Loss	-0.5
Polarization Loss	-0.5
TDRS Receive Antenna Gain	36.0
Total Received Power	-148.2 dbW
TDRS System Noise Temp (1054°R)	27.7
Boltzman's Constant	-228.6 dbW/°K-Hz
TDRS Noise Spectral Density	-200.9 dbW/Hz
Received C/No	52.7
Data Performance	
Transponder Loss	-2.0
Demodulation Loss	-1.5
Bit Rate Bandwidth (64 KBPS)	48.1
Received Eb/No	1.1
Theoretical Eb/No (BER=10^{-6})	10.6
Required System Margin	3.0
Required Eb/No	13.6
Margin (uncoded data)	-12.5 db
Encoding Gain	5.2
Margin (coded data)	-7.3 db

FIGURE 113

losses. In order to achieve a positive signal margin, a directional antenna can be added to the PDS or the transmitter power increased. By raising transmitter power to 50 watts and improving the transmitter to antenna coupling at least 3 db (losses = 1 db), a margin of 0 db is possible. This approach is not promising and a better approach would be a 1 ft dia steerable parabolic antenna with the 50 watt transmitter. By eliminating the antenna switching capability required for the four flush-mounted omni antennas, losses should not exceed 2 db and the margin now becomes +9 db.

Another possibility is to lower the bit rate to 16 KBPS (as was considered for the payload to Shuttle link). Combining the lower bit rate with an increase in allowable error rate to 10^{-5}, and using a 50 watt transmitter, a workable margin of 3.7 db is possible.

Whereas a payload to ground communication link can be implemented with omni antennas and relatively low transmitter power, the use of a relay satellite will require more costly hardware, e.g., a 50 watt transmitter and steerable parabolic antenna. A detailed analysis of the hardware requirements and the impact on PDS performance is needed to determine the desirability of this communication link. The TDRSS analysis needs to consider the probability of exclusive or shared usage, including sharing within the subject program or an unrelated program. NASA plans at the time of IUS design proposals did not include use of a TDRSS link with the IUS, nor with DoD payloads (Reference 15).

12.3 Mission Operations - Mission operational requirements can be divided into two phases; prior to separation from the Shuttle and after separation. Before separation, the primary concern is the extent of payload checkout required to make a launch decision. Ground and pre-deployment checkout considerations have been analyzed and the results are reported in this subsection. After separation, mission control is primarily a range safety consideration in the event of a system malfunction. This phase is analyzed in Section 14.

12.3.1 Payload Ground Checkout - A payload receives at least three checkouts before flight: factory, launch site and inflight pre-deployment. Confidence in the operational readiness of the vehicle is maximized if there is a high degree of similarity in the checkout techniques and equipment used. Since the Shuttle uses a computer for checkout, all ground checks should also be computer controlled with software similar to inflight checks.

Some payload peculiar ground equipment will be required; mostly power, coolant and test stimulus sources. The launch ranges are equipped with facilities that can read-out all telemetry data and test command receiver systems. Separate telemetry check-out equipment is not required. A typical telemetry data center is shown in Figure 114. This is the Vandenberg Automatic Data Equipment (VADE) which is used for programs such as Minuteman, Titan, and RVTO. The CDC 1700 computer is used to check each data sample against a pre-determined limit. Data within the limit is ignored and data outside the limit is processed by the CDC 3300 for recording and display. Some critical data is recorded on strip chart recorders from a CDC 1700 output. This facility has two CRT's, a line printer and thirteen oscillograph recorders. The combined capability can display 150 data channels and record 232 channels, all in real time. Obviously, to analyze all these data in real time requires 4 to 5 persons to monitor the equipment. With all data permanently recorded on tape, playbacks can be used to help resolve anomalies, and the need for fast real-time decision-making is reduced. Functionally, the data processing activity of this system and the Shuttle system are quite similar. The differences are that data are permanently recorded during ground checkout so that extensive analysis can be performed without repeating the checkout itself and a much larger quantity of data can be viewed simultaneously.

After the payload has been loaded onboard Shuttle at the launch pad, the Shuttle checkout system can be used in parallel with a checkout using range facilities. This will build confidence in the inflight checkout since any discrepancies can be quickly traced to either a hardware fault or a software error, and the problem corrected and a retest performed.

Ground checkout does not appear to impose any new requirements on DoD payloads. However, it is imperative that Shuttle launch delays due to payload malfunctions be minimized. This impacts the design philosophy for the payloads and is discussed further in Sections 17 and 19. On the other hand, pre-deployment payload checkout during orbit requires detailed considerations.

12.3.2 Payload Pre-deployment Checkout - The Shuttle Orbiter Systems Management equipment is shown in Figure 115. This equipment can be used to check a payload if the payload data rate does not exceed 64 KBPS, or if the payload data can be segregated into 2 to 5 links up to 64 KBPS each. The computer receives payload data from the Shuttle PCM system and checks it against limits stored in the Mass Memory. Data that exceeds tolerances is sent to the Caution and Warning system where it can turn on a light or sound an audio alarm. Selected data can be dis-

DATA PROCESSING SYSTEM (VAFB TDC-20 SYSTEM)

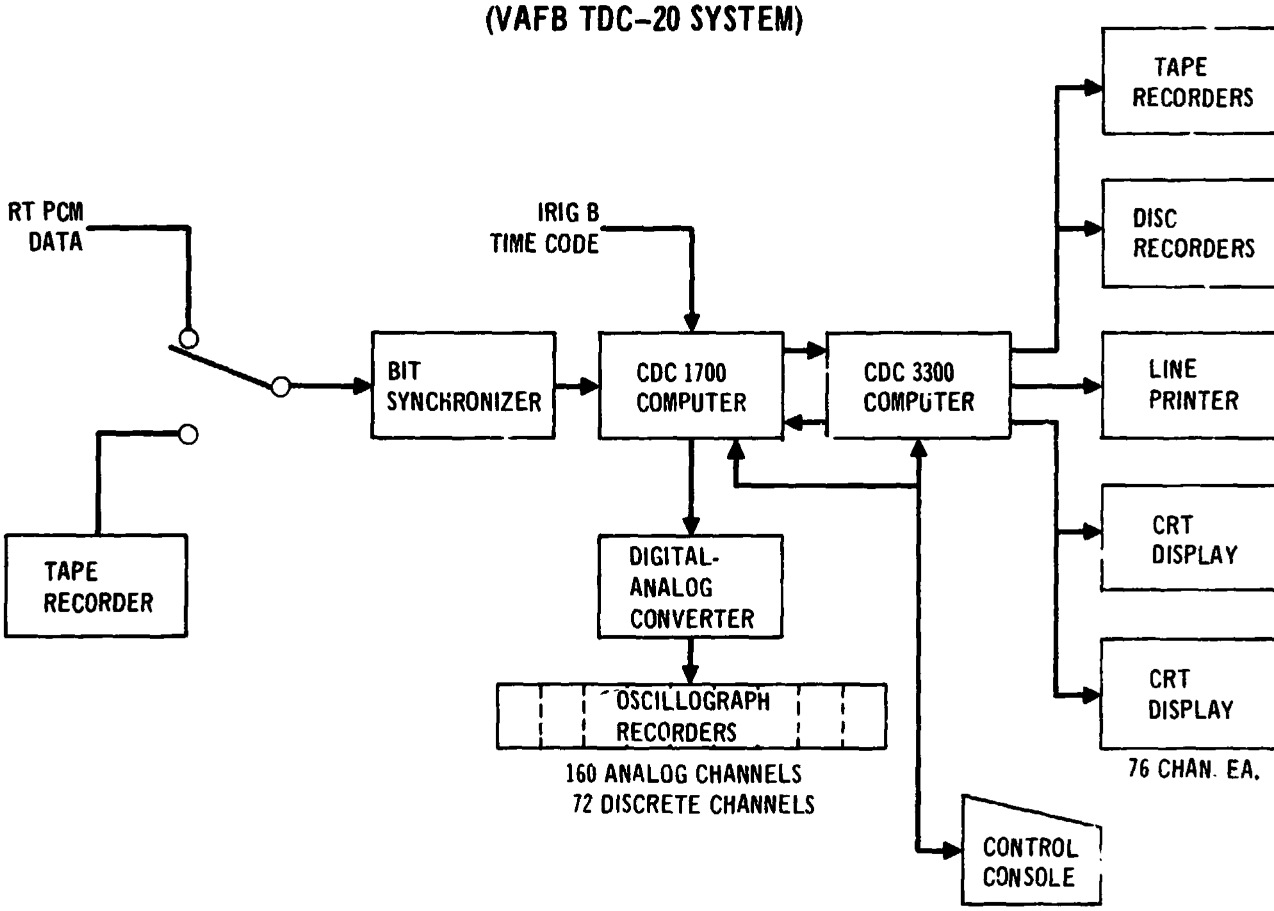


FIGURE 114

SHUTTLE SYSTEMS MANAGEMENT

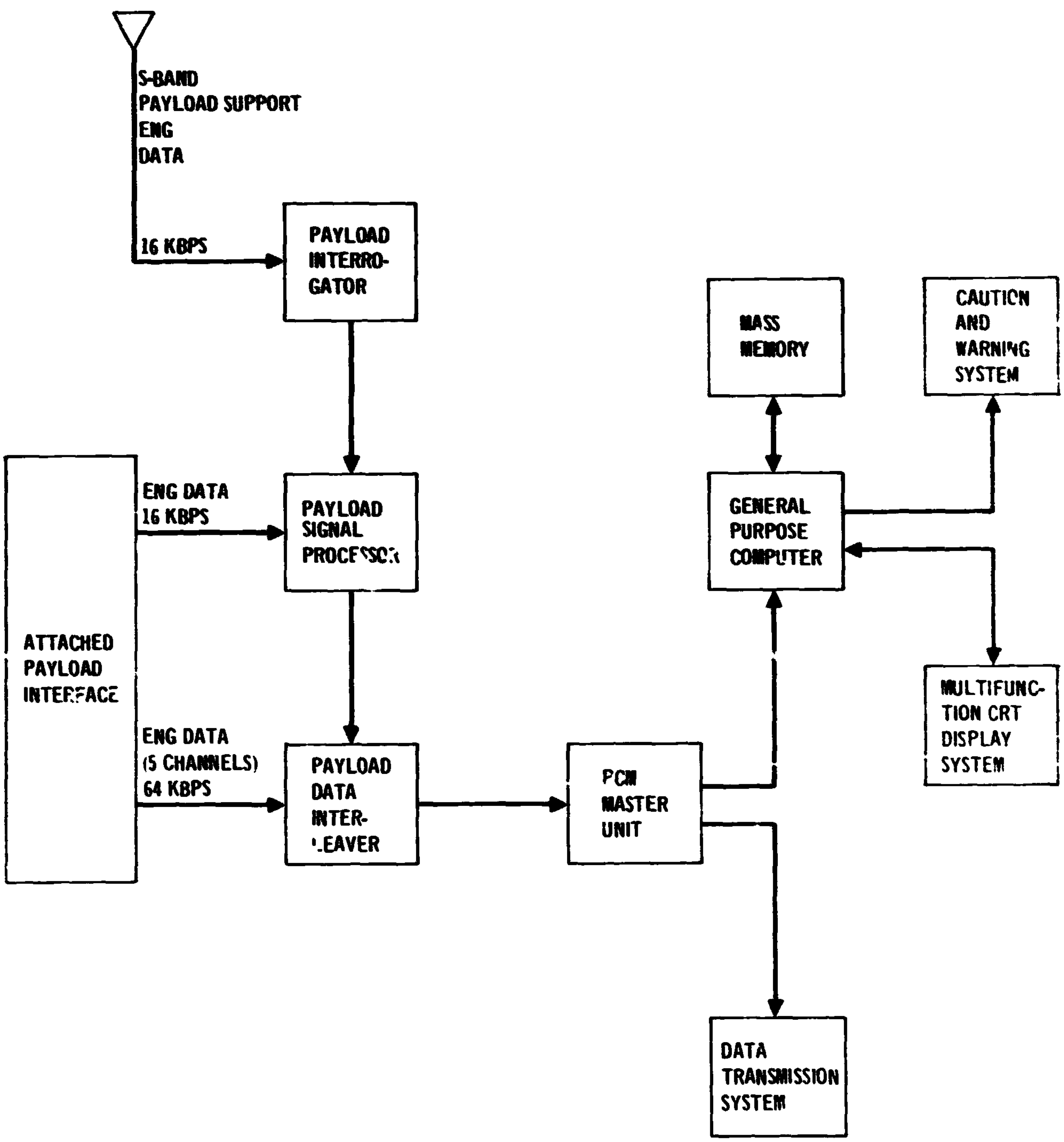


FIGURE 115

played on a multi-function CRT Display System. The display is in "pages" of a pre-determined format, with the crew being able to select the desired page, i.e., combination of data channels. 21 lines out of 26 on the CRT are allocated to data, by allowing 12 characters for data and data identification, 4 data channels can be displayed per line or 84 per page. The Shuttle Orbiter computer is an IBM 4 pi series model AP-101. This computer is in two packages, one containing the central processing unit (CPU) and the other the Input/Output Processor (IOP). The 65,536 word memory is divided between the two packages and acts as a single memory. Each package has a volume of 1500 cu. in. and weighs 55 pounds. Total power required is 640 watts.

Typically, a pre-deployment checkout will require looking at booster, deployable bus and RV data. In the simplest case, if the quantity of data to be analyzed is less than 64 KBPS, the entire checkout can be performed using only one Shuttle Engineering Data interface. A possible combination of data channels totalling 326 is shown in Figure 116. All the data can be contained in 4 pages (82 channels per page) and viewed in sequence on the CRT. The more severe case consists of 5 individual data links of 64 KBPS. The interface now consists of five engineering data links between Shuttle and payload (Figure 117). Assuming approximately 300 data channels per link, 18 pages with 84 channels per page will be required for a complete display. The time required to analyze 18 pages may be incompatible with mission timing and a more automated test or a different test system will be required. This situation represents the maximum capability of Shuttle. Additional hardware would have to be added to accommodate more RV's or higher data rates. Figure 117 also shows typical booster and RV data rates for current ground launch system. A minimum essential data rate for typical boosters like Transtage, Atlas E/F, and Delta exceed the single channel data rate of 64 KBPS. The RV data rates are less than the 64 KBPS. If the booster data rates cannot be further reduced, pre-deployment checkout will have to be made through a Shuttle to ground data link.

A maximum complexity condition is shown in Figure 118 where there are not only many RV's but also more than one data link per RV. This condition can be handled in three ways. The payload interface can include a switching capability so that the several links can be tested sequentially using the Shuttle System Management equipment. However, the time required would be extremely long, there would be software problems because of the many display pages needed and the large memory needed to store limit data and calibration conversions. A second possibility is

PRE-LAUNCH CHECKOUT USING ONE 64 KBPS LINKS

DATA CHANNEL SAMPLE RATE (SPS)	DATA CHANNELS				USAGE
	BOOSTER	RV BUS	RV #1	RV #2	
200	4	-	-	-	ACCELEROMETERS
					RATE GYROS
100	8	8	3	3	PROPULSION PRESSURES
50	10	4	6	6	CRITICAL TEMPS. CRITICAL DISCRETES
20	40	20	4	4	VOLTAGES
					CURRENTS
10	40	32	12	12	PRESSURES
5	60	32	10	10	DISCRETES
TOTAL CHANNELS	162	94	35	35	326 CHANNELS TOTAL
DATA RATES*(KBPS)	32	16	8	8	64 KBPS TOTAL

*SYNC WORD INCLUDED IN TOTAL

FIGURE 116

SEQUENTIAL PRE-LAUNCH CHECKOUT USING FIVE 64 KBPS LINKS

- MAXIMUM CAPABILITY USING CURRENT SHUTTLE DESIGN

BOOSTER	64 KBPS
RV BUS	64 KBPS
RV #1	64 KBPS
RV #2	64 KBPS
RV #3	64 KBPS

- ADDITIONAL RV'S OR GREATER DATA RATES WILL REQUIRE ADDED SHUTTLE HARDWARE
  - TYPICAL BOOSTER
  - TYPICAL RV

TRANSTAGE*	384	KBPS	REENTRY-F	40 KBPS**
ATLAS E/F*	225	KBPS	FIRE	47 KBPS**
DELTA*	69.2	KBPS	RVTO	58 KBPS

*REDUCED % OF CHANNELS ASSUMED NECESSARY FOR PRE-LAUNCH CHECKOUT

**PCM EQUIVALENT OF PDM/FM SYSTEM

FIGURE 117

OPTIONS TO MEET DOD MAXIMUM REQUIREMENTS

MAXIMUM PAYLOAD – 8 RV'S, 1 RV BUS, 1 IUS
MAXIMUM DATA LINKS – 5 PER RV, 1 PER BUS, 1 PER IUS

OPTIONS:	HARDWARE COST	SOFTWARE COST	CHECKOUT SPEED	CREW CONTROL
• USE 5 SHUTTLE ENGINEERING DATA INTERFACES – HARDWARE REQ'D-SEQUENTIAL SWITCHING CAPABILITY – CREW TESTS SEQUENTIALLY USING EXISTING PERFORMANCE MONITOR – GROUND PERSONNEL CAN SUPPORT	LOW	HIGH	SLOW	GOOD
• USE 1 SHUTTLE SCIENCE DATA INTERFACE – HARDWARE REQ'D – DIGITAL DATA INT'RL 'VR – GROUND PERSONNEL DO ALL TESTING – CREW COORDINATION VIA VOICE LINKS	LOW	LOW	FAST	POOR
• DOD FURNISHED OPERATION & CHECKOUT CONSOLE – HARDWARE REQ'D – COMPUTER, DISPLAYS, CONTROLS, PAYLOAD INTERFACE DATA BUS – CREW HAS PRIME TEST RESPONSIBILITY – GROUND PERSONNEL CAN SUPPORT IF DATA ALSO SENT VIA SCIENCE DATA DOWNLINK	HIGH	MED	MED	GOOD

FIGURE 118

to interleave all the data and use one of the science data downlinks such as the 5 MBPS direct link or the 50 MBPS satellite link. This transfers the checkout problem to a ground station where it is assumed a computer capability, such as that shown in Figure 115, can be used to perform a relatively fast checkout. However, the Shuttle crew is now out of the loop and must rely on voice communication with the ground to determine the status of the payload. A third possibility is to put an operation and checkout console onboard the Shuttle that is tailored to meet the payload requirements. This allows the crew to retain control of the payload; in addition software development is simplified in that it need not be integrated with or conform to Shuttle checkout software.

In summary, onboard pre-deployment payload checkout is possible for payloads with fairly low data rate requirements on the order of 64 KBPS. However, for PDS's with high booster data rates and many RV's onboard checkout becomes cumbersome. As a result, either transfer of checkout data to a ground station or the installation of an onboard checkout console is required.

12.3.3 Shuttle Onboard Checkout Console - A block diagram of an Operation and Checkout Console is shown in Figure 119. PCM data enters the bit synchronizer where jitter is removed and a clock and data signal produced that is compatible with the data formatter input. The data formatter converts the serial digital data into parallel data words and adds a format location tag. The block diagram assumes all input data coming from a single source. This is compatible with the concept that the same data is being interleaved in the PDS and being sent over a high rate data link to a ground station for back-up processing. If this is not done, the data signals can be sent to the checkout console individually and selected either manually or automatically. If data from multiple sources is to be intermixed during processing (a mixed source display is required), then a bit synchronizer and data formatter will be required for each data source.

The data comparator performs the same function as the CDC 1700 computer in Figure 114. In the present application, a microprocessor such as the Intel 8080 is envisioned for the comparator. The function of the data comparator is to compare each data sample with the preceding sample of that same data channel. If the current data sample is unchanged within a specified tolerance from the preceding sample, the data is ignored and no further processing is performed. Data which exhibits a significant change is sent to the data processor where it is multiplied by the cali-

SHUTTLE PAYLOAD
OPERATION AND CHECKOUT CONSOLE

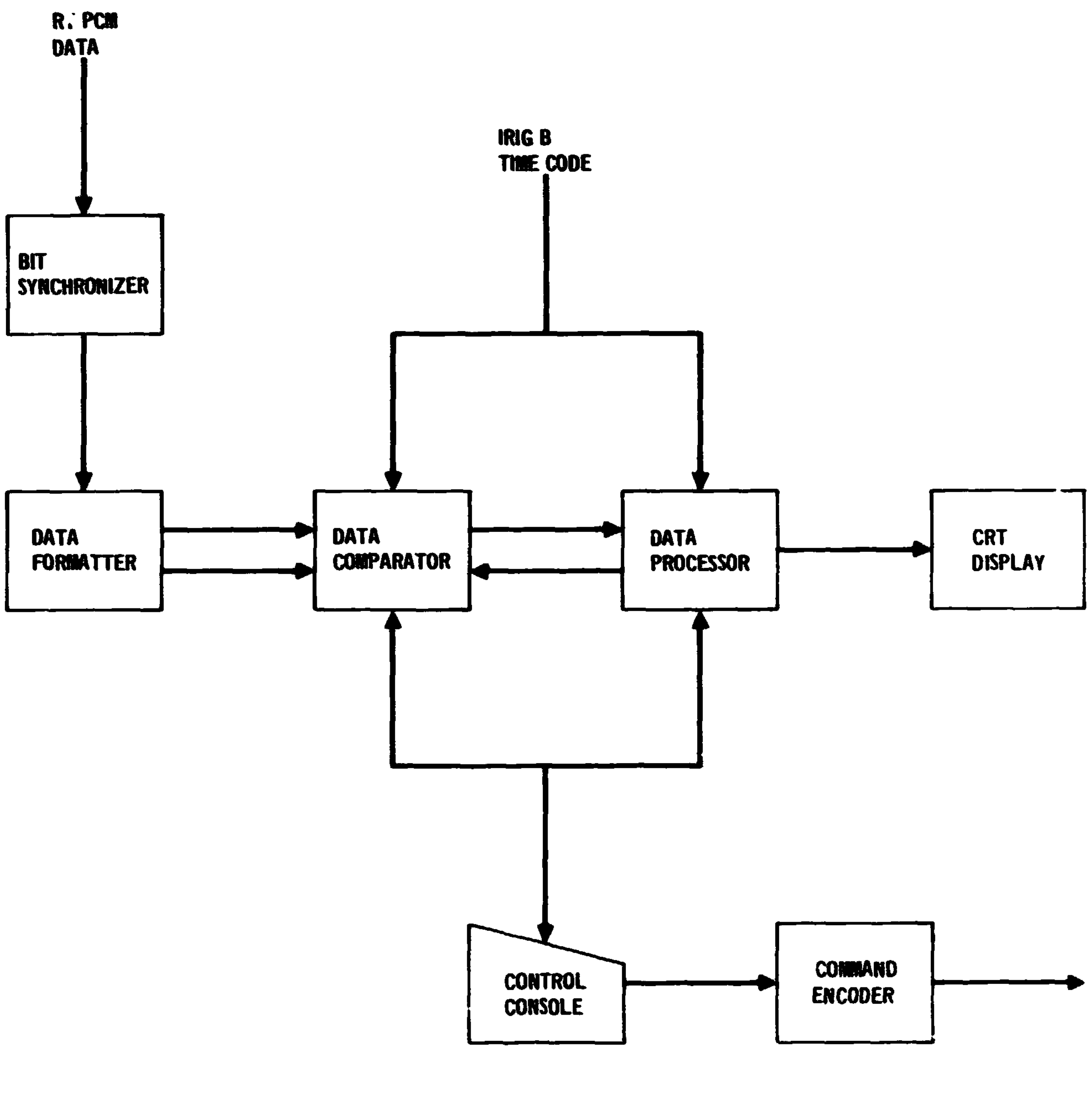


FIGURE 119

bration factor and alpha-numeric identification added. Then it is transmitted to the CRT display for display in engineering units. The data processor can also be used to trigger the Caution and Warning system or an alert signal on the checkout console whenever a parameter exceeds some pre-determined limit.

The control console is used to select the data page to be displayed and to modify the program when necessary, such as changing a number stored in memory. For some tests, a stimulus command may be required and this is generated at the control console and sent to the payload in standard command format through the command encoder. In addition to test functions, the control console can generate update commands for the PDS guidance system.

The data rate at which any checkout system can operate is difficult to determine without writing the computer program and then calculating the processing time for the specific computer selected. However, even the slower computers can operate faster than a viewer can evaluate a single display. One commercially available system built around a PDP-11/40 computer can process data at a rate of 80 to 100 KBPS. This is without any data comparator (see Figure 119). By adding a microprocessor to remove redundant data, the rate increases (depending on the degree of redundancy) and can go as high as 750 KBPS to 1 MBPS. In this system, the CRT displays 88 data channels per page which is updated at the rate of 6 times per second. This rate is adequate to maintain the intensity of the display but any dynamic data, which is changing at a faster rate, will be blurred. However, during checkout vehicle systems are usually quiescent and little dynamic data is generated. In ground checkout, recording oscillographs are used for dynamic data.

The control console, as conceived, will be capable of monitoring the data from the PDS, screening it for critical parameters, and displaying launch critical information. This will significantly reduce the amount of crew activity involved with pre-deployment payload checkout.

12.3.4 Planning Guidelines - In planning a payload checkout system, two data rates are critical; 16 KBPS and 64 KBPS. 16 KBPS is the maximum data rate between a separated payload and the Shuttle. 64 KBPS is the maximum data rate from a single Shuttle Engineering Data interface (see Figure 115). Assuming that the actual data rates from the two vehicles exceed these values, there are two ways to achieve compatibility with a single Engineering Data interface. Both require special circuitry in the PDS and RV PCM systems.

Since most data channels will be quiescent while a payload is carried aboard Shuttle, high data sample rates are not required. Thus, one solution which should be acceptable is to slow down all the sample rates in the PCM system by slowing down the clock in the timing subsystem. For example, if the normal data rate is 256 KBPS, a divide-by-four counter at the clock output would lower the rate to 64 KBPS. This technique is preferred, since it will verify the integrity of all data channels even though some may not contain any useful information. The other solution selects only those channels that are launch-decision critical and combines them into a separate PCM output signal for test use only, or possibly, for an immediate post-separation Shuttle link (Reference Section 12.2.2).

It can be argued that by the time a payload has passed a sequence of ground tests, including a launch pad mated Shuttle test, there is little need for extensive inflight testing. Since the checkout is primarily performed by one crew member, making 84 real time decisions, i.e., reviewing one page of data, should be sufficient to determine launch readiness. If time permits, of course, another page of 84 different parameters can be evaluated. On the other hand additional time may be required to repeat a test or modify a test routine to firm up a tentative decision.

In summary, the checkout equipment and method used for any specific payload, including a trade-off between an onboard checkout console and a Shuttle to ground checkout link, will be primarily determined by two factors:

a. the minimum acceptable vehicle PCM data rate for checkout

b. the ability of SCF/STDN ground personnel to support an inflight checkout

This latter factor will be discussed in Section 15.

The next section describes the electrical and coolant interfaces which in some ways affect the communications interfaces discussed in this section.

13.0 PAYLOAD SERVICING

In addition to the payload interfaces related to checkout and communications, the Shuttle also provides electrical power and coolant interfaces that are needed for payload support during ground checkout and captive flight. These interfaces are accommodated through panels at either the forward or aft end of the payload bay as shown in Figure 120.

There are also external umbilical panels available for ground servicing. These panels are located just aft of the payload bay with the H_2 fuel servicing on the left side and the O_2 oxidizer servicing on the right. Ground coolant connections are also available on the right hand panel (note: Reference 4 is not clear but this seems to be a separate water loop independent of the Shuttle-payload coolant system). Electrical connections are available on both panels. In addition, space is provided on the left side of Shuttle for a payload peculiar prelaunch panel. This panel can be used for any electrical or fluid connections not possible or practical via the standard umbilical panels. However, it will be disconnected 4 hours prior to launch. The standard panels disconnect at launch.

13.1 Electrical Power - The Shuttle's electrical power supply is derived from fuel cells and is normally in the range from 27 to 32 volts DC with a maximum peak-to-peak ripple of 1.6 volts. The power supply at the aft flight deck and at the aft payload bay can go as low as 24 volts. Fuel cell voltage can be as high as 40 volts under no-load conditions.

The electrical power and heat dissipation capability is shown in more detail in Figure 121. Electrically, power is available from three sources, i.e., fuel cell, main bus, and aft bus. Normally, the dedicated fuel cell is the preferred power source for payloads, since this will isolate the payload from any transients or other fluctuations on the Shuttle buses. As indicated in the heat dissipation columns, the usable power in orbit is limited to 6.3 kw by the heat dissipation system. If an additional radiator is provided for the payload, this power can be increased to 8.5 kw continuous and up to 12 kw for a 15 minute period once every three hours.

Since the solid rocket motor IUS has not been completely designed, the power required by an IUS booster is unknown. Boosters such as Transtage and Delta have power dissipation similar to what may be expected from an IUS. The Transtage vehicle requires between 500 and 600 watts; the proposed Delta IUS requires 470 watts continuous power and 1520 watts for a single 13 minute period. Payload checkout is not likely to occur during any booster peak load condition and two RVs are not

PAYLOAD BAY INTERFACES

SYSTEM	LOCATION STA	SIDE	RATING
Electrical			
Fuel Cell	695	RH	12 KW max
Main Bus	695	RH	8 KW max
Aft Bus B	1307	LH	2 KW max
Aft Bus C	1307	RH	2 KW max
Aft Crew Compartment	576	LH/RH	1 KW max
Coolant			
Primary loop	586	LH	2,000 lb/hr
Secondary loop	586	LH	2,000 lb/hr
Propulsion			
Liquid O_2	1307	RH	5 inch line
Gaseous O_2	1307	RH	1/2 inch line
He (ambient)	1307	RH	1/2 inch line
He (cold)	1307	LH	1/2 inch line
Liquid H_2	1307	LH	3 inch line
Gaseous H_2	1307	LH	1/2 inch line

FIGURE 120

PAYLOAD ELECTRICAL POWER AND HEAT DISSIPATION

MISSION PHASE	ELECTRICAL POWER (KW) FUEL CELL AVE	FUEL CELL PEAK	MAIN BUS AVE	MAIN BUS PEAK	AFT BUSES AVE	AFT BUSES PEAK	HEAT DISSIPATION BTU/HR	HEAT DISSIPATION KW	CONDITION
GROUND OPERATION (GSE PWR)	1	1.5	1	1.5	3	4	TBD	TBD	NORMAL CHECKOUT
	7	12	5	8	3	4	29,000	8.5	SHUTTLE POWER-DOWN
ASCENT/ DESCENT	1	1.5	1	1.5	2	3	5,200	1.52	DOORS CLOSED
ON-ORBIT	7	12	5	8	3	4	29,000	8.5	PAYLOAD RADIATOR
	6	TBD	5	8	3	4	21,500	6.3	DOORS OPEN
	1	1.5	1	1.5	2	3	5,200	1.52	DOORS CLOSED

NOTES: 1. THERE ARE TWO AFT BUSES, EACH SUPPLYING 1/2 OF SPECIFIED POWER.

2. ASCENT/DESCENT PEAK POWER LIMITED TO 2 MINUTES

3. ON-ORBIT PEAK POWER LIMITED TO 15 MIN PER 3 HOURS

FIGURE 121

likely to require checkout simultaneously. Therefore, a typical power budget would be:

Power Available	6,300 watts
Booster-500 watts	
RV #1-1,000 watts	
or RV #2-1,000 watts	
	1,500 watts
Margin	4,800 watts

Thus, the planned electrical loads seem to be well within Shuttle capability.

Electrical power is provided in the aft crew compartment for payload control and checkout equipment. This is rated at 1 KW maximum and 750 watts normal. This is standard Shuttle bus power between 24 to 32 VDC. A preliminary estimate of power requirements for an Onboard Checkout Console (such as shown in Figure 119) indicates 750 watts should be adequate. Power allocation is as follows:

Bit Synchronizer	30
Data Formatter	60
Data Comparator	150
Data Processor	200
Cathode Ray Display	175
Control Console	75
Command Encoder	60
	750 watts

The first four functions above provide a computer capability similar to the Shuttle computer. However, the Shuttle computer interfaces with up to 24 data bus devices, performs more functions and requires a larger memory than is needed for this data processing application. Thus, the data processing function can be accomplished with less power (440 watts) than is required for the Shuttle computer (640 watts).

The estimates above are based on power required by laboratory counterparts. Further design effort is required to determine power required by spaceborne equipment. Except for the cathode ray display, modern data processing equipment uses integrated circuits that operate at low voltages. Direct operation from 28 VDC will require less power than the laboratory equipment which requires conversion of 110 VAC to low voltage DC.

In conclusion, the electrical power interface is adequate for the DoD payloads.

13.2 Coolant Interfaces - Heat dissipation is provided by two coolant loops, both accessible on the left side of the payload bay. Coolant may be water, Freon

21 or Flutec PP50. Each loop can operate at a maximum flow rate of 2,000 lbs per hour. The normal dissipation capability described in Figure 121 is 21,500 BTU/hr (6.3 kw). By adding another heat exchanger, the payload heat output can be a maximum of 29,000 BTU/hr (8.5 kw). The additional unit is chargeable to the payload. This heat exchanger weighs 43.3 pounds and has a volume of 737 cubic inches. Normal Shuttle in orbit operation will be with payload bay doors open to maximize heat dissipation. With payload bay doors closed, the maximum heat load is 5,200 BTU/hr (1.52 kw). Typical operating parameters are 200 psi inlet pressure, 124°F inlet temperature and 45°F outlet temperature. The coolant pump to operate the loop is furnished by the payload. Details of heat dissipation concepts which interface with this system are provided in Section 14.

14.0 HEAT DISSIPATION CONCEPTS

The purpose of this analysis is to explore alternate concepts for rejecting heat from the reentry vehicles and the booster when they are in the Shuttle cargo bay. The exact amount of heat to be dissipated, duration and RV mass that could be flown in the 1980's is of course an open question. The following system characteristics compatible with example cases 2 and 4 of Section 3.2 were assumed for the analysis:

RV Mass: 1000 Lb_m
Heat Load: 1000 watts
Operation Time in Shuttle Bay: up to six hours
Initial Temperature: 70°F
Maximum Allowable Temperature: 100 to 150°F

Current ground launched reentry technology experiments are powered-up for approximately ten hours prior to launch and cooled by air conditioning. Shuttle will have similar capabilities with conditioned gas flow through the cargo bay prior to launch. Consequently, it is assumed that this gas flow will dissipate all waste heat prior to Shuttle launch. Missions involving KSC launched experiments that reenter at Kwajalein are air borne for 6 hours for a 4 orbit mission compared to 30 minutes in space for a current ground launch.

The Shuttle has primary and secondary heat exchange connectors up forward in the cargo bay (station 586). The reentry technology vehicle may or may not be installed near these connectors. If these connectors were used, quick disconnects, plumbing lines, fluid reservoirs, and pumps would have to be qualified Shuttle compatible. Also, the reentry experiment will require approximately half the cargo bay so the heat rejection requirements of the other payload(s) must be considered. Notwithstanding these design problems, heat rejection systems using the Shuttle heat exchanger were considered.

Passive, semi-active and active systems were investigated. The two passive systems considered were: radiation from the RV and a heat sink/insulated RV. The two semi-passive systems were: water boiler and heat pipes. The three active systems were: gaseous nitrogen, internal RV Freon and internal RV water cooling. The advantages and disadvantages of each system are listed in Figure 122.

The characteristics of each concept are defined further in Figures 123, 124 and 125. For the system described in Figure 123 which relies on heat rejection from the RV by radiation, only 300 watts can be rejected under the most favorable conditions

ALTERNATE HEAT REJECTION CONCEPTS

CONCEPT	ADVANTAGES	DISADVANTAGES
RADIATOR WITH EQUIPMENT MASS	SIMPLE, INEXPENSIVE, LOW WEIGHT	SELECTIVE COATING REQUIRED WILL NOT FUNCTION WELL IN DIRECT SUNLIGHT REENTRY HEAT PROTECTION SYSTEM WILL RETARD HEAT FLOW TO OUTER SURFACES OF RV
VEHICLE MASS WITH PCM	SIMPLE, INEXPENSIVE	ADDITIONAL MASS & PACKAGING OF PCM
WATER BOILER	LOW WEIGHT	REMOVAL OF WATER VAPOR, COLD-PLATED EQUIPMENT
HEAT PIPE	SIMPLE SYSTEM QUICK DISCONNECT TUBES NOT NEEDED	ADDITION OF HEAT PIPES TO RV ADDITION OF INTERFACE LOAD PLATE TO BOOSTER
N_2 COOLED	MINIMUM CHANGE TO RV	NEEDS PRESSURIZED RV INTERFACE CONNECTORS, N_2 BOTTLE, BOOSTER H/X
FREON COOLED	LIGHTER WEIGHT THAN N_2	HIGH PRESSURE SYSTEM RV COLDPLATED EQUIPMENT INTERFACE CONNECTORS, BOOSTER H/X
WATER COOLED	LOWER PRESSURE THAN FREON SMALLER PUMP NEEDED	RV COLDPLATED EQUIPMENT INTERFACE CONNECTORS, BOOSTER H/X

NOTE: PCM = PHASE CHANGE MATERIAL H/X = HEAT EXCHANGER

FIGURE 122

PASSIVE HEAT REJECTION SYSTEMS

RADIATION FROM RV

- 8 FT^2 EFFECTIVE RADIATION AREA
- α/ε = 0.3 NECESSARY IN DIRECT SUN FOR ZERO HEAT REJECTION
- FOR α/ε < 0.3 THERE IS VERY LITTLE HEAT REJECTION IN DIRECT SUN
- MAXIMUM OF 300 WATTS REJECTION WHEN LOOKING AT DEEP SPACE

HEAT SINK-INSULATED VEHICLE

	T_{MAX} 100°F	150°F
TIME TO REACH T_{MAX} USING VEHICLE MASS	2.0 HR	5.4 HR
*WEIGHT OF PCM TO KEEP VEHICLE BELOW T_{MAX} FOR 6 HRS	102 LB_m	21 LB_m

*PCM LATENT HEAT - 100 BTU/LB_m

NOTE: α = SOLAR ABSORPTIVITY
ε = EMITTANCE
PCM = PHASE CHANGE MATERIAL

FIGURE 123

SEMI-PASSIVE HEAT REJECTION SYSTEMS

WATER BOILER

	T_{MAX} 100°F	150°F
*WEIGHT OF WATER NECESSARY TO KEEP VEHICLE BELOW T_{MAX} FOR 6 HRS	10.2 LB_m	2.1 LB_m

*HEAT OF VAPORIZATION - 1000 BTU/LB_m

HEAT PIPES

- EQUIPMENT MOUNTED ON HEAT PIPES
- HEAT PIPES ROUTED TO PLATE AT REAR OF RV
- CONDUCTION TO BOOSTER HEAT REJECTION SYSTEM USING A SPRING OR PNEUMATIC LOADED PLATE

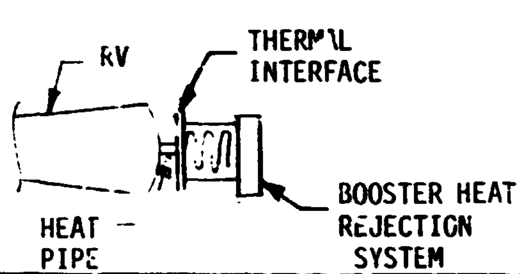


FIGURE 124

ACTIVE HEAT REJECTION SYSTEMS

GASEOUS NITROGEN COOLED

- o OPEN SYSTEM NOT CONSIDERED BECAUSE OF EXCESSIVE WEIGHT
- o CLOSED SYSTEM REQUIRES 200 FT^3/MIN
- o RV TO BOOSTER GAS DISCONNECT LINES
- o PRESSURIZED RV
- o GAS H/X & BLOWER ON BOOSTER

INTERNAL RV FREON LOOP

- o 400 LB_m/HR FLOW RATE
- o RV TO BOOSTER DISCONNECT LINES
- o FREON H/X & PUMP ON BOOSTER
- o COLDPLATED RV EQUIPMENT

INTERNAL WATER LOOP

- o 100 LB_m/HR FLOW RATE
- o WATER H/X & PUMP ON BOOSTER
- o COLDPLATED RV EQUIPMENT

FIGURE 125

and heat paths through the reentry heat protection system would have to be designed. The heat sink system of Figure 123 has been in use for high heat generation components. A factor of five reduction in PCM material mass can be realized by qualifying RV components to 150°F instead of 100°F.

The Water Boiler system of Figure 124 seems very light, but the additional weight for the fluid loop for transporting the heat to the water boiler and the cold plates must be considered. This system may alter current RV design philosophy. Higher temperature RV components significantly reduce water coolant requirements.

Heat pipes described in Figure 123 offer an excellent means of controlling the temperature of the RV components. Heat pipes would also have to be incorporated into the RV design.

The active systems of Figure 125 all reject heat to the booster which in turn must reject this energy along with its own waste heat to either the Shuttle or to space (by a radiator mounted on the side of the booster).

The recommended passive, semi-passive and active systems are listed in Figure 126. Further study of these approaches should be conducted to ascertain cost, impact on RV design, impact on Shuttle and the overall complexity. An alternative to developing a quasi-steady state heat rejection system is to operate the RV at high powers for only five minutes and to design hardware with low waste heat characteristics. Alternately, it also appears that some redesign of the RV subsystems will be required to accommodate any of these heat rejection concepts.

RECOMMENDED HEAT REJECTION SYSTEMS

CONCEPT 1 - RADIATOR WITH PCM

- o RV COVERED WITH RF TRANSPARENT ASTROQUARTZ (α/ε = .25)
- o HEAT SINK SUPPLEMENTED WITH PCM
- o ATTITUDE CONSTRAINTS FOR LONGER THAN NOMINAL MISSION DURATION

CONCEPT 2 - HEAT PIPE

- o HEAT PIPES ATTACHED TO HIGH HEAT REJECTION RV EQUIPMENT
- o CONTACT CONDUCTANCE TO BOOSTER USING PNEUMATIC LOADED PLATE
- o DECOUPLE BEFORE SPINUP BY RELEASING GAS PRESSURE

CONCEPT 3 - WATER COOLED RV EQUIPMENT

- o COLDPLATE HIGH HEAT REJECTION RV EQUIPMENT
- o QUICK DISCONNECT WATER LINES MECHANICALLY DECOUPLED PRIOR TO SPIN UP

FIGURE 126

15.0 RANGE SAFETY

After separation from the Shuttle, mission control requires that the Mission Director have a real time knowledge of the PDS position and have available a command link to the PDS. Range safety is primarily concerned with possible booster malfunctions that can cause a change in trajectory and result in payload impact in an undesirable location, such as a populated land mass. The need for safety rules, related to Shuttle-launched payloads, is discussed in Section 15.1 Tracking system accuracies and equipment are described in Section 15.2. Section 15.3 describes the command link requirements. Section 15.4 provides the tracking, communications, and range safety considerations for the example cases of Section 16.

15.1 Safety Considerations - In considering range safety, we have reviewed SAMTEC Manual 127-1, Range Safety Requirements (Reference [illegible]). This document divides vehicles into four categories for safety purposes. Reentry vehicles are included under "Ballistic Missiles/Space Vehicles". However, the document assumes that all vehicles are launched from VAFB and requirements for launch from an orbiting spacecraft are not stated nor can they be readily inferred. The following quotations illustrate the problem.

> "1.2.2.1 Flight Termination Systems - All missiles and space vehicles flown on the range must be equipped with flight termination systems... These systems must be capable of terminating thrust until the impact point is established or orbital injection occurs."
>
> "1.2.4.2 At least one tracking aid should be carried on all launch vehicles..."
>
> "2.5.1 General Vehicle Data. The following items are required for each missile flight."
>
> "2.5.1.7 Expected impact point...for each stage or jettisoned body..."
>
> "2.5.1.8 Impact dispersion data for each stage and jettisoned body."
>
> "2.5.2 Trajectory Data Requirements...from launch up to a point in flight where effective thrust of the final stage has terminated or to thrust termination or burn which places the vehicle in orbit."

We can infer from the above that an unpowered RV need not be equipped with any command-destruct system. This conclusion is supported by the fact that downrange sites such as Kwajalein and Canton Island do not have command transmitters. On the other hand, the booster will require command-destruct provisions.

There are four events in the mission that are critical. These are: separation from Shuttle, orbit plane change burn, deorbit burn, and spacing burns. PDS separation is under control of Shuttle. The Shuttle Payload Interrogator provides two-way communication with the PDS up to a range of 20 NMI. Any safety problems associated with separation must be considered in designing the payload interface; ground range safety personnel will not normally be involved. During the three burn periods, it will be desirable to have a command capability that can terminate the burn or otherwise modify PDS behavior. The best solution is to plan the burn periods while the PDS is within range of a ground station that has command capability. Since downlink telemetry data is also of interest during a burn, this is a further incentive for such burn scheduling. In some of the missions being studied in Section 16 this is not possible. For instance, the deorbit burn locations are east of the southern tip of Africa for case 2 or 3 of Section 16 where command capability can only come from a ship. However, cases 1, 4 and 5 will be within tracking range of Orroral, Australia shortly after deorbit burn. Therefore, any departures from planned trajectory can be detected and appropriate action taken. For Poker Flat missions, tracking from KMR, Hawaii or Guam is also probable for some opportunities.

While the PDS is not a launch vehicle in the usual sense, a tracking beacon may be required. It would seem desirable to have the various agencies concerned with range safety give consideration to defining requirements for Shuttle launched reentry vehicles. This would resolve the need for tracking beacons and also identify a minimum STDN-SGLS compatible command link. It is anticipated that some tracking of the PDS will be required for range safety purposes.

15.2 Tracking - Real time knowledge of position can be obtained by several methods:

a. ground radar tracking
b. coherent command and telemetry signal (SGLS or STDN)
c. Tracking and Data Relay Satellite (TDRS)
d. Global Positioning Satellite System

Methods a. and b. require line of sight contact between the PDS and ground. Method c. relays data from PDS to satellite to ground. Method d provides position fixes for the PDS which it must then relay to the ground. The accuracy of these systems is compared in Figure 127. Obviously, radar tracking is more accurate. The STDN tracking and the SGLS system use a PRN ranging code which is transmitted to the PDS by the command transmitter and returned to the ground via the telemetry transmitter. PDS range is determined from the codes round-trip travel time. Many fac-

TRACKING SYSTEM ACCURACIES

	UNITS	RADARS: FPS-16 (VAFB) (1)	RADARS: MPS-36 (KMR & VAFB) (2)	RADARS: VERLORT (POKER FLAT) (3)	SGLS (USAF) (4)	STDN (NASA) (5)	TDRSS (6)	GLOBAL POSITIONING SYSTEM
AZIMUTH & ELEVATION	MRAD	±0.1	±0.2	±1.0	±2.0	±0.6	NA	NA
RANGE	FEET	±15	±9	±75	±78	±119	±2100	NA
RANGE RATE	FT/SEC	NA	±1		±0.2	±0.8	±1.7	±10
ALTITUDE	FEET	NA	NA	NA	NA	NA	NA	±33
POSITION	FEET	NA	NA	NA	NA	NA	NA	±26

(1) MAXIMUM ACCURACY - DOES NOT INCLUDE PROPAGATION ERRORS
(2) AT S/N = 20 dB OR APPROX. 100 NM FOR PDS TARGET
(3) AT MAXIMUM RANGE OF 2300 NM
(4) 1 SIGMA SPECIFICATION ACCURACY
(5) DATA IDENTIFIED AS "REPRESENTATIVE" IN REFERENCE 17
(6) CALCULATED FOR SHUTTLE IN REFERENCE 4. SAME REPORT GIVES ±730 FT FOR STDN ACCURACY

FIGURE 127

tors affect the accuracy including vehicle orbital parameters such as aerodynamic drag and atmospheric refraction. Vehicles in highly stable orbits can be tracked more accurately than one with maneuvering capability such as Shuttle. This explains (in part) the seeming conflict between notes 5 and 6 of Figure 127. Range rate, on the other hand, is determined by measuring Doppler shift in the coherent carrier. The range rate accuracy is therefore a function of electronic measuring capability and is not dependent on PDS dynamics. Consequently, range rate accuracy is good for the SGLS/STDN tracking.

The TDRSS uses the same tracking principles as the SGLS/STDN system. In view of the greater range (TDRS average range is 23,000 NMI) and greater uncertainty of TDRS position, the accuracy is not as good. The Global Positioning System, which uses 24 satellites, takes data from any 4 satellites and correlates it to get a more accurate position than can be obtained from a single TDRS. However, the satellite data is processed on the PDS for position information. The data must then be input to the telemetry system for relay to the ground. The data can also be used to update an inertial guidance system. In terms of equipment the Global Positioning System requires only an antenna, receiver and processor on the PDS but to use TDRSS as a tracking aid will require a high power transmitter and possibly a steerable antenna (See Section 12.2.4).

In general, ground tracking stations would be preferred when the PDS is within the line of sight. Unfortunately, conventional C-band tracking stations are relatively few in number. NASA has C-band radars at Bermuda, Hawaii, Tananrive and on the Vanguard ship. However, as TDRSS becomes operational, ground telemetry facilities at Hawaii and Tananrive are scheduled to be closed. If the entire station is closed, the tracking radars will no longer be available. The Air Force has C-band capability at Vandenberg, Hawaii and Canton Island. The army has several tracking radars at Kwajalein, including C-band. These radars can skin track a 1 square meter target to 1700 NMI and a beacon equipped target to 32,000 NMI. Since the Transtage has a radar cross-section varying from 7.3 sq. meters (head on) to 13.8 sq. meters (broad-side), these radars should have no difficulty skin-tracking the PDS any time it is above the horizon. (A PDS altitude of 500 NMI is equal to a line-of-sight range of 1700 NMI at the horizon.)

The Poker Flat Research Range is not as well equipped for tracking as KMR. It has three radars and receives support from USAF radars at Ft. Yukon. Two of the radars are short range X-band and the third is a NASA S-band VERLORT with a range of 2300 NMI to beacon equipped targets. This radar will be useful for tracking the

PDS and RV's. The USAF radars are primarily scanning radars and can provide some back-up capability, but should not be considered a lock-on tracking facility.

It seems to be NASA's intent to equip the IUS with transponders compatible with the PRN ranging code used by SGLS and STDN. Use of TDRSS is not planned for the IUS or DoD payloads (see Reference 15). Thus, tracking and communications will be primarily by the STDN station at Fairbanks, Alaska and Orroral, Australia and the SCF station on Hawaii and Guam. Skin tracking from KMR is possible and a beacon transponder is not necessary. However, range safety may dictate the use of a beacon.

The capability of Shuttle to track its own payloads (look-down capability) is severely limited. The range of the Payload Interrogator is only 20 NMI. The rendezvous radar can track to 300 NMI if the PDS is equipped with a transponder. There is a possibility that for some mission strategies the Shuttle can track the PDS through plane change burn. This is for those burns where the angular change is less than 5 deg. The Shuttle rendezvous radar is Ku-band and a PDS transponder for this frequency is not usable with ground track C-band or other lower frequency radars such as the Kwajalein KREMS group. A similar problem exists for S-band tracking at Poker Flat. The S-band VERLORT operates in the 2700-2900 MHz range which is above the S-band telemetry range of 2200-2300 MHz and slightly below the Kwajalein TRADEX frequency of 2950 MHz. Thus, a PDS usable for either impact site could have three beacons. It is more likely that they will be mission-tailored, have only one beacon, and skin tracking at other frequencies will be adequate. If the mission plan keeps Shuttle in reasonable range of the PDS, a Ku-band transponder should be the choice. The physically smaller payloads will be more difficult to skin track and should have a C-band transponder.

In summary, for typical DoD missions to KMR or Poker Flat tracking will be provided by some of the following: the STDN stations at Fairbanks, Alaska and Orroral, Australia; the SCF stations on Hawaii and Guam; the VERLORT S-band radar at Poker Flat (beacon track); the Army C-band tracking radars at Kwajalein (skin or beacon track); Shuttle tracking radar. Some missions could be tailored to allow Shuttle tracking during the plane change burn and Orroral, Australia, tracking during deorbit and spacing burns. However, in general, direct ground coverage of PDS burns will not be possible. Consequently, either a TDRS link will be required or a tracking ship will have to be stationed within line of sight of the PDS burn.

15.3 Command Links - During a PDS burn, commands may be required for range safety purposes. The decision to send a command originates with the Mission Direc- t~~. This decision is relayed to a tracking station within range of the PDS, or can be sent via TDRS. Unlike a TDRS telemetry downlink, a command uplink is easily implemented using the same command receiver and antenna used for a direct ground-to-space link. The antenna location on the PDS will probably differ from the location for a direct ground link but, normally, reliability considerations require use of 2 or more antennas so that locations can be chosen to be compatible with both communication methods.

The PDS command receiver can be either SGLS compatible or USB compatible, but not both. SGLS commands are a three level (ternary) code transmitted at a rate of 2,000 pulses per second. USB commands are an error detecting bi-level code transmitted at 8 KBPS. For the missions considered in this study, STDN station contacts ai most probable. Consequently, USB compatible command receivers will be required. These are also compatible with Shuttle initiated commands.

Another possible command mode is the C-band tracking radar. The Air Force has used tracking radar as a command link on the Minuteman III program. This system uses a five pulse code to send arm and destruct commands to the vehicle. The MPS-36 radars at Vandenberg and Kwajalein are equipped to use this system. The PDS would require a beacon transponder that can recognize the command pulses and send them on to a second package containing command decoder electronics.

15.4 Example Missions - The communications and range safety requirements of the example cases of Section 16 are considered in this section. Cases 1 through 3 consider impact at KMR; Cases 4 and 5 at Poker Flat and Case 6 at Meck Island of KMR. In these missions, the communications equipment is assumed the same. This includes a STDN compatible telemetry transmitter and command receiver and a Shuttle compatible Ku-band radar transponder.

15.4.1 Typical Mission - A typical mission showing the ground station locations is given in Figure 128. The Shuttle is in its fourth orbit after a KSC launch when the payload is separated from the Shuttle. The PDS then follows its own trajectory to KMR. Shaded areas of Figure 128 show Shuttle communication range with ground stations for a 160 NMI orbit. Early in the fourth orbit, there will be a brief contact with Madrid. Although mission rules and planning are not established for this type mission, it would seem desirable to have voice contact between the Shuttle crew and ground mission control at the time of payload separation. Much depends on

TYPICAL KMR MISSION

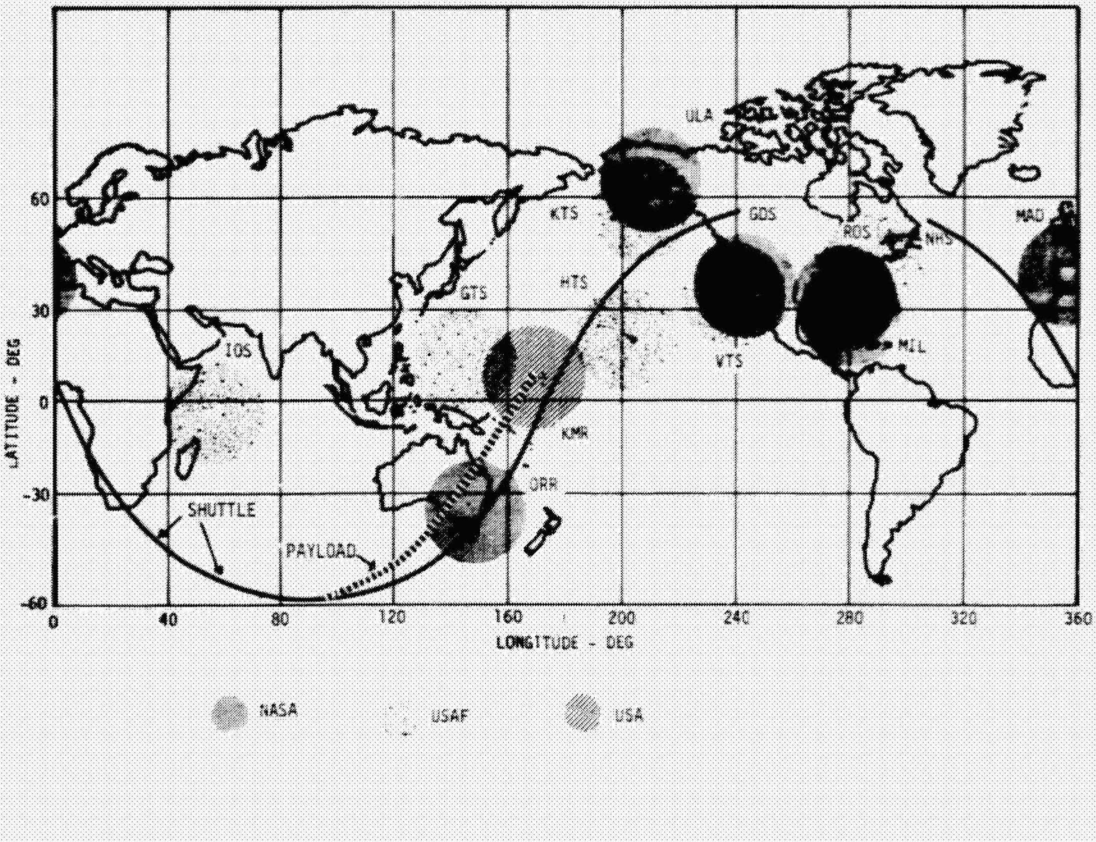


FIGURE 128

the amount of ground involvement in prelaunch checkout. For example, payload data can be received at VAFB on the third orbit, the data analyzed and a launch decision made, relayed to Madrid, then relayed to Shuttle. If constant communication between Shuttle and ground is available through TDRSS, the mission control task is easier to plan. After payload deployment direct grade track of the PDS will be possible from ORR and KMR.

The line of sight communication path to the ground is illustrated in Figure 129. As was stated above, there is no direct communication at the time of separation. Shortly after separation, the STDN station at Orroral, Australia will receive the PDS telemetry signal and a few minutes later, voice and telemetry from Shuttle. ORR will be able to receive a signal throughout the payload deployment phase and near the end of payload deployment, the Kwajalein station will begin to receive signals. The two stations overlap in their coverage of the payload but, because of its lower altitude, there is a gap in coverage of Shuttle communications. This gap is probably not serious from a mission standpoint.

The Australian STDN station normally communicates by voice and teletype to Johnson Space Center at Houston while KMR has a similar link to Vandenberg. Since there are voice links between Houston and Vandenberg, a mission director can obtain real time information on the mission as soon as a signal is acquired at ORR. Real time monitoring of payload separation from Shuttle will require either a TDRSS link or a ship stationed in the frigid waters of the extreme southern region of the Indian Ocean. Thus, for KMR missions, the key ground stations are the STDN station at Orroral for telemetry reception and, if needed, a command uplink and the KMR facility for telmeetry reception and C-band radar tracking. These comments apply in general to the example cases presented in Section 16. Specifics of these missions are described in the following text.

15.4.2 Example Case 1 - Figure 130 identifies the ground coverage for PDS plane change, deorbit maneuver, and spacing burns. The PDS plane change is made at the Shuttle orbit altitude and has no ground coverage. If tracking is required for range safety, there are three options possible: (1) Shuttle can track during plane change if the PDS has a Ku-band transponder, (2) a tracking ship could be positioned off the souther tip of Africa or (3) the PDS data could be relayed to a TDRS and then to an appropriate ground station.

After the completion of the short duration plane change burn, the PDS deorbit maneuver is performed southwest of Australia again with no ground coverage. Shuttle cannot monitor this burn because it has moved away from the PDS several hundred

VEHICLE TO GROUND COMMUNICATION

(NOMINAL SOUTHWESTERN APPROACH TO KMR)

		DISTANCE (NMI)	
		ORR	KMR
1	BOOSTER – SHUTTLE SEPARATION	1500	4340
2	ORR ACQUIRES TELEMETRY SIGNAL	1200	4000
3	ORR ACQUIRES SHUTTLE SIGNAL	1000	3800
4	KMR ACQUIRES TELEMETRY SIGNAL	900	2500
5	ORR LOSES TELEMETRY SIGNAL	2000	1100

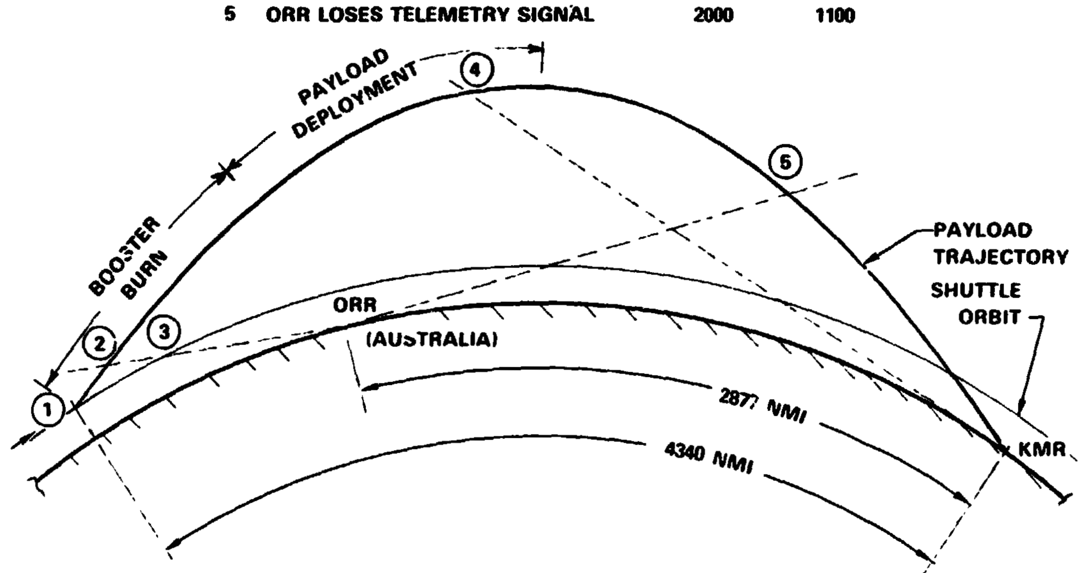


FIGURE 129

CASE 1 MISSION GROUND COVERAGE – FIRST OPPORTUNITY KSC LAUNCH

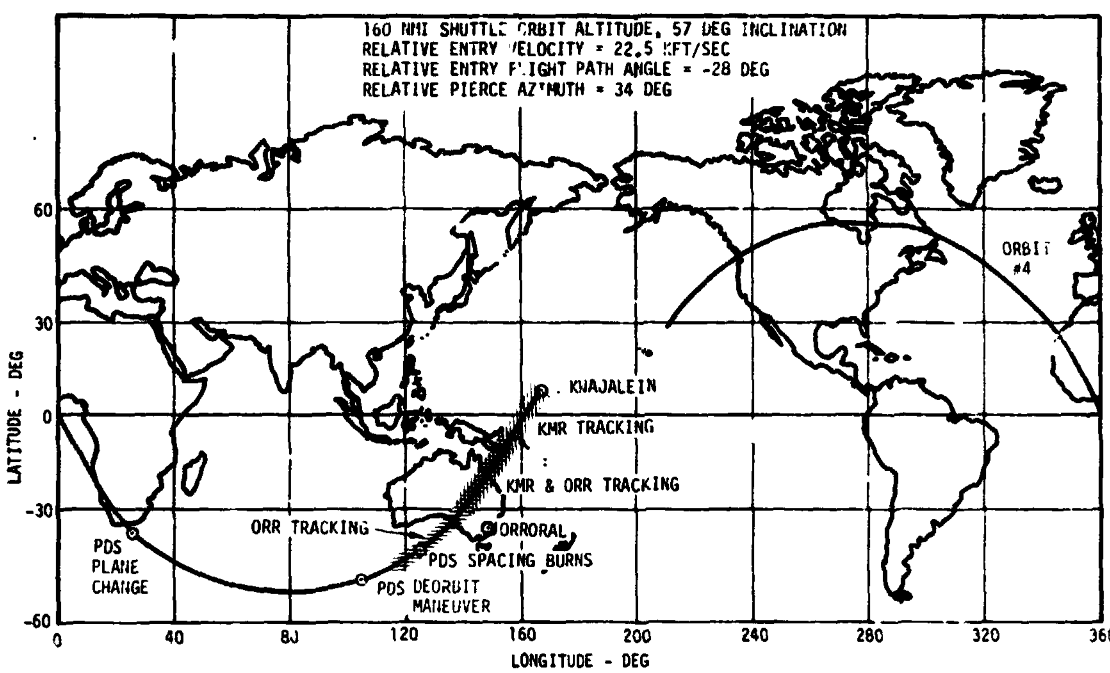


FIGURE 130

nautical miles in crossrange and this burn rapidly separates the two even more so in altitude. Therefore, either a tracking ship or TDRS link is required to monitor this burn.

Shortly after the deorbit burn, the Australian tracking station at Orroral, acquires the PDS, tracks it through the spacing burns, and later hands over to KMR tracking. KMR tracks the payloads to impact. Figure 131 shows the tracking coverage for ORR and KMR on a trajectory plot. KMR acquires the payloads 15 minutes into the deorbit trajectory and prior to apogee. ORR looses the first payload track 12 minutes later. This allows sufficient time for handover.

In summary, a TDRS link is recommended to provide the plane change and deorbit maneuver tracking and telemetry link for this mission. The ORR and KMR tracking stations provide the remainder of the coverage to impact.

15.4.3 Example Cases 2 and 3 - The shallow flight path angle reentry of cases 2 and 3 create special tracking and telemetry problems because of the difficulty in obtaining ground coverage. Figure 133 shows the ground track and coverage for the PDS. The only coverage available from existing ground stations is from ORR and KMR. The KMR coverage is minimal because the payload does not come over the horizon until it is almost at the pierce point. Figure 133 shows the altitudes at which coverage is available from ORR and KMR. Note that none of the PDS burns are covered. Shuttle cannot track these payloads because of the limited capability (300 NMI) of the rendezvous radar. For these cases it is recommended that a TDRS link be used for PDS telemetry and tracking. This insures complete coverage of the entire mission from plane change maneuver to impact.

15.4.4 Example Cases 4 and 5 - A launch from KSC is shown in Figure 134. Orbit 5 provides good coverage at ORR immediately following the PDS deorbit maneuver. After the end of ORR coverage, both KMR and GTS will track the vehicle simultaneously. On the other hand, only the GTS site can communicate with the PDS on orbit 6. The three stations are compatible with respect to S-band telemetry reception but if a command link from GTS or KMR is desired, the system differences noted in paragraph 15.3 must be resolved.

Figure 135 shows three possible ground tracks into Poker Flat from a VAFB launch. Shuttle orbit 12 is preferred for communications since the Madrid STDN station can support a post-separation telemetry check of the PDS. Subsequently, ORR can monitor the deorbit and spacing maneuver and KMR can also track the vehicles. Orbit 13 provides marginal contact at ORR and fair contact at KMR while excellent contact

CASE 1 DEORBIT TRAJECTORY GROUND COVERAGE – FIRST OPPORTUNITY
KSC LAUNCH

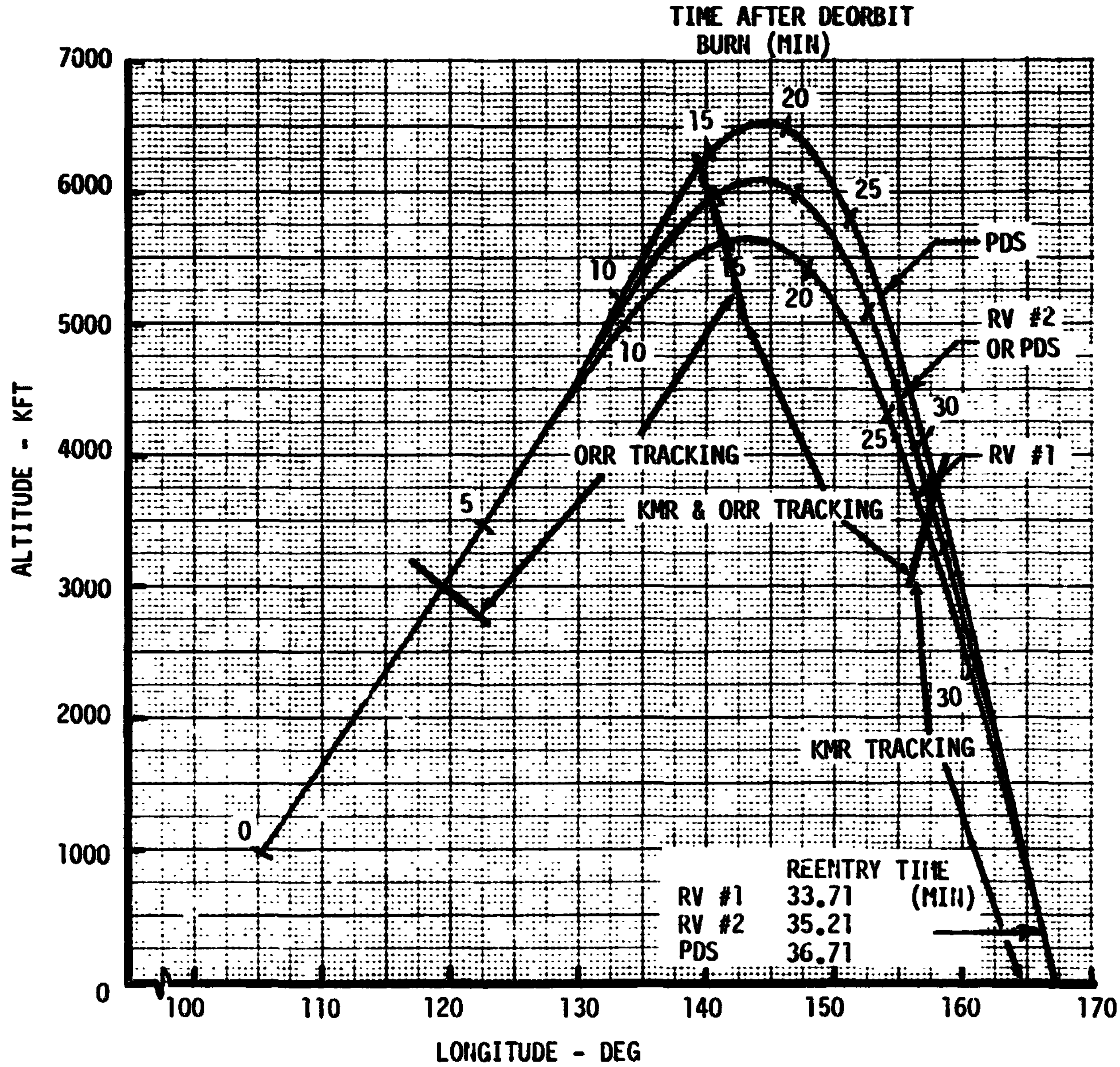


FIGURE 131

CASES 2 & 3 MISSION GROUND COVERAGE
KSC LAUNCH

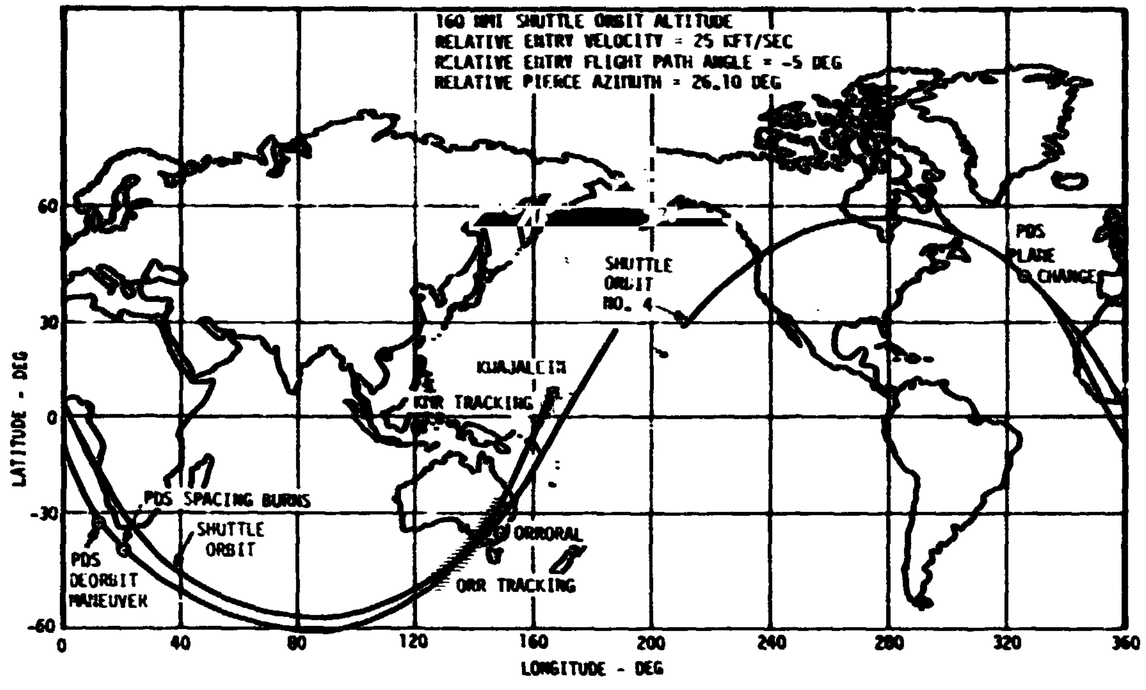


FIGURE 132

CASES 2 & 3 DEORBIT TRAJECTORY GROUND COVERAGE
KSC LAUNCH

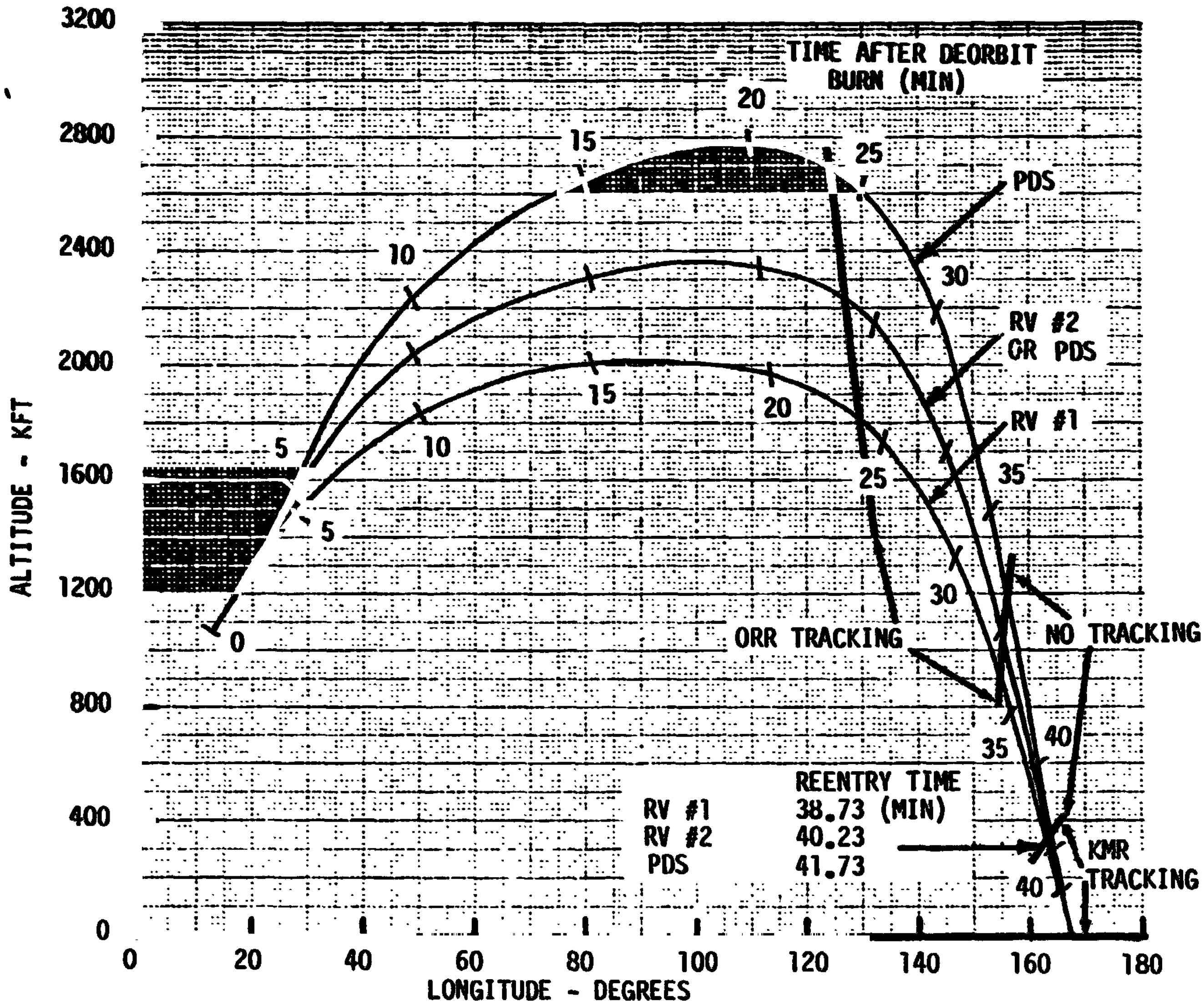


FIGURE 133

CASES 4 & 5 MISSION GROUND COVERAGE
KSC LAUNCH

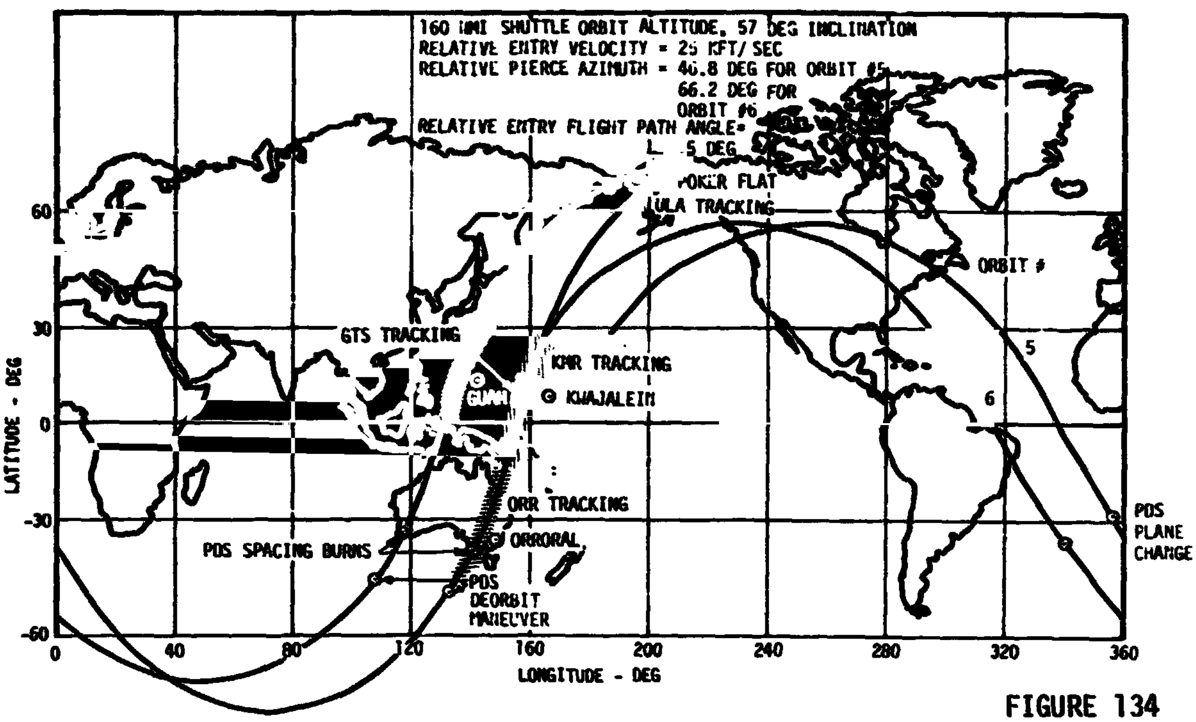


FIGURE 134

CASES 4 & 5 MISSION GROUND COVERAGE
VAFB LAUNCH

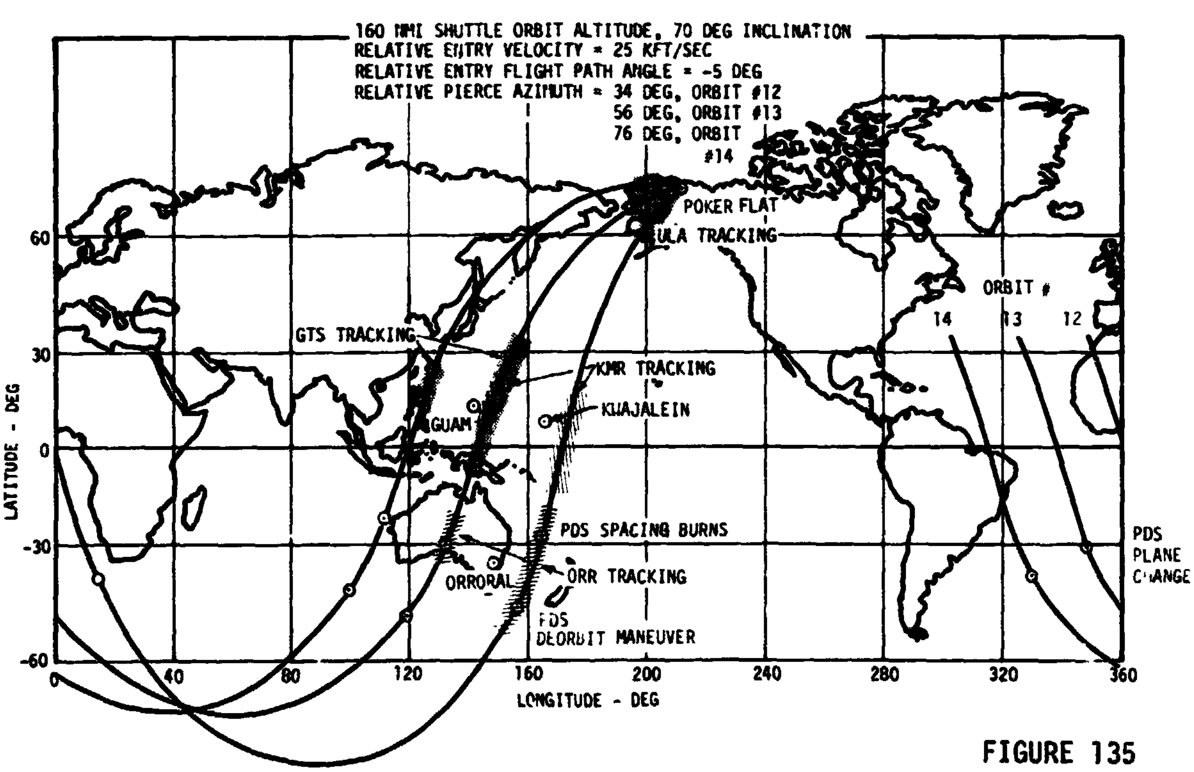


FIGURE 135

with GTS is achieved. Orbit 14 provides only fair contact with GTS and no other contacts.

Figure 136 shows the coverage for the first opportunity deorbit trajectories for a KSC and VAFB launch. The ORR station just misses tracking the deorbit burn for the KSC launch. Good coverage of apogee is obtained for both trajectories. Coverage at Fairbanks and Poker Flat is poor because of the shallow flight path angle reentry. If more complete tracking is desired, a TDRS link is recommended.

Example Case 6 - Figure 137 shows a ground track and coverage from Fairbanks and KMR of the first opportunity deorbit. The deorbit maneuver is performed just before tracking is initiated. By adjusting reentry conditions at Meck or Shuttle orbit altitude complete coverage could be provided. The remainder of the trajectory is covered including an overlap near apogee as shown in Figure 138.

In conclusion, tracking and telemetry from ground stations during PDS burns is not guaranteed for a wide range of missions. Therefore, if coverage is required for range safety purposes, it is recommended that a TDRS link be considered for the PDS. This requires a 50 watt transmitter and steerable parabolic antenna as described in Section 12. The alternatives of tracking ships or tracking from Shuttle do not appear feasible. Several tracking ships would be required to cover both the plane change and deorbit burns. Shuttle tracking ranges of several thousand NMI would be required. Therefore, satellites like the TDRS appear the most practical solution in the 1980 time frame.

CASES 4 & 5 DEORBIT TRAJECTORY GROUND COVERAGE

FIRST OPPORTUNITY

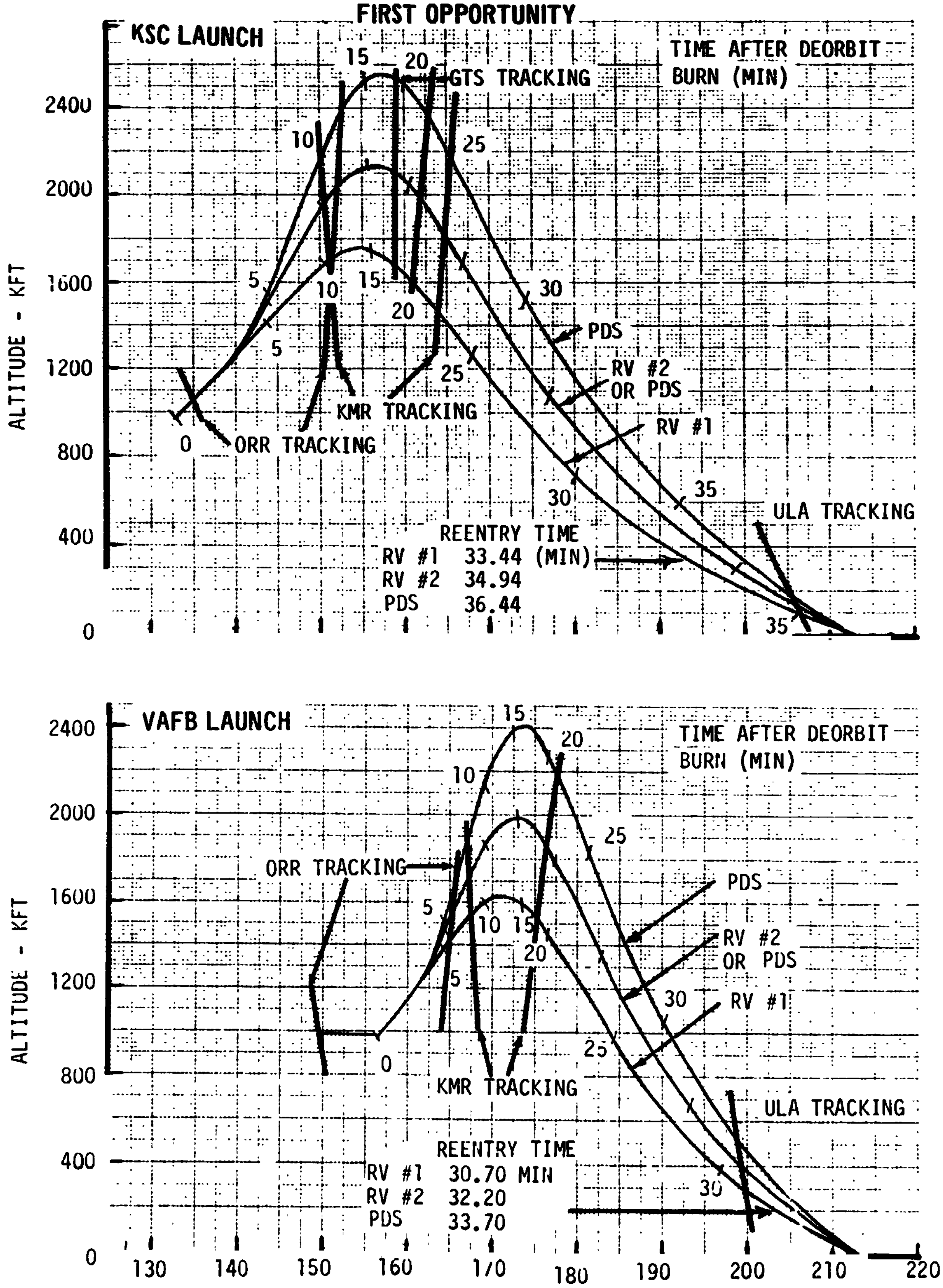


FIGURE 136

CASE 6 MISSION GROUND COVERAGE
VAFB LAUNCH

160 NMI SHUTTLE ORBIT ALTITUDE, -90 DEG INCLINATION
RELATIVE ENTRY VELOCITY = 22.5 KFT/SEC
RELATIVE ENTRY FLIGHT PATH ANGLE = -28 DEG
RELATIVE PIERCE AZIMUTH = 184.275 DEG

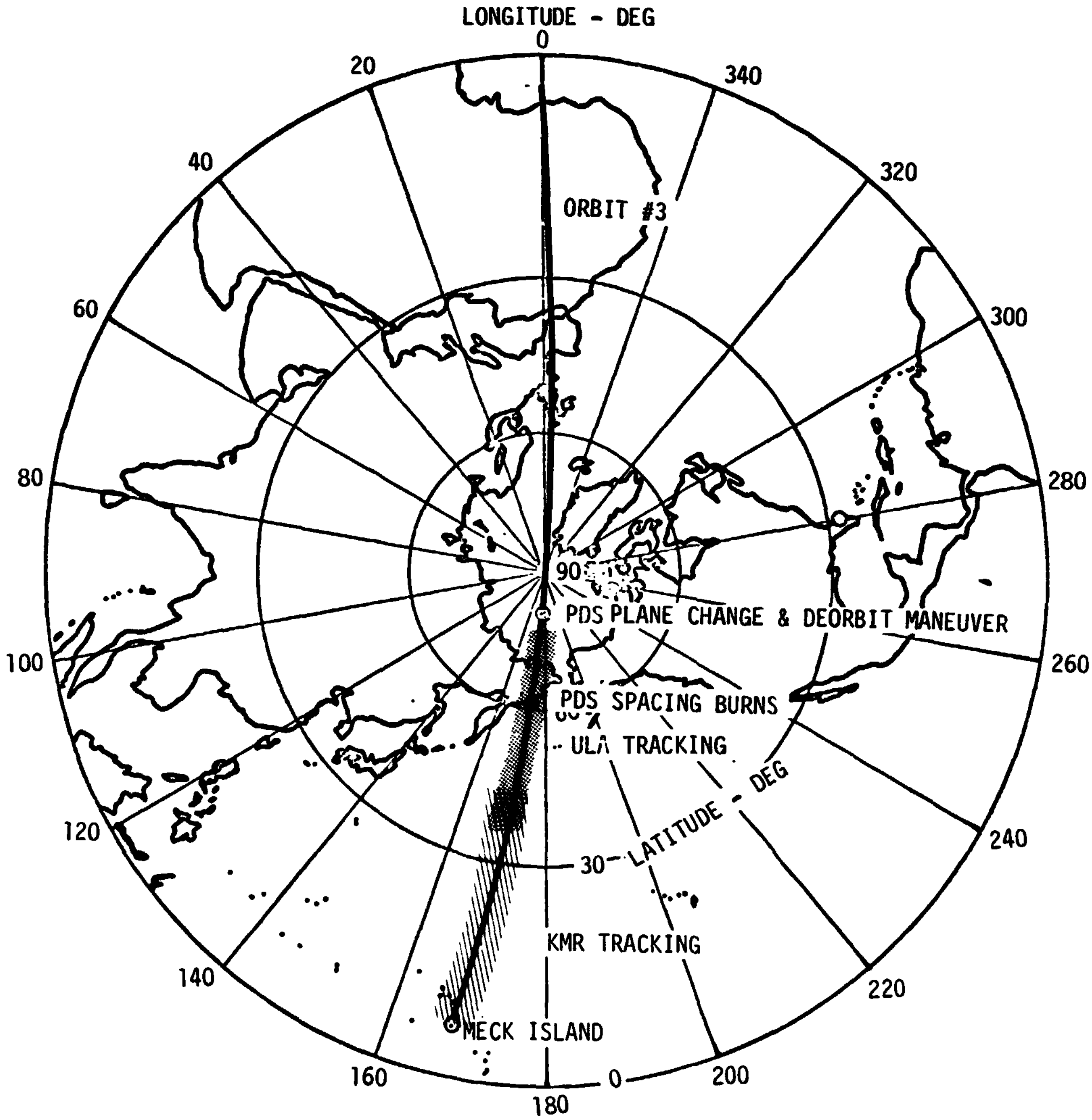


FIGURE 137

CASE 6 DEORBIT TRAJECTORY GROUND COVERAGE
VAFB LAUNCH

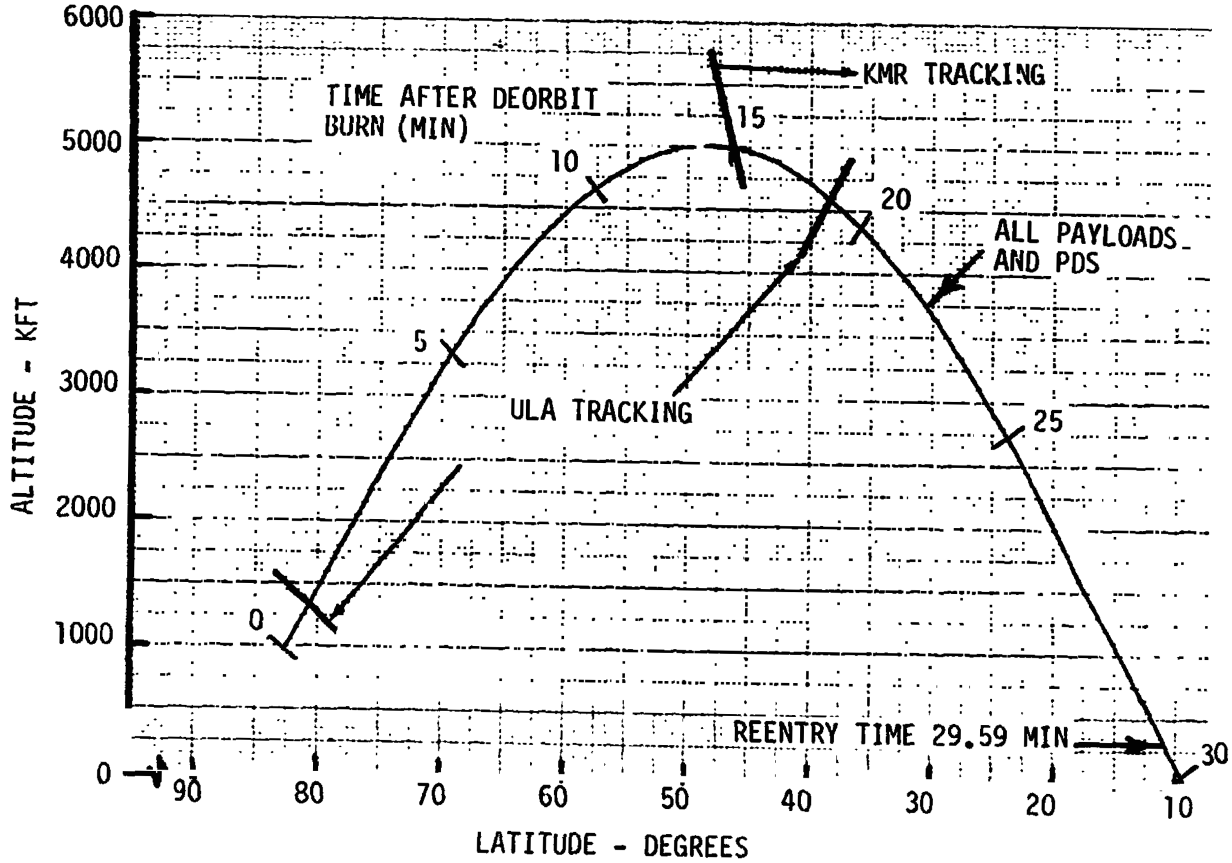


FIGURE 138

16. ANALYSIS OF SPECIFIC MISSIONS

This section presents the detailed analysis of six example cases requested in Reference 2. Each case was analyzed in a similar manner which involved selecting the Shuttle orbit for PDS deployment; determining the PDS plane change, deorbit, and spacing burn requirements; defining the number of deorbit opportunities available; establishing the propellant requirements of the Transtage booster. The approach used will be discussed in detail for the first case. For the other cases, the approach will only be discussed where it differs from the first. The major conclusion from the analyses of the six cases was that all could be done with ease from the nominal Shuttle orbit altitude with a Transtage class or smaller booster.

16.1 <u>Example Case 1 Results</u> - This case requires deorbiting two payloads of 600-lb and 30-lb such that they reenter at KMR with a relative velocity of 22.5 kfps and flight path angle of -28 deg. This represents a typical VAFB to KMR ground launch pierce point condition. However, in this case the payloads will approach the KREMS radars through the southwest corridor described in Section 5.

The KREMS southwest corridor at Kwajalein requires an approach azimuth (pierce point azimuth) of between 34 and -5 degrees. For a maximum northward Shuttle launch from KSC into a 160 NMI orbit, a PDS plane change is required to meet this constraint. More eastward launches from KSC require even more of a plane change.

Because there is a range of allowable pierce point azimuths at KMR, it is possible to find a pierce point azimuth which minimizes the plane change ΔV requirement. This is accomplished by first determining the location of the deorbit burn as a function of pierce point azimuth at KMR. Figure 139 shows typical ground tracks and deorbit locations for three pierce point azimuths. (Note the change in deorbit burn location.) A PDS plane change maneuver must be made at some point of the Shuttle orbit to place the PDS at this deorbit location. The minimum plane change ΔV occurs 90 degrees from this deorbit point. The plane change does not in general place the PDS at the deorbit burn location with the proper azimuth. However, depending upon the range of allowable pierce point azimuths, there may be one which has a deorbit point and deorbit azimuth consistent with a single plane change maneuver.

Figure 140 is a plot of the inertial azimuth at the deorbit burn location as a function of the pierce point relative azimuth. The line labeled "deorbit burn azimuth" is the required azimuth at the deorbit burn location to achieve the required pierce point conditions and impact point location. (The impact point is given in

CASE 1 SENSITIVITY OF DEORBIT GROUND TRACK TO PIERCE POINT AZIMUTH AT KWAJALEIN

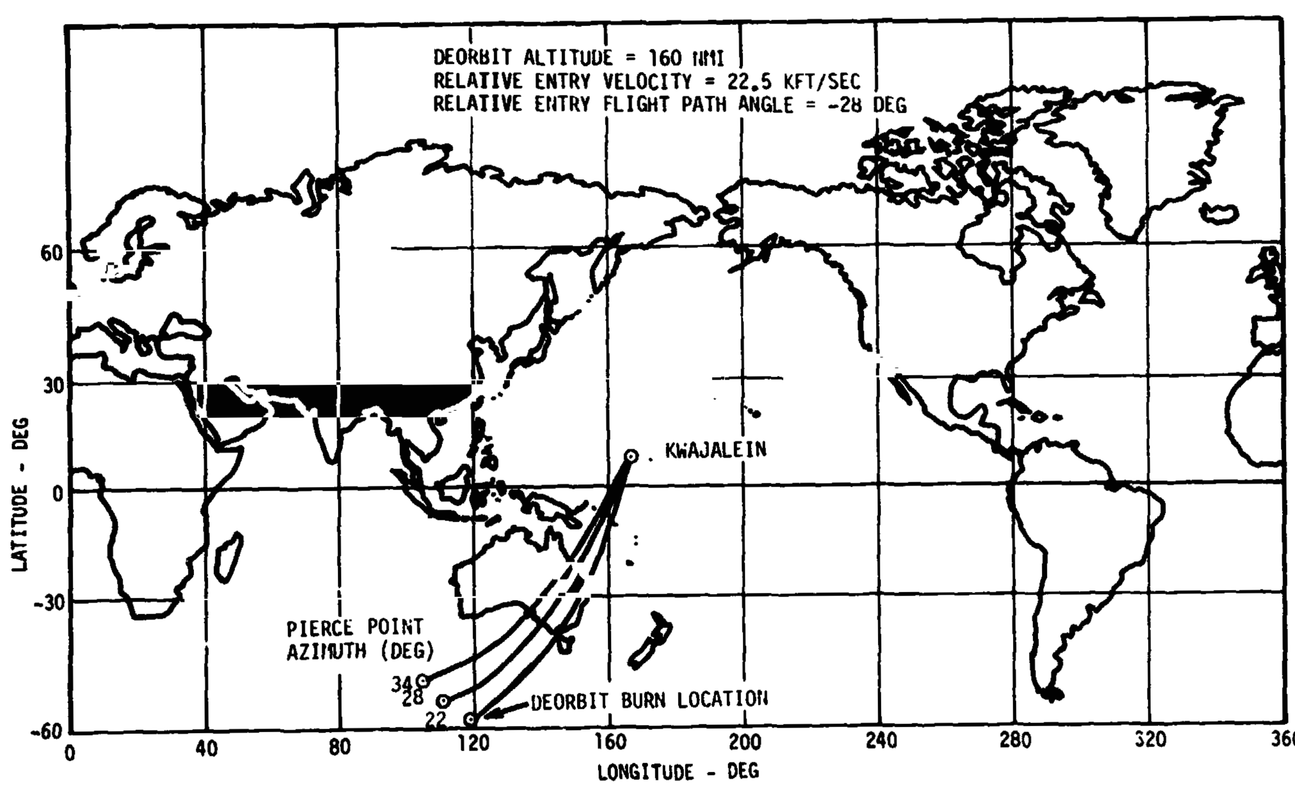


FIGURE 139

CASE 1 PIERCE POINT AZIMUTH DEFINITION

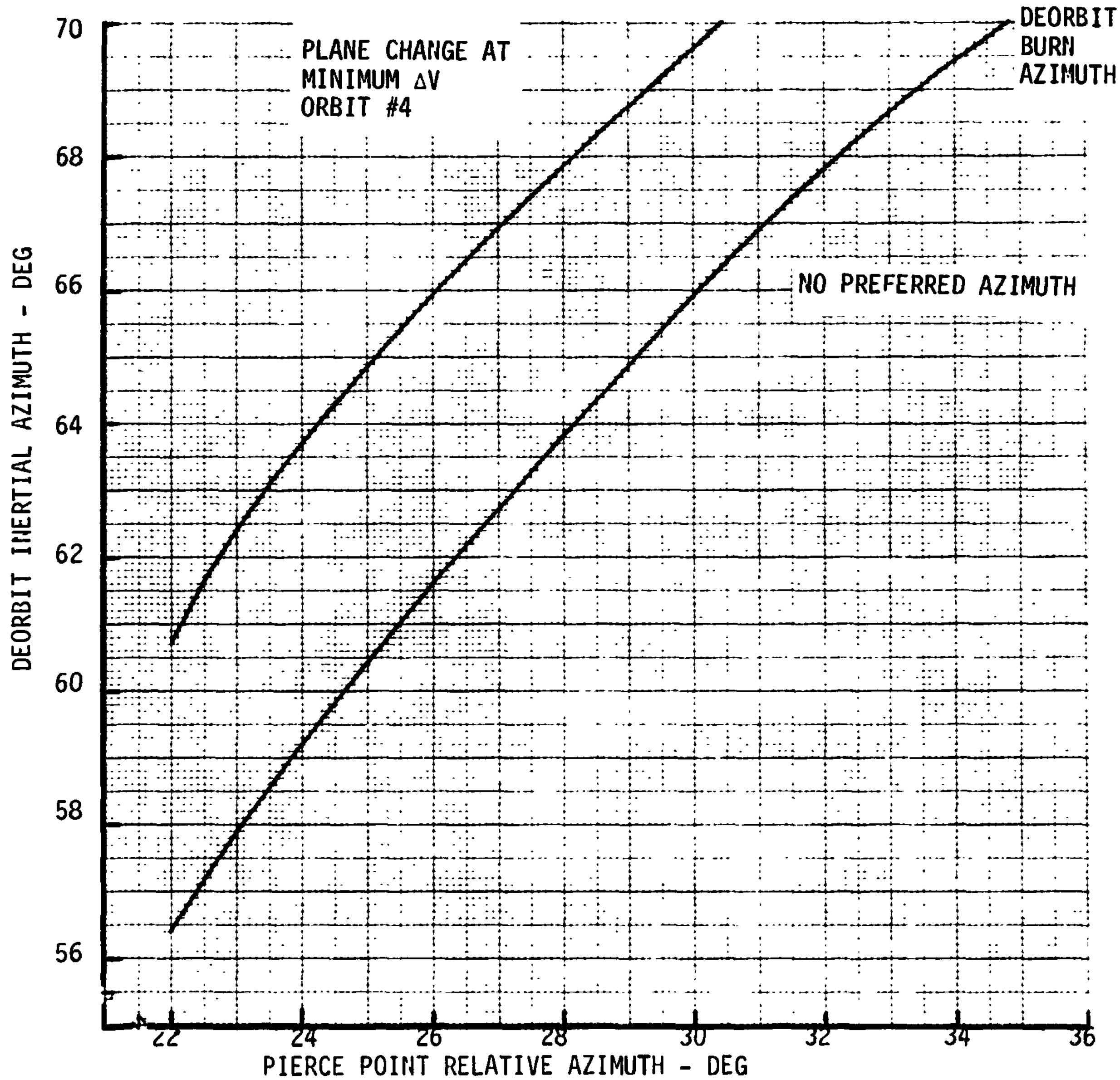


FIGURE 140

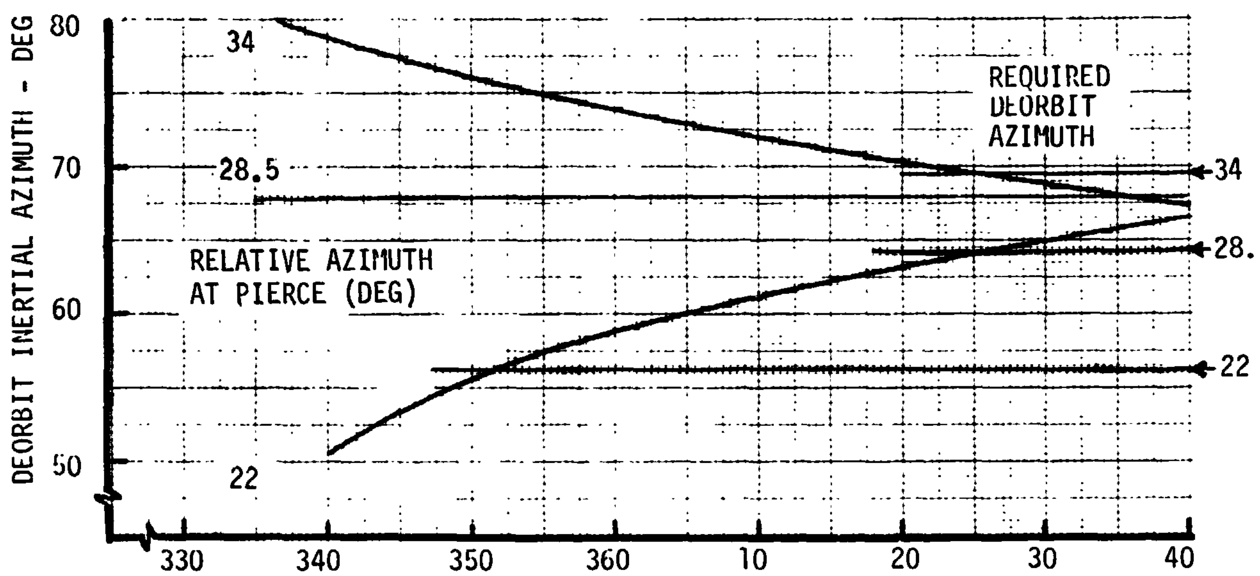
Section 5.) The line labeled "plane change at minimum ΔV orbit #4" is the resultant inertial azimuth at the deorbit burn location if a PDS plane change is performed during Shuttle orbit 4, 90 deg away from the deorbit location, and results in an orbit which passes through the deorbit burn location. Because the two curves do not intercept, no plane change at the minimum plane change ΔV point will place the PDS at the deorbit burn location with the required deorbit burn azimuth. Therefore, the plane change must be made at a nonoptimum point or a second plane change is required.

Figure 141 presents the data to select a plane change location at a nonoptimum ΔV location. In the top plot, the inertial azimuth at the deorbit location is plotted as a function of the longitude on Shuttle orbit 4 where the nonoptimum plane change is made. The curved lines starting on the left represent the variation of azimuth achieved at the deorbit location as a function of the plane change longitude. The straight lines starting at the right merely identify the required deorbit azimuth. The intersection of the two lines for the same relative azimuth identifies where the nonoptimum plane change must occur. For instance, a plane change at about 26 deg west longitude will provide the proper azimuth at the deorbit point for a 34 deg relative azimuth reentry at KMR. The resultant plane change ΔV is read at the corresponding longitude on the bottom plot. It is 2570 fps and is only slightly greater than the optimum of 2250 fps. Not much of a ΔV penalty results. Therefore, this plane change location and a pierce point azimuth of 34 deg was selected for the first opportunity deorbit for case 1.

Another interesting possibility arises when the top plot of Figure 141 is reconsidered. Note that the deorbit inertial azimuth curve for a 28.5 deg pierce point azimuth is almost independent of longitude and only about 3.5 deg different than the required value. In addition, referring to the bottom figure almost no plane change is required to hit the deorbit point. These two facts indicate the possibility of a combined plane change-deorbit maneuver at the deorbit point. This in fact is possible and provides a 28 deg relative pierce point azimuth at KMR. The plane change required at the deorbit burn point is only one degree and is easily combined with the deorbit burn.

Figures 142 and 143 provide the ground tracks for the first opportunity and single burn deorbit opportunity, respectively. In either case, both maneuvers occur during Shuttle orbit 4 and the PDS passes over Australia during the deorbit trajectory. At the PDS plane change point the Shuttle continues in its original orbit and the PDS orbit inclination is changed.

CASE 1 PIERCE POINT AZIMUTH DEFINITION (ORBIT #4)



CASE 1 PLANE CHANGE IMPULSIVE ΔV VARIATION WITH MANEUVER LOCATION (ORBIT #4)

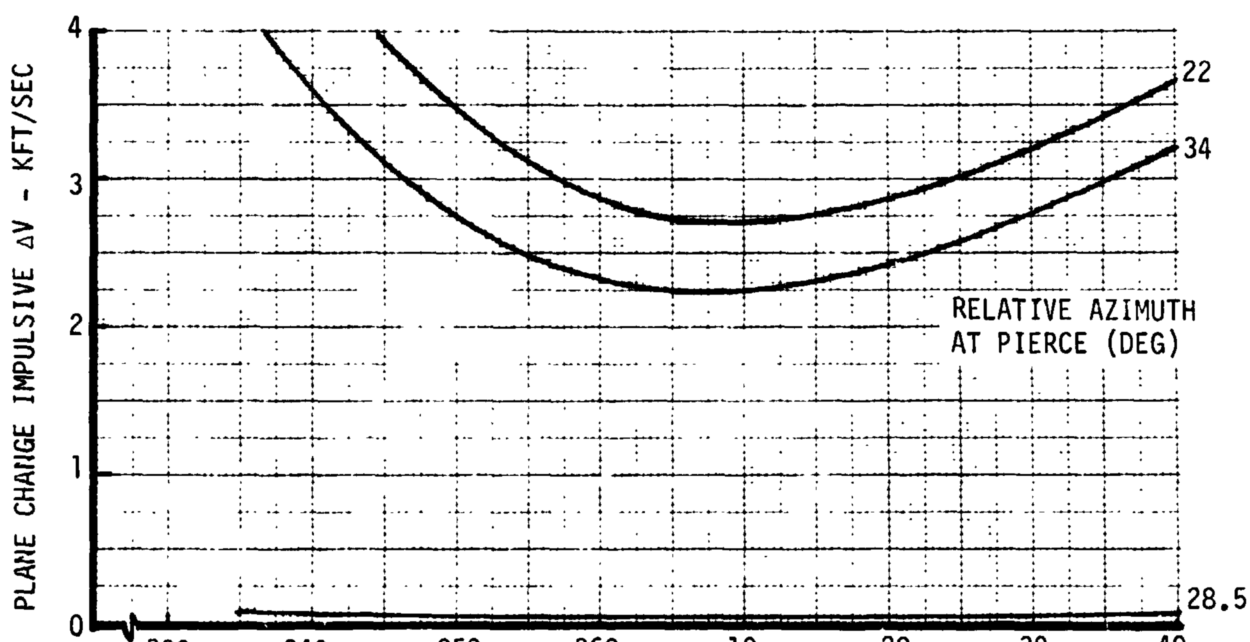


FIGURE 141

CASE 1 MISSION GROUND TRACK & MANEUVER LOCATIONS – FIRST OPPORTUNITY

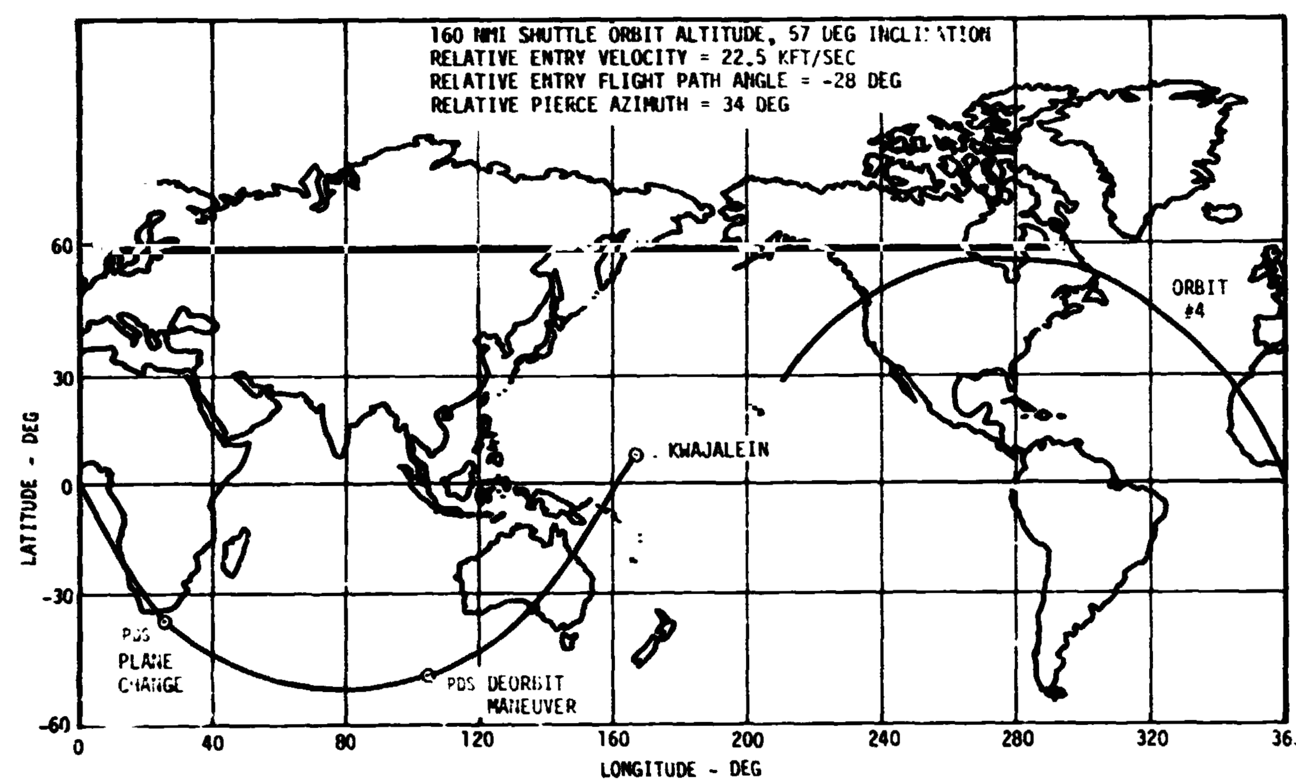


FIGURE 142

CASE 1 MISSION GROUND TRACK & MANEUVER LOCATIONS – SINGLE BURN OPPORTUNITY

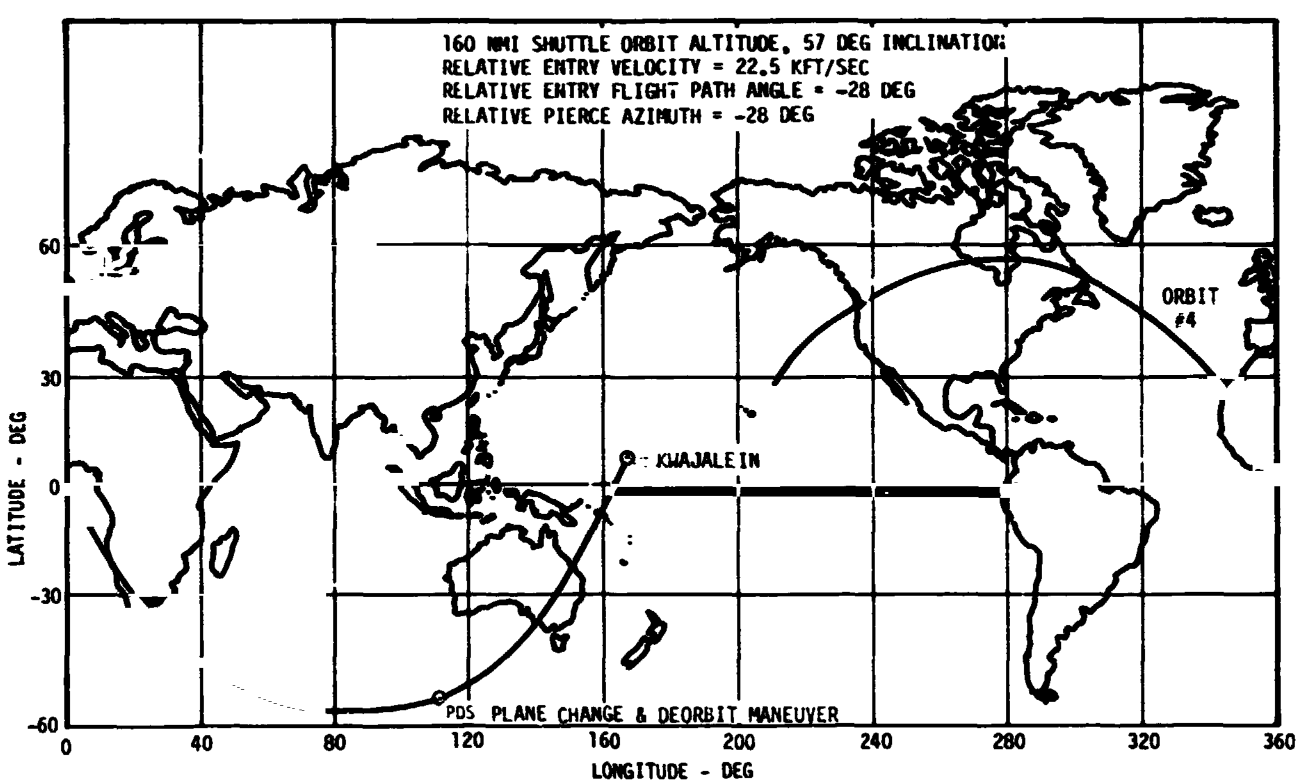


FIGURE 143

The deorbit trajectories for the two payloads and the booster are shown in Figure 144. The 90 sec spacing at pierce is achieved by spacing burns initiated 7.5 and 8.5 min after deorbit burn initiation for the second RV and booster, respectively. The trajectory from pierce to impact is assumed to be a vacuum trajectory, i.e., no atmospheric effects. Note that the first RV reenters 33.7 min after the deorbit burn initiation.

Detailed information concerning the deorbit trajectory maneuver locations and characteristics are provided in Figures 145 and 146. The ground range to impact is measured along a great circle to the impact point and not along the trajectory ground track. The latitude and longitude are given in geodetic coordinates and reflect a spherical earth. Altitude is the geometric altitude with impact assumed to be sea level. The azimuth, velocity and flight path angle given are relative values. Time to impact is measured from first payload impact. Atmospheric effects are neglected from reentry to impact. Inclusion of the atmospheric effects would not change the results significantly. The impulsive ΔV is that required to change plane, deorbit, or space the payloads. For this case, the deorbit ΔV is large due to the large flight path angle change required during the deorbit maneuver. The payloads are targeted to impact at the same point. Consequently, the ground range, latitude, longitude and altitude are equal at impact. In addition, payloads pass through 300 kft spaced 90 seconds apart.

The performance requirements of Transtage are provided on the computer output of Figures 147 and 148. This was obtained by exercising option 4 of the sizing program described in Section 9. The results shown assumed propellant is offloaded to minimize launch weight. In Figure 147, the launch mass of the bus include all inert mass and propellant (see Section 8); the spin system and attachments include the tubular strut support as well as the spin tables (see Section 11); the total without payload is the sum of the previous columns; and the total with payload adds the payload weight from the payload summary. This last number, 22849.6 lbm, is the total PDS mass in the Shuttle payload bay. The propellant mass only applies to the bus. In this case the bus function is served by Transtage. The propellant mass given in the last column, i.e., 18221.1 lbm, represents the total required for this mission. The burnout mass for the booster/bus is the inert mass; for the spin system it is the launch mass (it has no propellant); the total without payload is the sum of bus and spin system; the total with payload adds the RV mass to give the total dry weight of the PDS which is 4628.5 lbm. In the next

CASE 1 DEORBIT TRAJECTORIES – FIRST OPPORTUNITY
KSC LAUNCH

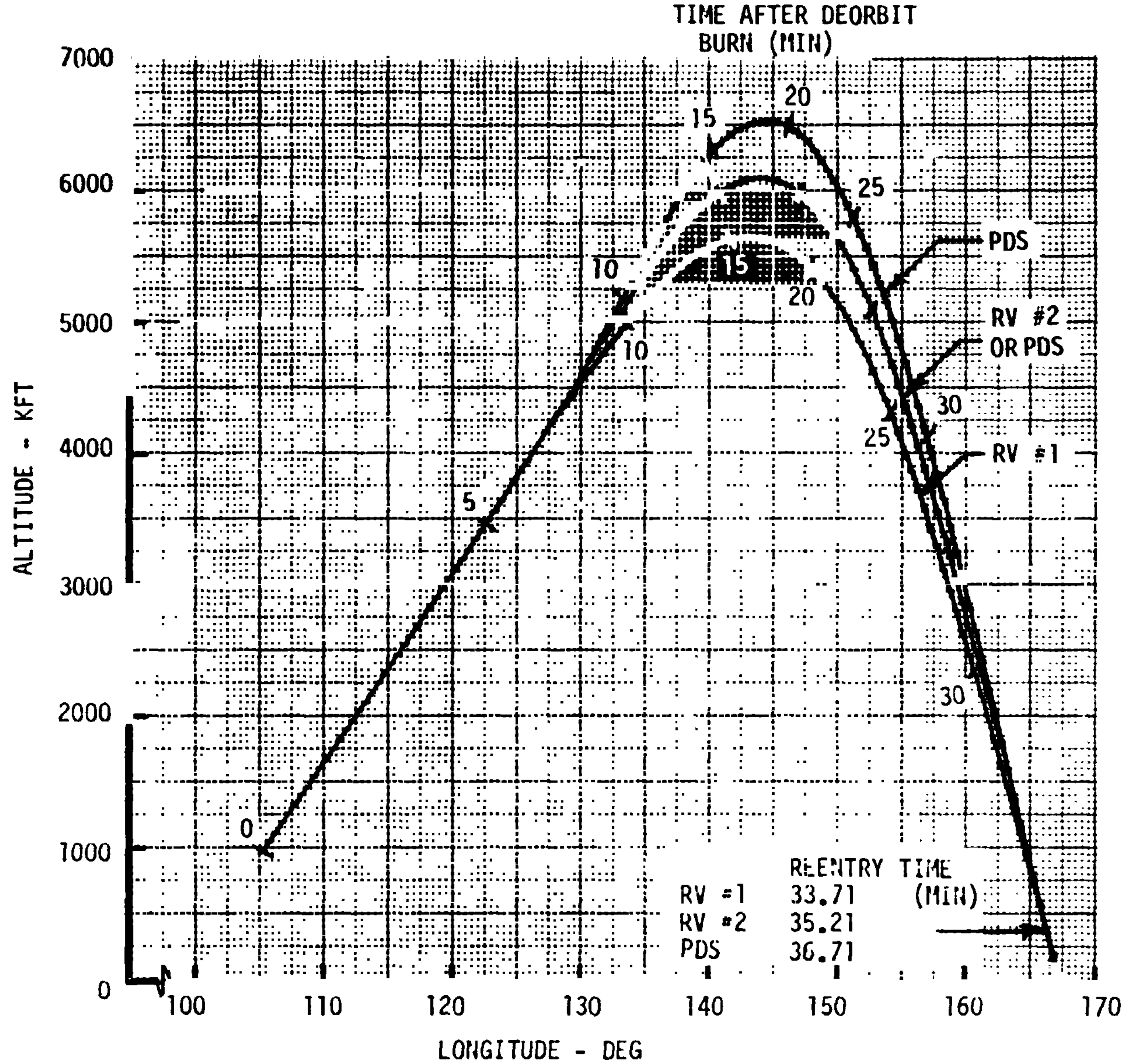


FIGURE 144

CASE 1 DEORBIT SUMMARY – FIRST OPPORTUNITY
KSC LAUNCH

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER		8140.99	-36.13	26.26	972.2	133.68	48.97	24.41	0.00	2570
DEORBIT MANEUVER	1	4787.30	-49.87	105.31	972.2	68.33	34.18	21.55	27.19	11150
	2	3648.67	-39.79	128.21	4311.5	48.61	26.68	17.06	20.90	1016
	3	3520.59	-38.36	130.25	4657.9	46.81	25.68	16.57	22.51	986
ENTRY	1	91.39	7.64	166.54	300.0	34.00	0.47	22.50	-28.00	--
	2	87.08	7.88	166.36	300.0	33.93	-1.03	22.41	-30.31	--
	3	84.06	8.16	166.20	300.0	33.88	-2.53	22.38	-32.49	--
IMPACT (NO ATMOSPHERE)	1	0.00	8.90	167.40	0.0	34.26	0.00	22.92	-28.31	--
	2	0.00	8.90	167.40	0.0	34.19	-1.50	22.83	-30.60	--
	3	0.00	8.90	167.40	0.0	34.12	-3.00	22.80	-32.77	--

*IMPACT OF FIRST PAYLOAD

TOTAL ΔV = 15722 FPS

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FIGURE 145

CASE 1 DEORBIT SUMMARY – SINGLE BURN OPPORTUNITY KSC LAUNCH

PHASE	GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER	4734.26	-54.47	111.37	972.2	62.65	33.54	21.54	27.14	--
DEORBIT MANEUVER 1	4734.26	-54.47	111.37	972.2	62.65	33.54	21.54	27.14	11240+
2	3592.88	-42.62	134.18	4286.6	41.74	26.04	17.09	20.71	1075
3	3464.28	-41.00	136.06	4629.3	39.94	25.04	16.57	22.38	1023
ENTRY 1	91.51	7.56	166.67	300.0	28.00	0.47	22.50	-28.00	--
2	86.51	7.81	166.45	300.0	27.92	-1.03	22.41	-30.45	--
3	84.76	8.08	166.24	300.0	27.84	-2.53	22.34	-32.71	--
IMPACT(NO ATMOSPHERE) 1	0.00	8.90	167.40	0.0	28.25	0.0	22.92	-28.34	--
2	0.00	8.90	167.40	0.0	28.16	-1.50	22.83	-30.77	--
3	0.00	8.90	167.40	0.0	28.08	-3.00	22.77	-33.00	--

*IMPACT OF FIRST PAYLOAD

TOTAL ΔV = 13338 FPS

+COMBINED PLANE CHANGE AND DEORBIT BURN

FIGURE 146

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 1 FIRST OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 630.0 LBM

NUMBER	1	2
M-LBM	600.0	30.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	21972.1	247.5	22219.6	22849.6
PROPELLANT MASS (LBM)	.0	18221.1	.0	18221.1	18221.1
BURNOUT MASS (LBM)	.0	3751.0	247.5	3998.5	4628.5
LENGTH (FT)	.00	14.83	7.00	21.83	22.0[illegible]
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	5490.0			5490.0
BURN TIME (MIN)	.00	5.82			5.82
MASS FRACTION	.000	.829			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 4810.93 LBM
PERCENT OFFLOAD = 20.89 PERCENT
EXCESS DELTA-V = 1852.27 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	2570.0	1.70	1603.3	22849.6	17528.4	5321.2	5321.2	-.0
2	11150.0	3.82	3609.4	17528.4	5549.1	11979.3	17300.5	600.0
3	1016.0	.16	148.4	4949.1	4456.6	492.4	17793.0	30.0
4	986.0	.14	129.0	4426.6	3998.5	428.1	18221.1	3998.5

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FIGURE 147

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 1 SINGLE BURN OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 630.0 LBM

NUMBER	1	2
M-LBM	600.0	30.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	16971.0	247.5	17218.5	17848.5
PROPELLANT MASS (LBM)	.0	13220.0	.0	13220.0	13220.0
BURNOUT MASS (LBM)	.0	3751.0	247.5	3998.5	4628.5
LENGTH (FT)	.00	14.83	7.00	21.83	22.00
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	3983.2			3983.2
BURN TIME (MIN)	.00	4.22			4.22
MASS FRACTION	.000	.779			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 9812.02 LBM
PERCENT OFFLOAD = 42.60 PERCENT
EXCESS DELTA-V = 4246.82 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	11240.0	3.91	3691.0	17848.5	5598.2	12250.3	12250.3	600.0
2	1075.0	.17	158.1	4998.2	4473.5	524.6	12775.0	30.0
3	1023.0	.14	134.1	4443.5	3998.5	445.0	13220.0	3998.5

FIGURE 148

line, the length of the bus and spin system are combined to give the total without payload. The last column which gives 22 ft includes the payload. For tubular spin systems the total length with and without payload is not much different. The payloads protrude only slightly from the tubes. The diameter, total impulse, burn time and thrust are self-explanatory. The mass fraction is the ratio of propellant to bus launch mass.

Offload requirements define how much propellant was offloaded to perform this mission. (Fully loaded Transtage has 23032 lbm of propellant.) The excess ΔV is the ΔV capability which could be achieved if the propellant were not offloaded. The first opportunity case only requires 20% offload indicating the Transtage is of reasonable size for this mission. However, the single burn opportunity described in Figure 148 requires 42% offload and indicates Transtage is oversized.

The burn summary provides the burn number; impulsive ΔV required, DV; the burn time, BURN-T; total impulse for that burn, I-TOT; the initial PDS mass, INITIAL; the mass at the completion of the burn, B/O; the propellant used, PROP; the total propellant used through the present burn, TOTAL; and the mass deployed before the next burn, M-DEPLOY. A zero mass deployed indicates only a plane change maneuver was performed. Note that the burn time is quite long for burn 2, the deorbit burn in Figure 147, but very short for the low total impulse spacing burns 3 and 4.

In conclusion, example case 1 can be achieved with a Shuttle launch from KSC. Only one opportunity at KMR is provided due to the large ΔV required at deorbit. A combined plane change and deorbit burn is feasible indicating a possible application of a solid rocket motor. By performing a combined maneuver, the booster propellant requirements are significantly reduced.

16.2 Example Cases 2 and 3 Results - These cases require deorbiting one or two 1000-lb payloads at KMR with a relative velocity of 25 kfps and a flight path angle of -5 deg. This represents a class of reentry conditions which are easily achieved from Shuttle orbit. The approach used to define plane change and deorbit requirements is analogous to that described for case 1 in Section 16.1.

Figure 149 shows the sensitivity of deorbit burn location to pierce point relative azimuth. Note that the deorbit locations are far removed from KMR because of the shallow angle entry requirement.

Figure 150 contains the working data which defines the pierce azimuth for a single plane change maneuver from orbit 4. The solid line represents the variation of the inertial azimuth at the deorbit point as a function of the relative pierce point azimuth. The dashed line represents the variation of the plane change trajec-

CASES 2 & 3 SENSITIVITY OF DEORBIT GROUND TRACK TO PIERCE POINT AZIMUTH AT KWAJALEIN

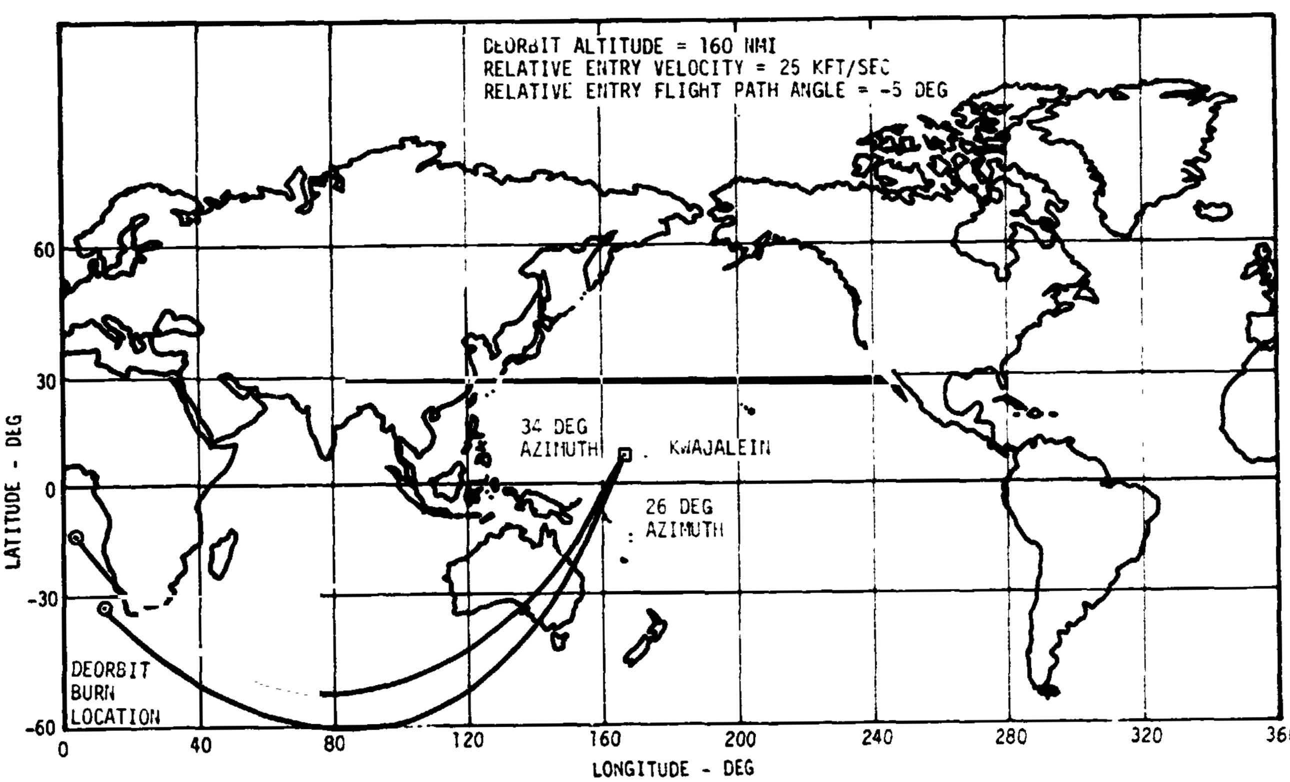

FIGURE 149

CASES 2 & 3 PIERCE POINT AZIMUTH DEFINITION

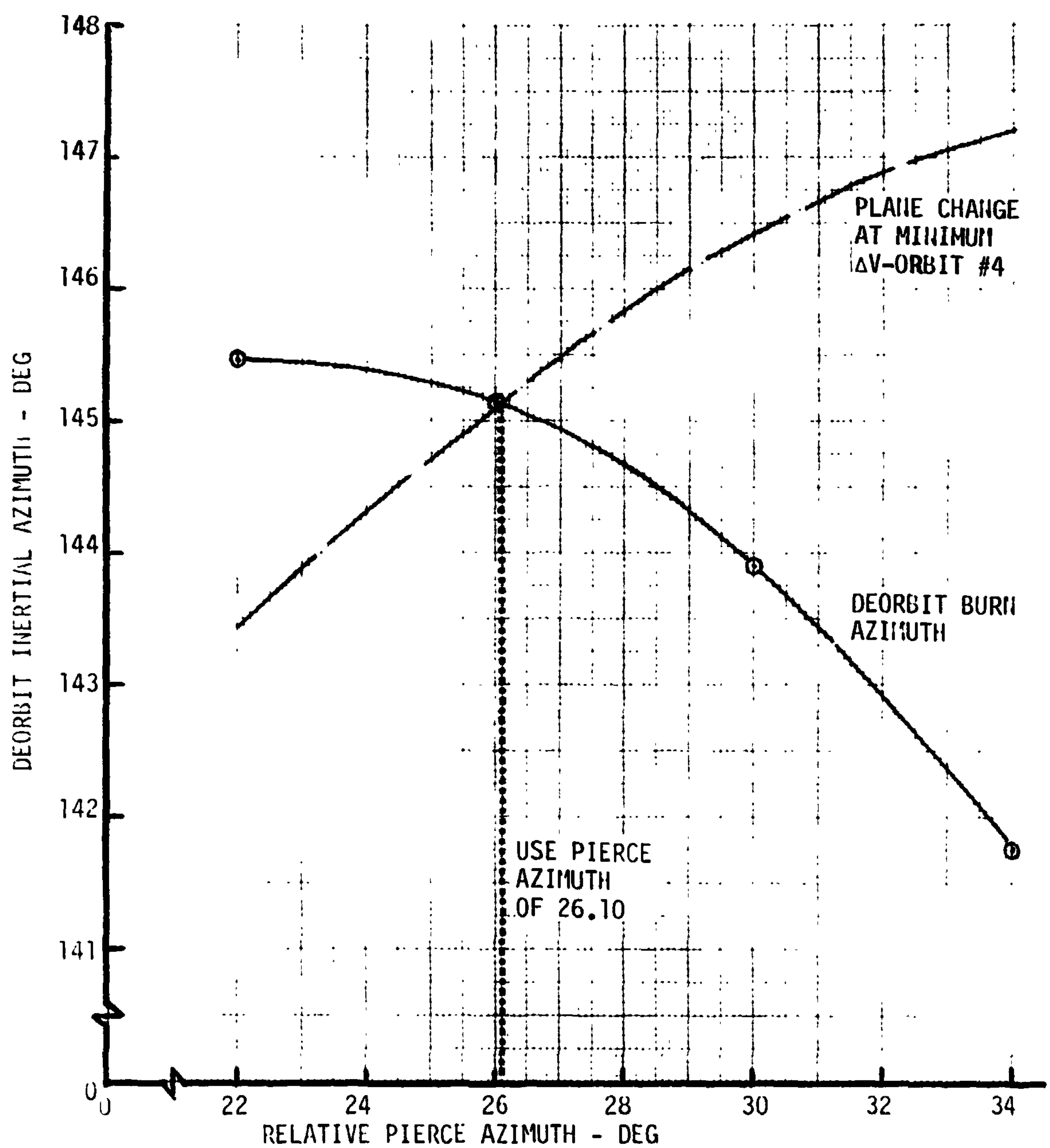

FIGURE 150

tory inertial azimuth at the deorbit point as a function of the relative pierce point azimuth. The crossover point is the point at which the plane change and deorbit trajectory azimuth are equal and no second plane change burn is required. This corresponds to a relative pierce point azimuth of 26.1 deg. Figure 151 gives the ground tracks for the plane change and deorbit trajectories. The plane change maneuver is made over the mid-Atlantic and the deorbit burn off the east coast of South Africa.

The above analysis addressed only the first payload deorbit trajectory. In case 2 the PDS and in case 3 another payload and the PDS must be deorbited. To define these payload state vectors and ΔV requirements, they were assumed targeted to the same impact point as the first payload but 90 sec apart in time at the pierce point. The trajectories for cases 2 and 3 are shown in Figure 152.

Case 2 includes only the first RV and PDS trajectory. The middle trajectory labeled RV #2 or PDS corresponds to the PDS trajectory for case 2. The first RV reenters 38.7 minutes after deorbit burn initiation with the other payloads spaced 90 seconds behind. The apogee altitudes are low because of the shallow angle reentry. Figure 153 provides a detailed summary of these trajectories. For case 2, item 1 and 2 refer to the RV and PDS, respectively. For case 3, item 2 is the second RV and item 3 is the PDS. Note the low ΔV requirements for these missions. This is further emphasized by the design summary of Figures 154 and 155. Here a propellant offloading of 84 and 78% for cases 2 and 3, respectively, could be achieved. However, nearly 10,000 fps plane change ΔV is required to have a second consecutive deorbit opportunity at KMR. Therefore, it may be advantageous to use a fully loaded Transtage with an excess ΔV capability of 11371 or 9341 ft/sec to allow for a second consecutive deorbit opportunity. Obviously, the Transtage is oversized for case 2 and 3 single opportunity reentry at KMR. The Minibus concept of Section 11 may be more practical for these cases.

To further summarize case 2 and 3 missions, the mission sequence shown in Figure 156 is provided. From the time of Shuttle launch at point 1 to achievement of circular orbit is almost one full orbit. The PDS deployment occurs at the end of the third orbit over the Pacific. By the time the PDS passes over Canada it is ready for the plane change maneuver at point 4. As it approaches the coast of Africa the deorbit burn is completed, the RV deployed, and the spacing burn performed. Approximately 40 minutes later the RV reenters at KMR. Six hours after Shuttle launch the payload impacts at KMR. These time lines may vary depending upon specific launch conditions and deorbit requirements, but the mission sequence seems to allow sufficient time for all prelaunch checkout and commit decision to be made.

CASES 2 & 3 MISSION GROUND TRACK AND MANEUVER LOCATIONS – FIRST OPPORTUNITY KSC LAUNCH

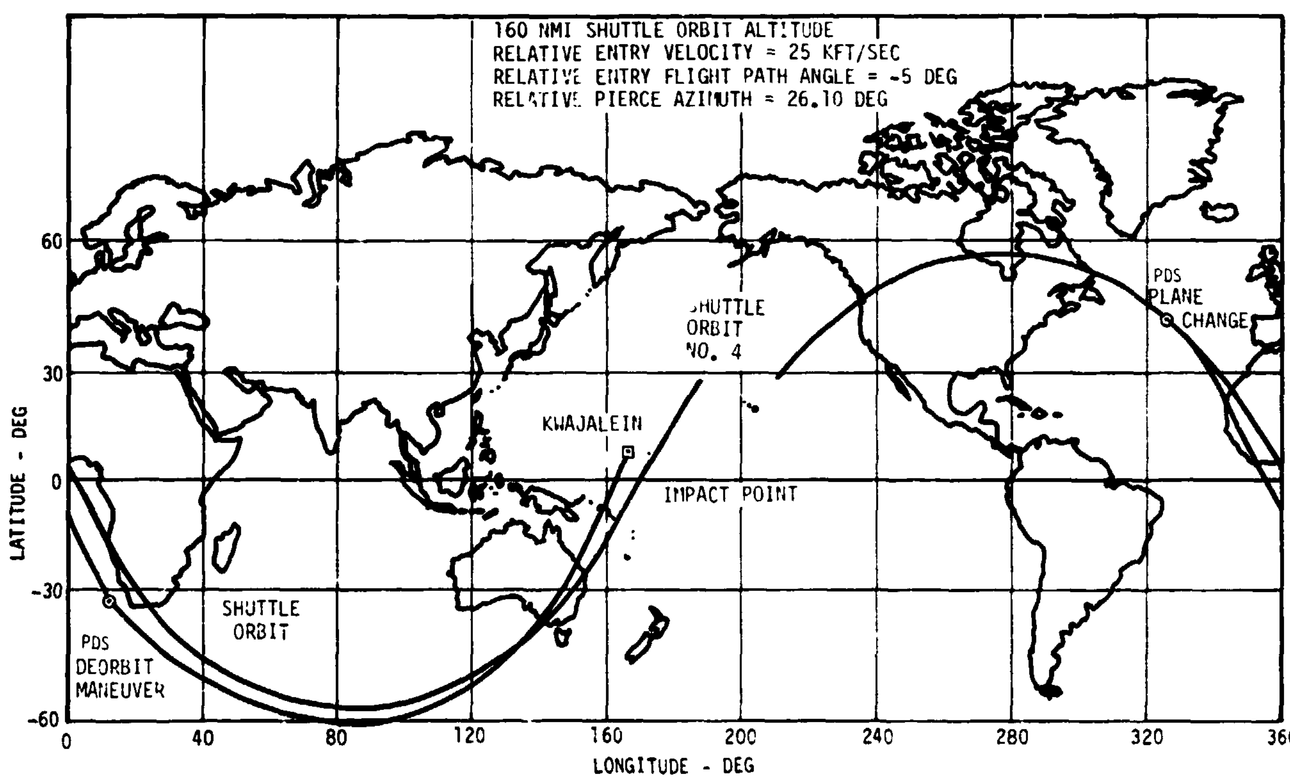

FIGURE 15i

CASES 2 & 3 DEORBIT TRAJECTORIES – FIRST OPPORTUNITY

KSC LAUNCH

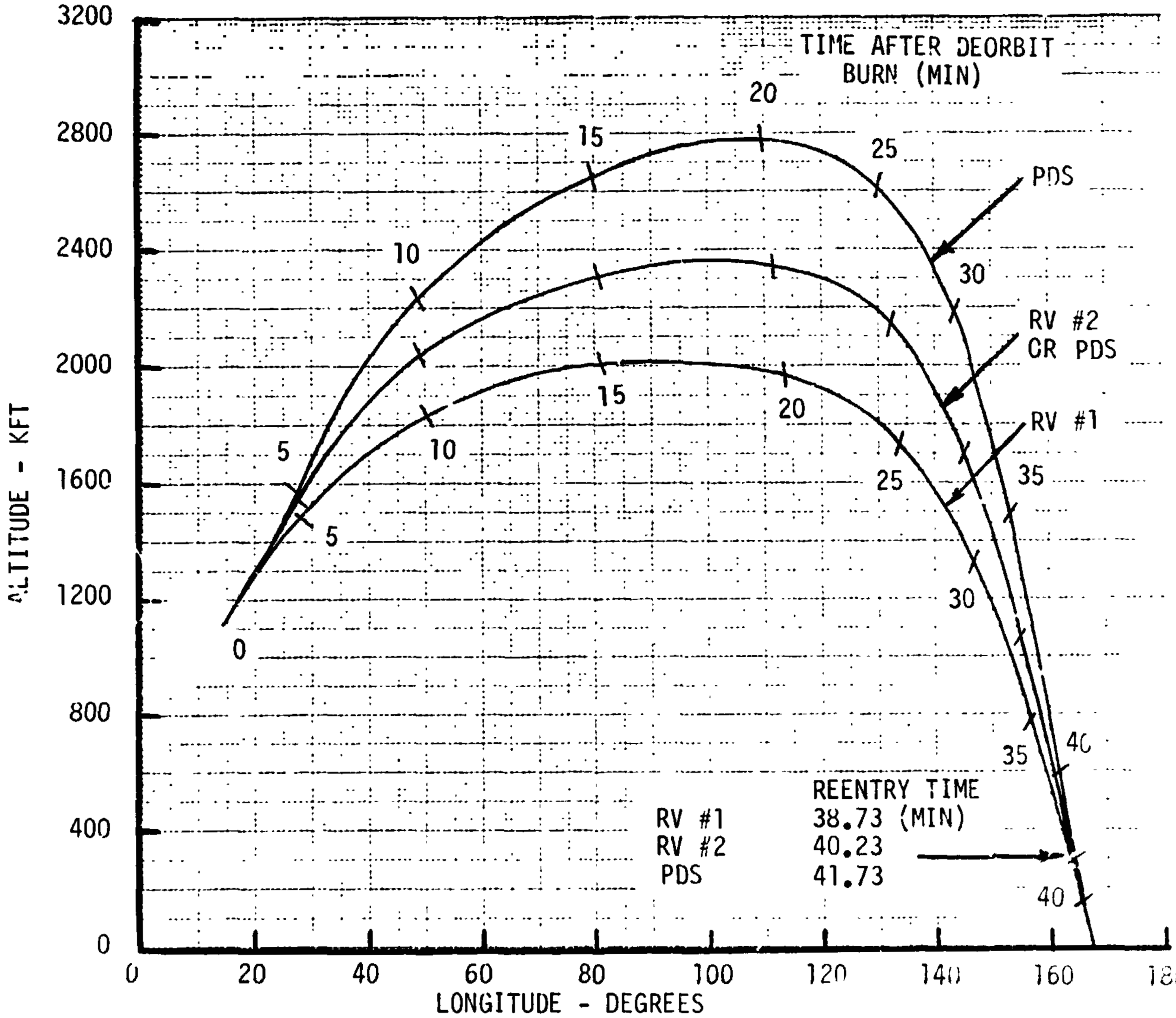

FIGURE 152

CASES 2 & 3 DEORBIT SUMMARY – FIRST OPPORTUNITY

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER		7456.25	43.61	326.12	972.2	139.68	63.31	24.61	0.00	2865
DEORBIT MANEUVER	1	8828.35	-31.41	12.59	972.2	147.79	41.01	24.16	4.57	1990
	2	8183.14	-40.61	20.52	1297.2	142.29	38.01	23.72	5.34	561
	3	7972.73	-43.42	23.56	1425.4	139.95	37.01	23.52	6.6[illegible]	631
ENTRY	1	561.08	.53	163.25	300.0	26.10	2.28	25.00	-5.00	--
	2	463.86	2.08	163.77	300.0	26.11	0.78	24.97	-6.19	--
	3	378.33	[illegible].48	164.18	300.0	26.14	-0.72	24.90	-7.57	--
IMPACT(NO ATMOSPHERE)	1	0.00	8.90	167.40	0.0	26.63	0.0	25.38	-4.96	--
	2	0.00	8.90	167.40	0.0	26.60	-1.07	25.33	-6.18	--
	3	0.00	8.90	167.40	0.0	26.61	-2.23	25.30	-7.58	--

*IMPACT OF FIRST PAYLOAD

TOTAL ΔV = 5416 FPS CASE #2
= 6047 FPS CASE #3

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FIGURE 153

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 2 FIRST OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 1
TOTAL MASS = 1000.0 LBM
NUMBER 1
M-LBM 1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	7410.4	270.0	7680.4	8680.4
PROPELLANT MASS (LBM)	.0	3659.4	.0	3659.4	3659.4
BURNOUT MASS (LBM)	.0	3751.0	270.0	4021.0	5021.0
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	1102.6			1102.6
BURN TIME (MIN)	.00	1.17			1.17
MASS FRACTION	.000	.494			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 19372.65 LBM
PERCENT OFFLOAD = 84.11 PERCENT
EXCESS DELTA-V = 11371.42 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	2865.0	.71	669.2	8680.4	6459.3	2221.1	2221.1	.0
2	1990.0	.33	361.2	6459.3	5260.6	1198.7	3419.8	1000.0
3	561.0	.08	72.2	4260.6	4021.0	239.6	3659.4	4021.0

FIGURE 154

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 3 FIRST OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 2000.0 LBM

NUMBER	1	2
M-LBM	1000.0	1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	8738.0	392.9	9130.8	11130.8
PROPELLANT MASS (LBM)	.0	4987.0	.0	4987.0	4987.0
BURNOUT MASS (LBM)	.0	3751.0	392.9	4143.9	6143.9
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	1502.6			1502.6
BURN TIME (MIN)	.00	1.59			1.59
MASS FRACTION	.000	.571			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 18045.03 LBM
PERCENT OFFLOAD = 78.35 PERCENT
EXCESS DELTA-V = 9341.39 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	2865.0	.91	858.1	11130.8	8282.8	2848.1	2848.1	.0
2	1990.0	.49	463.1	8282.8	6745.6	1537.1	4385.2	1000.0
3	561.0	.10	97.3	5745.6	5422.6	323.1	4708.3	1000.0
4	631.0	.09	84.0	4422.6	4143.9	278.7	4987.0	4143.9

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FIGURE 155

SHUTTLE MISSION SEQUENCE FOR PAYLOAD IMPACT AT KWAJALEIN

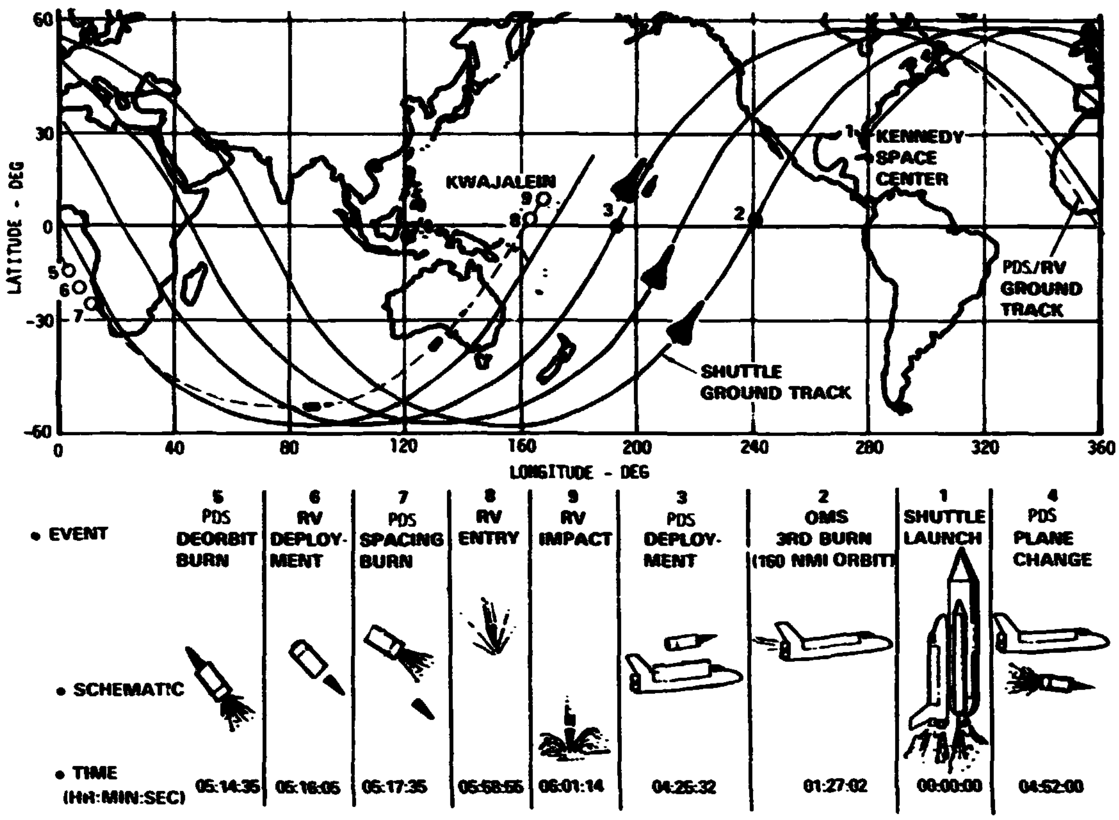

FIGURE 156

16.3 Example Cases 4 and 5 Results - These cases require deorbiting one or two 1000-lb payloads at Poker Flat with a relative velocity of 25 kfps and a flight path angle of -5 deg. These reentry conditions are readily achievable from Shuttle orbit. In addition, because there are no known azimuth constraints at Poker Flat, many deorbit opportunities exist.

Figure 157 shows the sensitivity of deorbit burn location to pierce point relative azimuth. Note that the deorbit locations are south of Australia and the 40 to 60 deg relative azimuth tracks pass over KMR and Guam.

Figure 158 provides the working data which define the pierce point azimuth for a single plane change maneuver and launches from either KSC or VAFB. The intercepts of the straight lines with the deorbit burn azimuth curve define pierce point relative azimuths achievable with a single plane change. There are two possibilities for orbit 5 and 6 from a KSC launch and three possibilities for orbits 12, 13 and 14 from a VAFB launch.

Figures 159 and 160 provide the ground tracks for the KSC and VAFB launches, respectively. In Section 12, an orbit 5 from KSC and 12 from VAFB are favored because of ground coverage from Australia and KMR. Note that the plane change maneuvers for all cases occur in the South Atlantic.

Deorbit trajectories are provided in Figure 161 for the first opportunity from the KSC and VAFB launch. Reentry time for the VAFB launch is shorter because of the slightly different inertial velocities required.

Figures 162 through 166 provide the detailed trajectory information for each of the opportunities. Note the much higher ΔV required for the KSC launches compared to the VAFB launches. This is because a significant orbital inclination change is required for the KSC launches. For all cases the pierce point is significantly displaced from the impact point because of the shallow entry angles.

Figures 167 through 176 provide the design summaries for these missions. Again Transtage is oversized for all the cases. The VAFB launches for case 4 have a minimum of 87% propellant offloaded. In this case a strong need for the Minibus concept is indicated. Generally, Shuttle provides excellent mission flexibility for providing payload impact at Poker Flat. Many deorbit opportunities are available, ΔV requirements are minimal, and ground coverage is available.

16.4 Example Case 6 Results - This case requires deorbiting six payloads and the booster near Meck Island at KMR with a very close spacing of payloads in both

CASES 4 & 5 SENSITIVITY OF DEORBIT GROUND TRACK TO PIERCE POINT AZIMUTH AT POKER FLAT

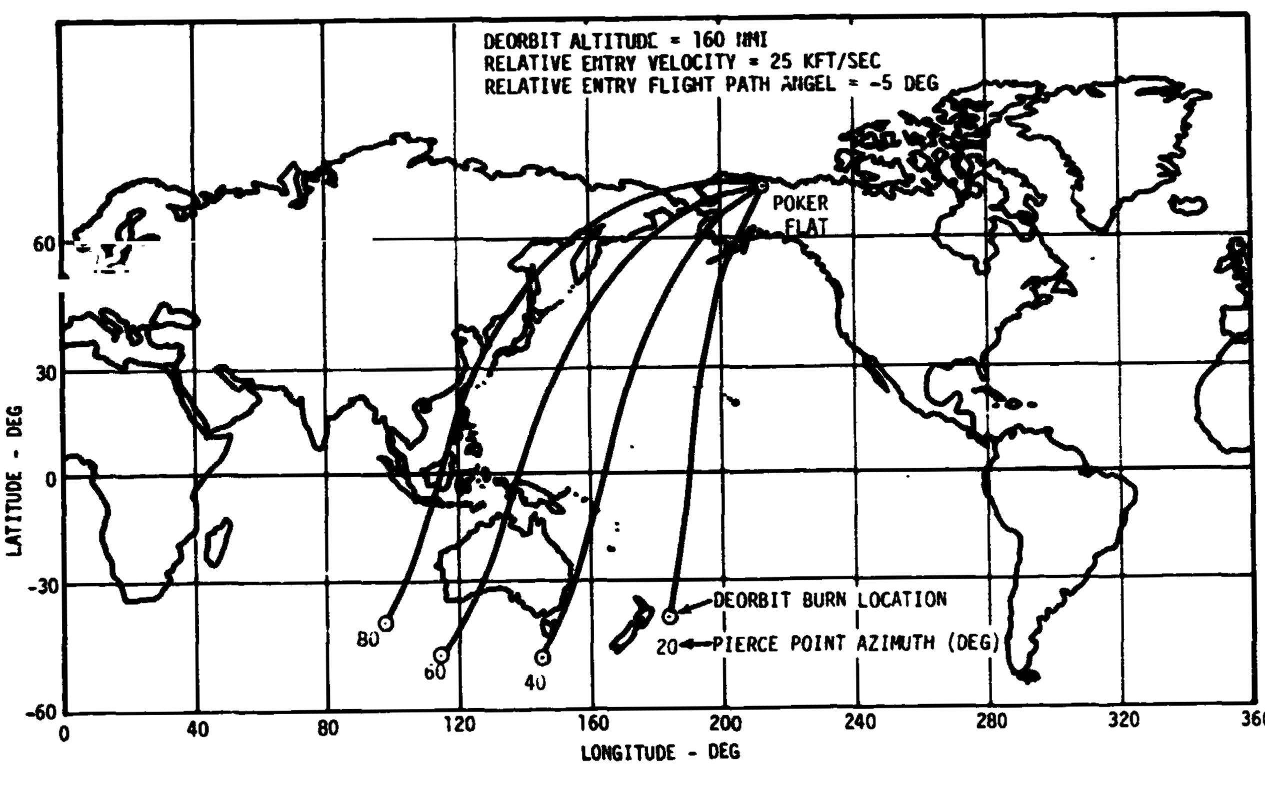

FIGURE 157

CASES 4 & 5 PIERCE POINT AZIMUTH DEFINITION

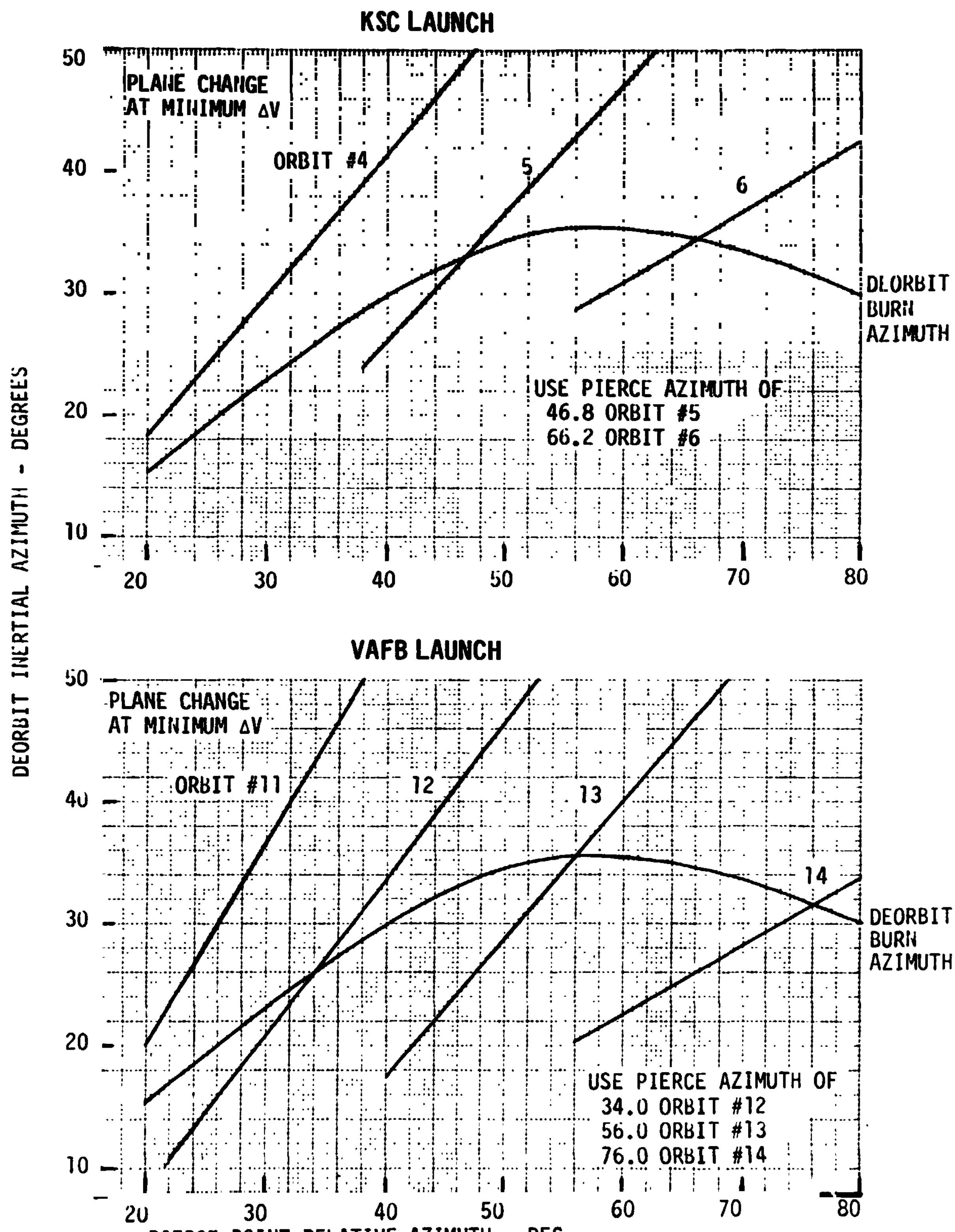

FIGURE 158

CASES 4 & 5 MISSION GROUND TRACK & MANEUVER LOCATIONS

KSC LAUNCH

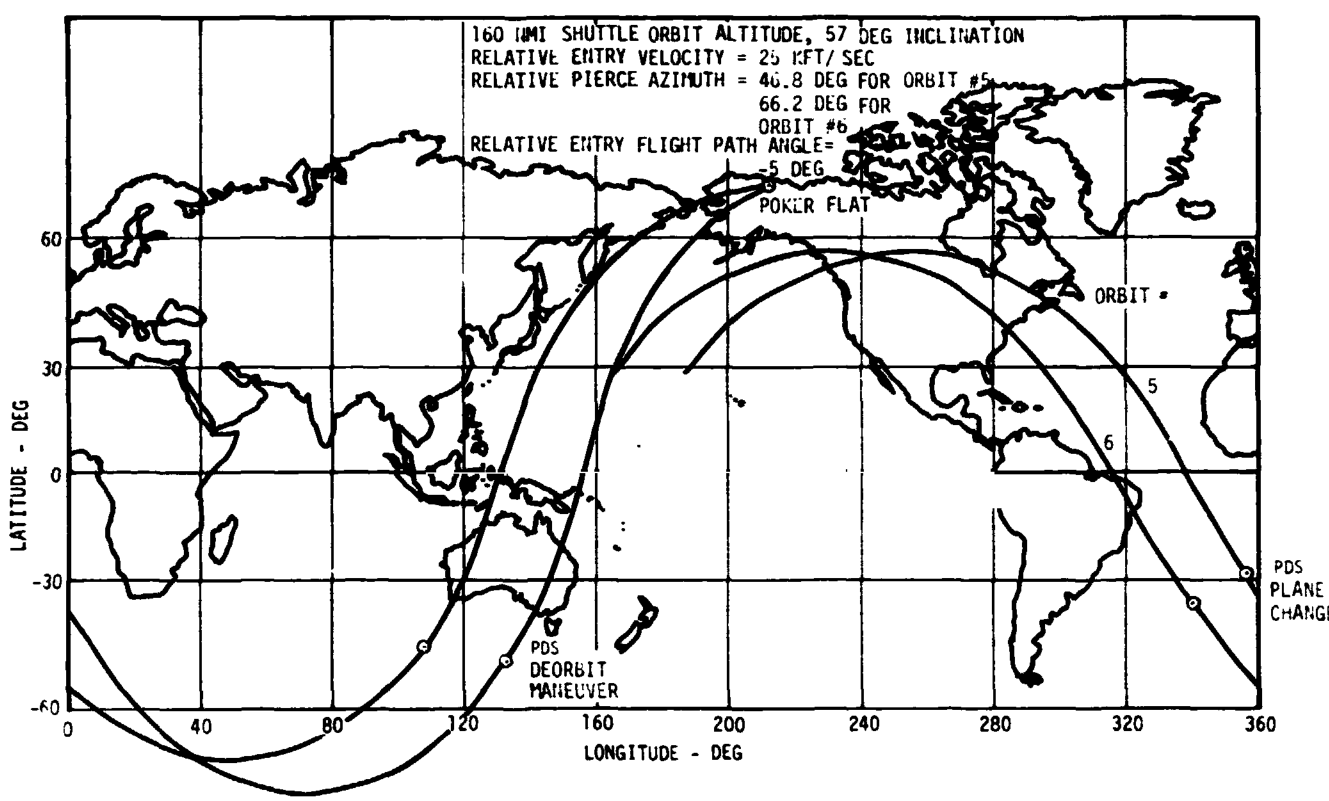

FIGURE 159

CASES 4 & 5 MISSION GROUND TRACK & MANEUVER LOCATIONS

VAFB LAUNCH

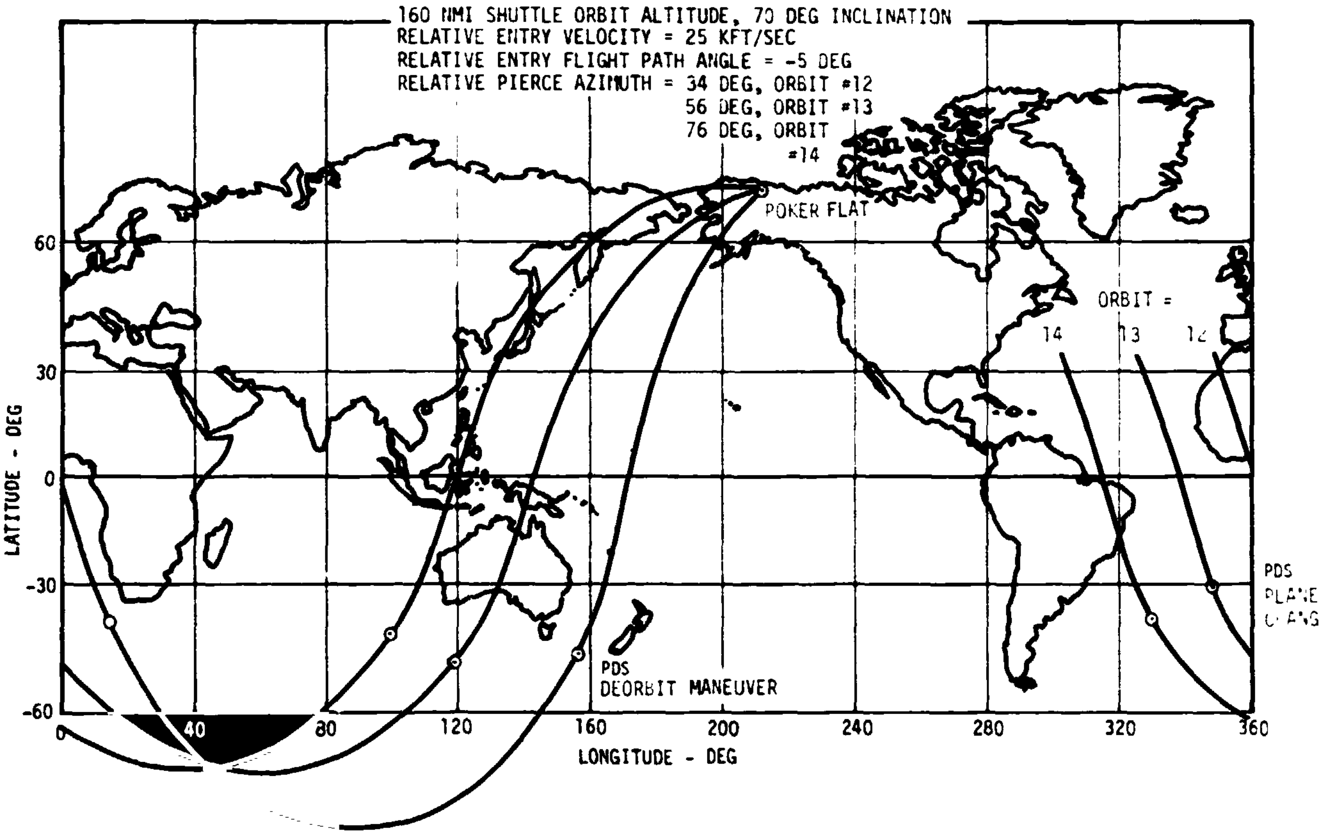

FIGURE 160

CASES 4 & 5 DEORBIT TRAJECTORIES – FIRST OPPORTUNITY

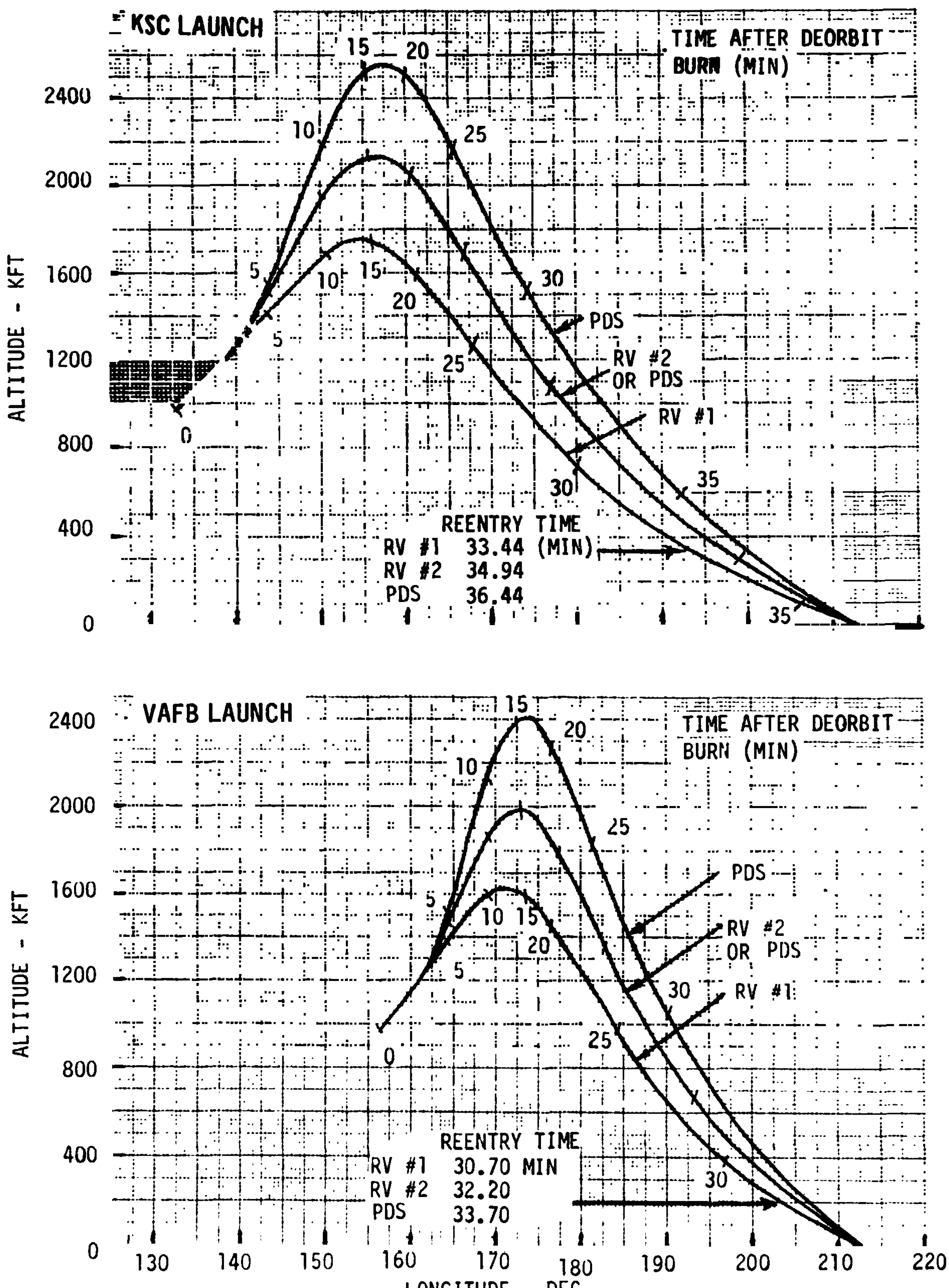

FIGURE 161

CASES 4 & 5 DEORBIT SUMMARY – FIRST OPPORTUNITY KSC LAUNCH

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER		8151.46	-28.50	356.32	972.2	159.88	58.29	24.84	0.0	6712
DEORBIT MANEUVER	1	7910.40	-50.13	132.78	972.2	31.28	35.69	24.18	4.21	1910
	2	7234.83	-40.19	140.26	1265.0	24.79	32.69	23.72	5.28	747
	3	7017.72	-36.89	142.14	1390.8	23.25	31.69	23.43	6.84	775
ENTRY	1	553.72	61.69	195.47	300.0	46.80	2.25	25.00	-5.00	--
	2	425.86	63.16	198.80	300.0	50.23	0.75	24.87	-6.62	--
	3	338.09	64.13	201.27	300.0	52.86	-0.75	24.77	-8.33	--
IMPACT (NO ATMOSPHERE)	1	0.0	67.00	213.00	0.0	63.70	0.00	25.37	-5.09	--
	2	0.0	67.00	213.00	0.0	63.58	-0.96	25.26	-6.76	--
	3	0.0	67.00	213.00	0.0	63.88	-2.12	25.13	-8.49	--

TOTAL ΔV = 9369 FPS CASE 4
10144 FPS CASE 5

*IMPACT OF FIRST PAYLOAD

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FIGURE 162

CASES 4 & 5 DEORBIT SUMMARY – SECOND OPPORTUNITY
KSC LAUNCH

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER		8250.48	-36.13	340.40	972.2	152.58	60.11	24.74	0.00	5482
DEORBIT MANEUVER	1	8267.14	-46.65	108.24	972.2	32.53	37.51	24.19	4.33	1930
	2	7591.90	-36.85	115.65	1276.6	26.47	34.51	23.72	5.30	675
	3	7375.19	-33.60	117.57	1403.6	25.01	33.51	23.49	6.75	719
ENTRY	1	556.45	64.86	190.78	300.0	66.20	2.26	25.00	-5.00	--
	2	438.57	65.62	195.10	300.0	70.59	0.76	24.90	-6.47	--
	3	351.38	66.10	198.45	300.0	74.08	-0.74	24.84	-8.06	--
IMPACT (NO ATMOSPHERE)	1	0.0	67.00	213.00	0.0	87.59	0.00	25.38	-5.05	--
	2	0.0	67.00	213.00	0.0	87.44	-0.99	25.30	-6.56	--
	3	0.0	67.00	213.00	0.0	87.82	-2.16	25.20	-8.17	--

*IMPACT OF FIRST PAYLOAD

TOTAL ΔV = 8087 FPS CASE #4
= 8806 FPS CASE #5

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FIGURE 163

CASES 4 & 5 DEORBIT SUMMARY – FIRST OPPORTUNITY
VAFB LAUNCH

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER		9068.14	-39.58	15.33	972.2	160.75	58.54	24.93	0.00	1983
DEORBIT MANEUVER	1	7356.91	-47.64	156.28	972.2	23.18	32.94	24.18	4.00	1860
	2	6680.49	-37.11	161.68	1245.6	18.41	29.94	23.69	5.07	796
	3	6463.27	-33.67	163.04	1365.9	17.27	28.94	23.36	6.78	866
ENTRY	1	549.84	60.00	199.67	300.0	34.00	2.24	25.00	-5.00	--
	2	417.01	61.86	202.14	300.0	36.61	0.74	24.84	-6.70	--
	3	324.91	63.14	203.99	300.0	38.67	-0.76	24.64	-8.59	--
IMPACT (NO ATMOSPHERE)	1	0.0	67.00	213.00	0.0	47.01	0.00	25.37	-5.17	--
	2	0.0	67.00	213.00	0.0	46.91	-0.95	25.20	-6.92	--
	3	0.0	67.00	213.00	0.0	47.11	-2.09	25.03	-8.84	--

*IMPACT OF FIRST PAYLOAD

TOTAL ΔV = 4639 FPS CASE #4
= 5505 FPS CASE #5

FIGURE 164

CASES 4 & 5 DEORBIT SUMMARY – SECOND OPPORTUNITY VAFB LAUNCH

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER		8105.88	-30.50	348.01	972.2	156.51	59.53	24.77	0.0	1303
DEORBIT MANEUVER	1	8154.74	-49.40	119.31	972.2	33.56	36.93	24.18	4.29	1930
	2	7479.50	-39.67	127.23	1273.5	26.82	33.93	23.72	5.29	695
	3	7262.30	-36.42	129.24	1399.9	25.22	32.93	23.49	6.78	736
ENTRY	1	565.78	63.13	192.50	300.0	56.00	2.26	25.00	-5.00	--
	2	434.03	64.26	196.74	300.0	59.85	0.76	24.90	-6.52	--
	3	347.22	65.00	199.64	300.0	62.90	-0.74	24.30	-8.14	--
IMPACT (NO ATMOSPHERE)	1	0.0	67.00	213.00	0.0	75.28	0.0	25.38	-5.06	--
	2	0.0	67.00	213.00	0.0	75.15	-0.98	25.26	-6.62	--
	3	0.0	67.00	213.00	0.0	75.47	-2.15	25.20	-8.27	--

*IMPACT OF FIRST PAYLOAD

TOTAL ΔV = 3928 FPS CASE #4
= 4664 FPS CASE #5

FIGURE 165

CASES 4 & 5 DEORBIT SUMMARY – THIRD OPPORTUNITY
VAFB LAUNCH

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
PLANE CHANGE MANEUVER		8186.65	-39.58	329.68	972.2	156.20	59.95	24.83	0.0	1878
DEORBIT MANEUVER	1	8232.79	-42.40	100.54	972.2	29.16	37.35	24.19	4.32	1930
	2	7558.26	-32.35	106.89	1274.7	24.27	34.35	23.75	5.13	619
	3	7341.34	-29.04	108.58	1397.5	23.09	33.35	23.49	6.59	720
ENTRY	1	556.18	66.57	189.41	300.0	76.00	2.26	25.00	-5.00	--
	2	447.90	66.96	193.87	300.0	80.53	0.76	24.90	-6.33	--
	3	357.56	67.17	197.69	300.0	84.49	-0.74	24.84	-7.92	--
IMPACT (NO ATMOSPHERE)	1	0.0	67.00	213.00	0.0	98.78	0.0	25.38	-5.05	--
	2	0.0	67.00	213.00	0.0	98.64	-1.03	25.30	-6.42	--
	3	0.0	67.00	213.00	0.0	98.95	-2.17	25.20	-8.03	--

TOTAL ΔV = 4427 FPS CASE #4

= 5147 FPS CASE #5

*IMPACT OF FIRST PAYLOAD

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FIGURE 166

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 4 FIRST OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 1
TOTAL MASS = 1000.0 LBM
NUMBER 1
M-LBM 1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	11733.5	270.0	12003.5	13003.5
PROPELLANT MASS (LBM)	.0	7982.5	.0	7982.5	7982.5
BURNOUT MASS (LBM)	.0	3751.0	270.0	4021.0	5021.0
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	2405.1			2405.1
BURN TIME (MIN)	.00	2.55			2.55
MASS FRACTION	.000	.680			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 15049.48 LBM
PERCENT OFFLOAD = 65.34 PERCENT
EXCESS DELTA-V = 7453.50 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	6712.0	2.07	1957.5	13003.5	6506.7	6496.8	6496.8	-.0
2	1910.0	.37	350.6	6506.7	5343.1	1163.6	7660.4	1000.0
3	747.0	.10	97.0	4343.1	4021.0	322.1	7982.5	4021.0

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FIGURE 167

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 4 SECOND OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 1
TOTAL MASS = 1000.0 LBM
NUMBER 1
M-LBM 1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	10138.6	270.0	10408.6	11408.6
PROPELLANT MASS (LBM)	.0	6387.6	.0	6387.6	6387.6
BURNOUT MASS (LBM)	.0	3751.0	270.0	4021.0	5021.0
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	1924.6			1924.6
BURN TIME (MIN)	.00	2.04			2.04
MASS FRACTION	.000	.630			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 16644.40 LBM
PERCENT OFFLOAD = 72.27 PERCENT
EXCESS DELTA-V = 8721.98 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	5482.0	1.57	1484.7	11408.6	6480.9	4927.7	4927.7	-.0
2	1930.0	.37	352.5	6480.9	5311.0	1170.0	6097.6	1000.0
3	675.0	.09	87.4	4311.0	4021.0	290.0	6387.6	4021.0

FIGURE 168

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 4 FIRST OPPORTUNITY
VAFB LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 1
TOTAL MASS = 1000.0 LBM
NUMBER 1
M-LBM 1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	6705.3	270.0	6975.3	7975.3
PROPELLANT MASS (LBM)	.0	2954.3	.0	2954.3	2954.3
BURNOUT MASS (LBM)	.0	3751.0	270.0	4021.0	5021.0
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	390.1			390.1
BURN TIME (MIN)	.00	.94			.94
MASS FRACTION	.000	.441			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 20077.72 LBM
PERCENT OFFLOAD = 87.17 PERCENT
EXCESS DELTA-V = 12192.65 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	1983.0	.47	444.5	7975.3	6499.9	1475.4	1475.4	.0
2	1860.0	.36	341.9	6499.9	5365.1	1134.8	2610.2	1000.0
3	796.0	.11	103.7	4365.1	4021.0	344.1	2954.3	4021.0

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FIGURE 169

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 4 SECOND OPPORTUNITY
VAFB LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 1
TOTAL MASS = 1000.0 LBM
NUMBER 1
M-LBM 1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	6155.7	270.0	6425.7	7425.7
PROPELLANT MASS (LBM)	.0	2404.7	.0	2404.7	2404.7
BURNOUT MASS (LBM)	.0	3751.0	270.0	4021.0	5021.0
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	724.5			724.5
BURN TIME (MIN)	.00	.77			.77
MASS FRACTION	.000	.391			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 20627.26 LBM
PERCENT OFFLOAD = 89.56 PERCENT
EXCESS DELTA-V = 12884.75 FT/SEC

BURN SUMMARY

BURN NO.	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	1303.0	.30	231.4	7425.7	6491.8	933.9	933.9	.0
2	1930.0	.37	353.1	6491.8	5319.0	1171.9	2105.8	1000.0
3	695.0	.10	90.0	4319.0	4021.0	298.9	2404.7	4021.0

FIGURE 170

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 4 THIRD OPPORTUNITY
VAFB LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 1
TOTAL MASS = 1000.0 LBM
NUMBER 1
M-LBM 1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W/ PAY
LAUNCH MASS (LBM)	.0	6559.6	270.0	6829.6	7829.6
PROPELLANT MASS (LBM)	.0	2808.6	.0	2808.6	2808.6
BURNOUT MASS (LBM)	.0	3751.0	270.0	4021.0	5021.0
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	846.2			846.2
BURN TIME (MIN)	.00	.90			.90
MASS FRACTION	.000	.428			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 20223.44 LBM
PERCENT OFFLOAD = 87.81 PERCENT
EXCESS DELTA-V = 12371.42 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	1873.0	.44	415.5	7829.6	6450.6	1378.9	1378.9	.0
2	1930.0	.37	350.9	6450.6	5286.1	1164.5	2543.4	1000.0
3	619.0	.08	79.9	4286.1	4021.0	265.1	2808.6	4021.0

FIGURE 171

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 5 FIRST OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 2000.0 LBM

NUMBER	1	2
M-LBM	1000.0	1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/C PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	14468.9	392.9	14861.7	16361.7
PROPELLANT MASS (LBM)	.0	10717.9	.0	10717.9	10717.9
BURNOUT MASS (LBM)	.0	3751.0	392.9	4143.9	6143.9
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	3229.3			3229.3
BURN TIME (MIN)	.00	3.42			3.42
MASS FRACTION	.000	.741			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 12314.14 LBM
PERCENT OFFLOAD = 53.47 PERCENT
EXCESS DELTA-V = 5315.19 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	6712.0	2.69	2538.3	16361.7	8437.3	8424.4	8424.4	-.0
2	1910.0	.48	454.6	8437.3	6928.4	1508.9	9933.3	1000.0
3	747.0	.14	132.5	5928.4	5488.7	439.7	10373.0	1000.0
4	775.0	.11	103.9	4488.7	4143.9	344.9	10717.9	4143.9

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FIGURE 172

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 5 SECOND OPPORTUNITY
KSC LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 2000.0 LBM

NUMBER	1	2
M-LBM	1000.0	1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	12336.5	392.9	12729.3	14729.3
PROPELLANT MASS (LBM)	.0	8585.5	.0	8585.5	8585.5
BURNOUT MASS (LBM)	.0	3751.0	392.9	4143.9	6143.9
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	2586.8			2586.8
BURN TIME (MIN)	.00	2.74			2.74
MASS FRACTION	.000	.696			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 14446.54 LBM
PERCENT OFFLOAD = 62.72 PERCENT
EXCESS DELTA-V = 6625.88 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	5482.0	2.03	1916.9	14729.3	8367.4	6362.0	6362.0	-.0
2	1930.0	.48	455.1	8367.4	6856.8	1510.5	7872.5	1000.0
3	675.0	.13	118.7	5856.8	5462.9	393.9	8266.4	1000.0
4	719.0	.10	96.1	4462.9	4143.9	319.0	8585.5	4143.9

FIGURE 173

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 5 FIRST OPPORTUNITY
VAFB LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 2000.0 LBM

NUMBER	1	2
M-LBM	1000.0	1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	8019.3	392.9	8412.1	10412.1
PROPELLANT MASS (LBM)	.0	4268.3	.0	4268.3	4268.3
BURNOUT MASS (LBM)	.0	3751.0	392.9	4143.9	6143.9
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	1286.0			1286.0
BURN TIME (MIN)	.00	1.36			1.36
MASS FRACTION	.000	.532			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 18763.73 LBM
PERCENT OFFLOAD = 81.47 PERCENT
EXCESS DELTA-V = 9988.43 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	1983.0	.61	580.4	10412.1	8486.0	1926.2	1926.2	.0
2	1860.0	.47	446.4	8486.0	7004.4	1481.5	3407.7	1000.0
3	796.0	.15	142.6	6004.4	5531.1	473.3	3881.0	1000.0
4	866.0	.12	116.7	4531.1	4143.9	387.2	4268.3	4143.9

FIGURE 174

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 5 SECOND OPPORTUNITY
VAFB LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 2000.0 LBM

NUMBER	1	2
M-LBM	1000.0	1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	7206.9	392.9	7599.7	9599.7
PROPELLANT MASS (LBM)	.0	3455.9	.0	3455.9	3455.9
BURNOUT MASS (LBM)	.0	3751.0	392.9	4143.9	6143.9
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	1041.3			1041.3
BURN TIME (MIN)	.00	1.10			1.10
MASS FRACTION	.000	.480			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 19576.11 LBM
PERCENT OFFLOAD = 85.00 PERCENT
EXCESS DELTA-V = 10775.92 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	1303.0	.39	363.8	9599.7	8392.4	1207.4	1207.4	-.0
2	1930.0	.48	456.5	8392.4	6877.3	1515.0	2722.4	1000.0
3	695.0	.13	122.5	5877.3	5470.7	406.6	3129.0	1000.0
4	736.0	.10	98.5	4470.7	4143.9	326.9	3455.9	4143.9

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FIGURE 175

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 5 THIRD OPPORTUNITY
VAFB LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 2
TOTAL MASS = 2000.0 LBM
NUMBER 1 2
M-LBM 1000.0 1000.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	7713.9	392.9	8106.8	10106.8
PROPELLANT MASS (LBM)	.0	3962.9	.0	3962.9	3962.9
BURNOUT MASS (LBM)	.0	3751.0	392.9	4143.9	6143.9
LENGTH (FT)	.00	14.83	2.83	17.67	25.67
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	1194.0			1194.0
BURN TIME (MIN)	.00	1.26			1.26
MASS FRACTION	.000	.514			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 19069.10 LBM
PERCENT OFFLOAD = 82.79 PERCENT
EXCESS DELTA-V = 10276.99 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	1878.0	.57	536.3	10106.8	8326.8	1780.0	1780.0	.0
2	1930.0	.48	452.9	8326.8	6823.6	1503.2	3283.2	1000.0
3	619.0	.11	108.5	5823.6	5463.4	360.2	3643.4	1000.0
4	720.0	.10	96.3	4463.4	4143.9	319.5	3962.9	4143.9

FIGURE 176

time and range. Details of the spacing requirements are contained in Reference 21). The purpose of this mission is to provide targets for the Site Defense Radar located on Meck Island. The nominal reentry conditions are 22.5 kft/sec and a -28 deg flight path angle. Because the SDR is a phased array radar facing northeast, only approaches from the north are possible. This dictates a due south launch from VAFB.

Figure 177 shows the sensitivity of deorbit burn location to pierce point relative azimuth. Note that some of the deorbit locations are over Russia. Figure 178 provides the working data which defines the pierce point azimuth for a combined plane change-deorbit maneuver. In the top plot, the intercepts of the orbit 3 and 4 ground tracks with the loci of deorbit burns are positions at which a combined burn is feasible. The difference in required and actual azimuth at these points is indicated in the bottom plot. Note that orbit 3 has almost the required azimuth. Therefore, a combined burn on orbit 3 is possible.

Figure 179 provides the resultant ground track for this maneuver and Figure 180 the deorbit trajectory of the first payload. Another opportunity requiring a separate plane change maneuver over Russia on orbit 4 is possible but was not considered in detail. An orbit 3 deorbit summary is provided in Figure 181. Note that the ΔV requirements are dominated by the deorbit burn. Spacing burns are only for slight crossrange and altitude spacing at pierce. The payload summary of Figure 182 indicates again the Transtage is oversized for this mission and a smaller booster would be more ideal. Because of the combined burn, a solid rocket could be used to provide the high ΔV burn and small vernier rockets could provide the spacing.

16.5 Summary of Example Case Results - Figure 183 summarizes the Shuttle orbit conditions and PDS requirements for all cases. In cases 1 and 6 a combined plane change and deorbit burn was identified indicating the application for a SRB. For case 2 and 3 only one deorbit opportunity was established; a second opportunity would require approximately 10,000 fps plane change. Cases 4 and 5 are achievable from either VAFB or KSC with multiple opportunities on consecutive orbits. Total ΔV requirements are small for most missions. The most severe ΔV requirement exists for the first opportunity of the first case. Only 20% of the propellant needs off-loaded. However, by taking advantage of the combined burn over 42% offloading can be achieved. At the other extreme over 89% offloading is required for a case 4 launch from VAFB and a second opportunity deorbit. In fact for cases 2 through 5 a Minibus concept could perform these missions with much lower initial PDS mass and length. The combined burns of cases 1 and 6 could also be achieved with either a Minibus or SRB concept.

CASE 6 SENSITIVITY OF DEORBIT GROUND TRACK TO PIERCE POINT AZIMUTH AT MECK ISLAND

DEORBIT ALTITUDE = 160 NMI
RELATIVE ENTRY VELOCITY = 22.5 KFT/SEC
RELATIVE ENTRY FLIGHT PATH ANGLE = -28 DEG

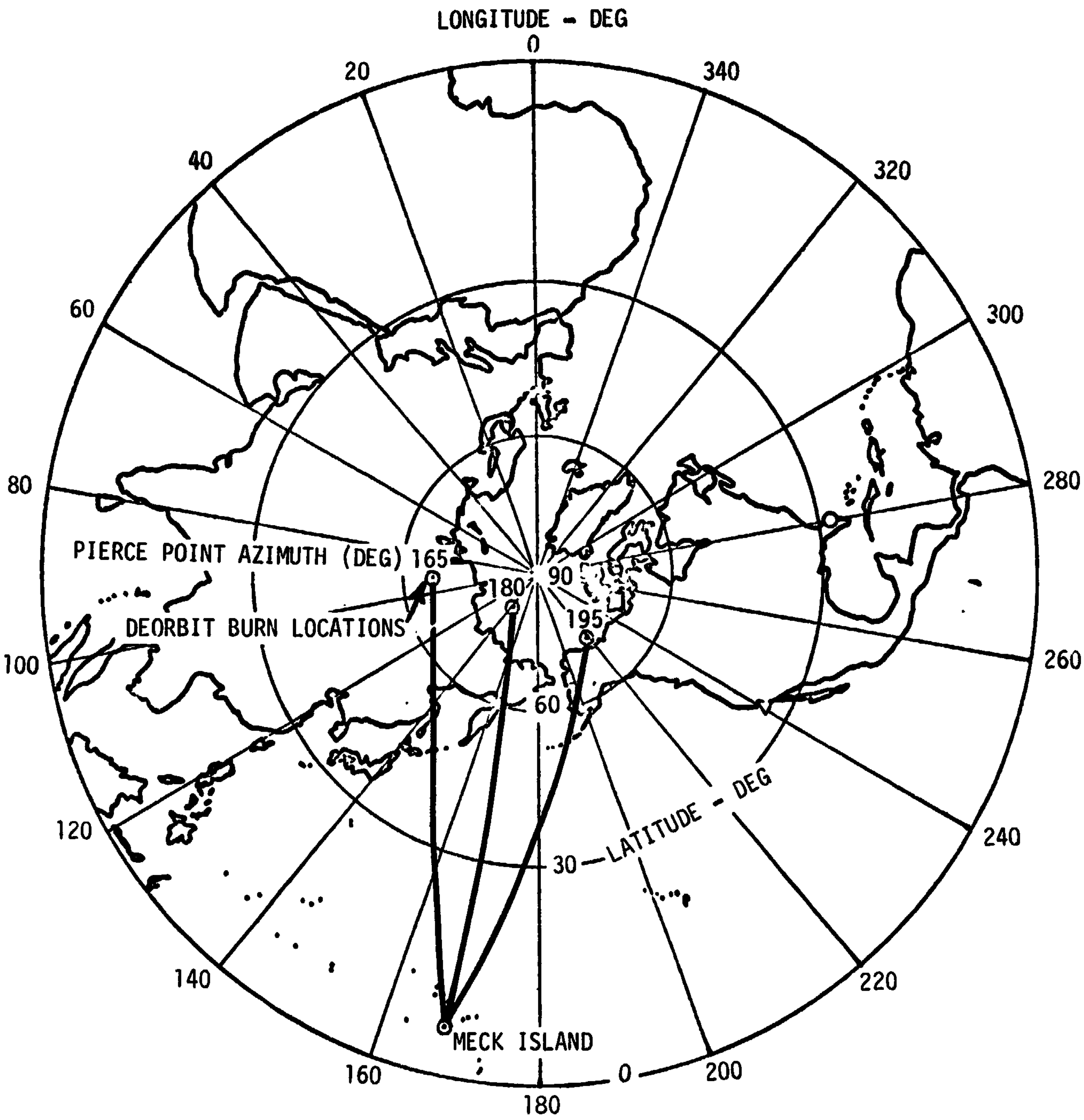
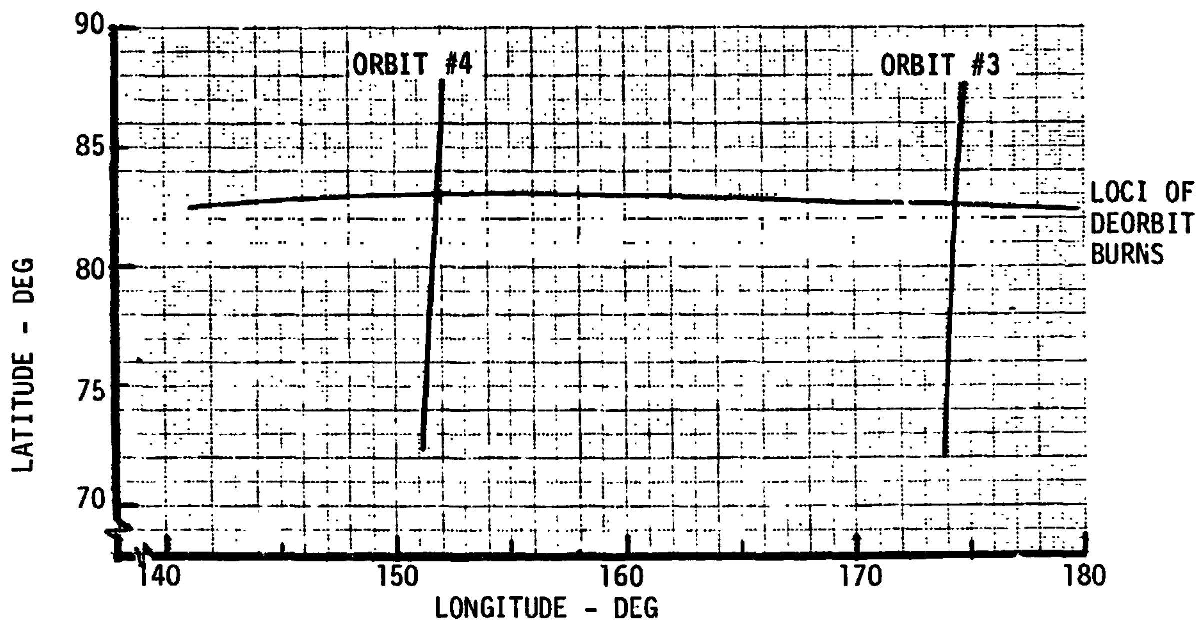


FIGURE 177

CASE 6 LATITUDE AND LONGITUDE SINGLE BURN LOCATIONS



CASE 6 PIERCE POINT AZIMUTH DEFINITION

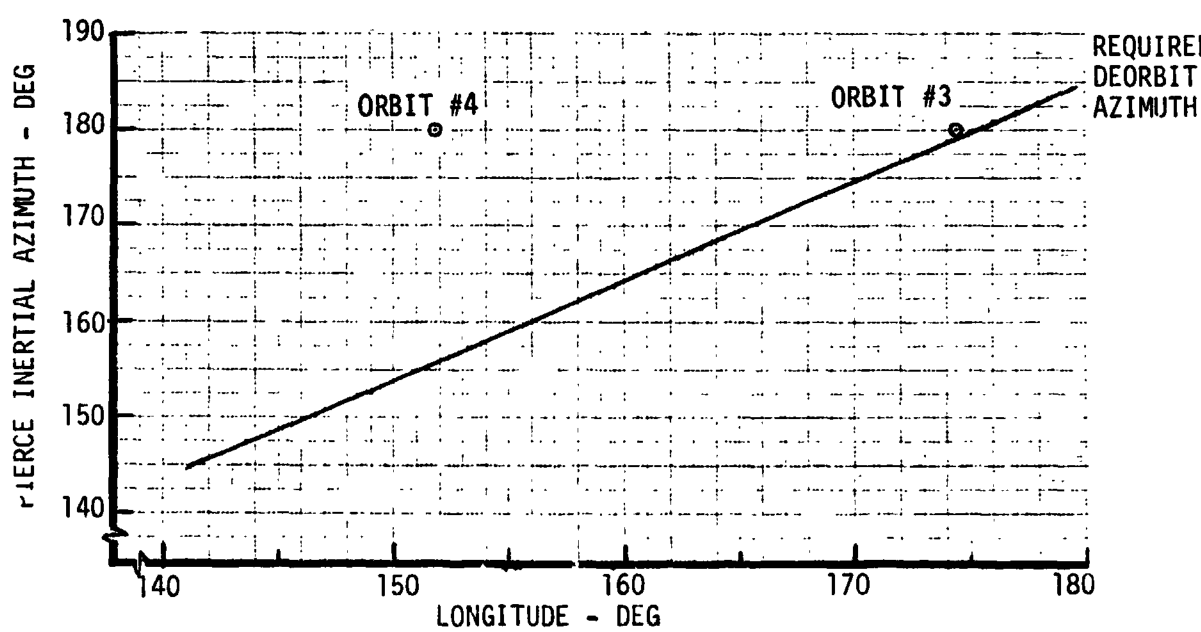


FIGURE 178

CASE 6 MISSION GROUND TRACK & MANEUVER LOCATIONS – SINGLE BURN OPPORTUNITY VAFB LAUNCH

160 NMI SHUTTLE ORBIT ALTITUDE, -90 DEG INCLINATION
RELATIVE ENTRY VELOCITY = 22.5 KFT/SEC
RELATIVE ENTRY FLIGHT PATH ANGLE = -28 DEG
RELATIVE PIERCE AZIMUTH = 184.275 DEG

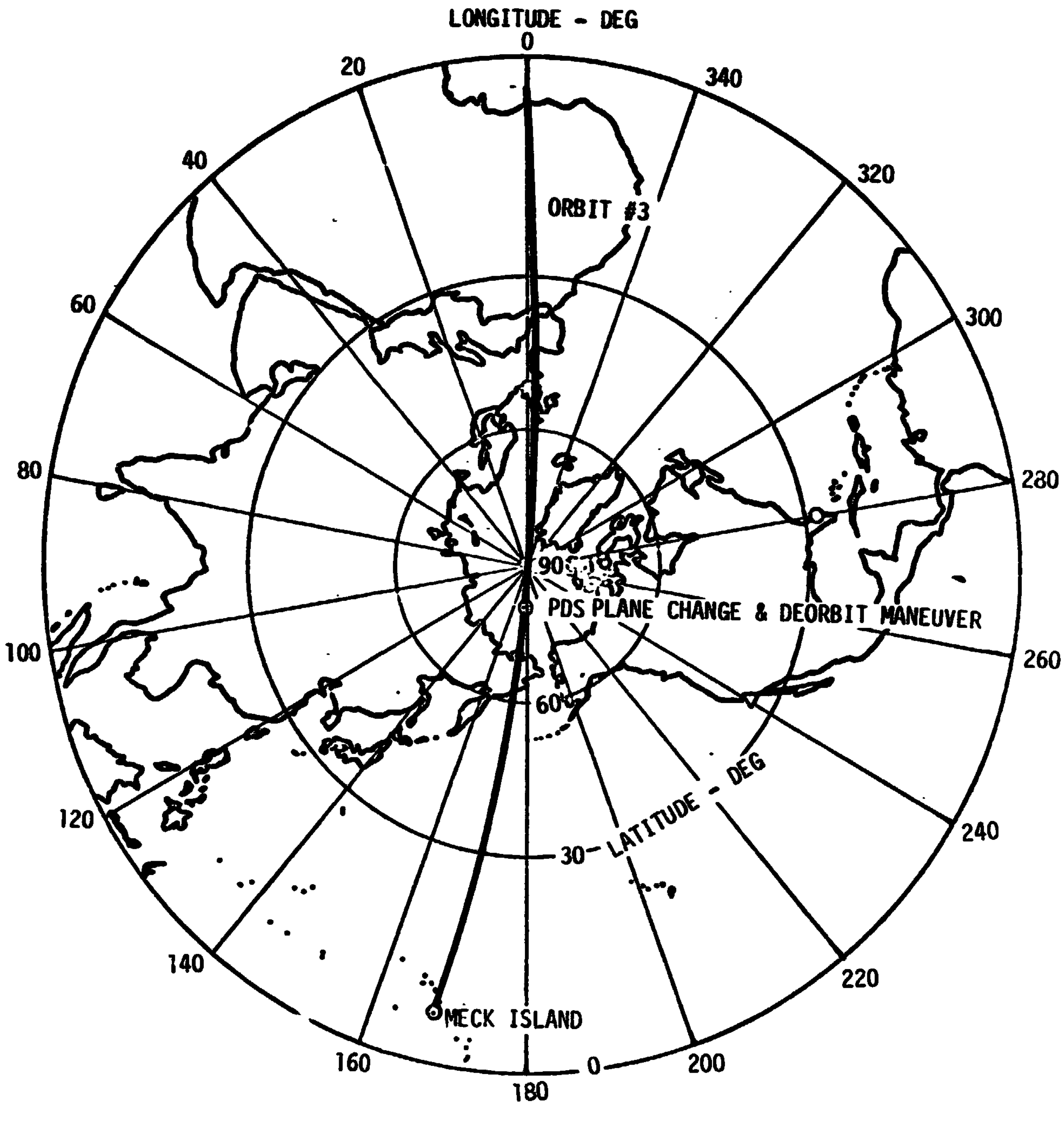


FIGURE 179

CASE 6 DEORBIT TRAJECTORY – SINGLE BURN OPPORTUNITY
VAFB LAUNCH

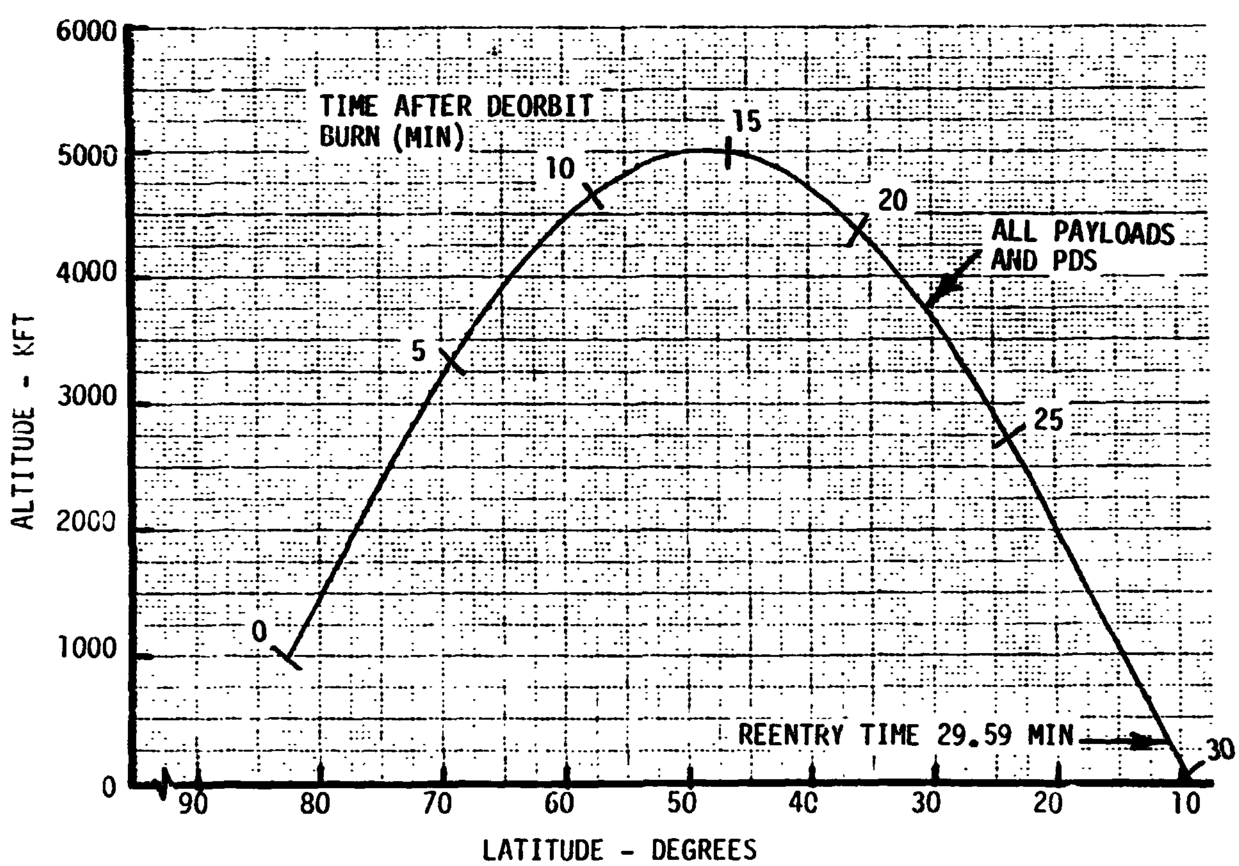


FIGURE 180

CASE 6 DEORBIT SUMMARY – SINGLE BURN OPPORTUNITY
VAFB LAUNCH

PHASE		GROUND RANGE TO IMPACT (NMI)	LATITUDE (DEG)	LONGITUDE (DEG)	ALTITUDE (KFT)	RELATIVE AZIMUTH (DEG)	TIME TO IMPACT* (MIN)	RELATIVE VELOCITY (KFT/SEC)	RELATIVE FLIGHT PATH ANGLE (DEG)	IMPULSIVE ΔV (FT/SEC)
DEORBIT MANEUVER	1	4377.68	82.68	174.47	972.2	179.81	30.05	21.52	26.86	[illegible]
	2	3221.20	63.35	173.19	4120.6	-177.44	22.59	17.39	14.84	[illegible]
	3	3152.15	62.20	173.07	4249.1	-177.36	22.09	17.22	13.81	[illegible]
	4	3084.32	61.07	172.96	4367.3	-177.42	21.59	17.06	12.77	[illegible]
	5	3016.43	59.94	172.84	4475.5	-177.28	21.09	16.93	11.75	[illegible]
	6	2949.72	58.83	172.72	4573.9	-177.15	20.59	16.80	10.72	[illegible]
	7	2883.03	57.72	172.61	4662.7	-176.76	20.09	16.70	9.53	[illegible]
ENTRY	1	52.36	10.77	167.84	300.0	-175.73	0.47	22.50	-28.00	-
	2	52.36	10.77	167.84	300.0	-175.73	0.46	22.51	-28.02	-
	3	52.62	10.77	167.89	300.0	-175.76	0.46	22.51	-28.00	-
	4	54.07	10.77	168.05	300.0	-175.86	0.47	22.51	-27.99	-
	5	54.07	10.77	168.05	300.0	-175.87	0.46	22.51	-28.01	-
	6	54.07	10.77	168.05	300.0	-175.87	0.46	22.51	-28.02	-
	7	52.50	10.77	167.87	300.0	-175.74	0.47	22.51	-27.99	-
IMPACT (NO ATMOSPHERE)	1	0	9.25	167.73	0.0	-175.83	0.00	22.92	-28.46	-
	2	0	9.25	167.73	0.0	-175.83	-0.01	22.90	-28.47	-
	3	1.65	9.25	167.78	0.0	-175.86	0.00	22.93	[illegible]	-
	4	13.17	9.25	167.94	0.0	-175.96	-0.02	22.90	[illegible]	
	5	13.17	9.25	167.94	0.0	-175.97	0.00	22.90	[illegible]	-
	6	13.17	9.25	167.95	0.0	-175.98	-0.01	22.90	[illegible]	-
	7	3.29	9.25	167.75	0.0	-175.85	0.00	22.93	[illegible]	-

*IMPACT OF FIRST PAYLOAD

TOTAL ΔV = 11810.7 FPS

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FIGURE 181

TRANSTAGE PAYLOAD AND PERFORMANCE SUMMARY
CASE 6 SINGLE BURN OPPORTUNITY
VAFB LAUNCH

PAYLOAD SUMMARY

TOTAL NUMBER = 6
TOTAL MASS = 820.0 LBM

NUMBER	1	2	3	4	5	6
M-LBM	350.0	30.0	30.0	30.0	350.0	30.0

DESIGN SUMMARY	BOOSTER	BUS	SPIN SYS & ATTACHMENT	TOTAL W/O PAY	TOTAL W PAY
LAUNCH MASS (LBM)	.0	15981.9	588.1	16570.0	17390.0
PROPELLANT MASS (LBM)	.0	12230.9	.0	12230.9	12230.9
BURNOUT MASS (LBM)	.0	3751.0	588.1	4339.1	5159.1
LENGTH (FT)	.00	14.83	6.17	21.00	21.17
DIAMETER (FT)	.00	10.00	10.00		
TOTAL IMPULSE (KLB-SEC)	.0	3685.2			3685.2
BURN TIME (MIN)	.00	3.90			3.90
MASS FRACTION	.000	.765			
THRUST (KLB)	.000	15.733			

OFFLOAD REQUIREMENTS

OFFLOADED PROPELLANT = 10801.08 LBM
PERCENT OFFLOAD = 46.90 PERCENT
EXCESS DELTA-V = 4683.29 FT/SEC

BURN SUMMARY

BURN NO	DV (FT/SEC)	BURN-T (MIN)	I-TOT (KLB-S)	INITIAL (LBM)	B/O (LBM)	PROP (LBM)	TOTAL (LBM)	M-DEPLOY (LBM)
1	11515.0	3.86	3642.2	17390.0	5301.8	12088.2	12088.2	350.0
2	7.0	.00	1.1	4951.8	4948.2	3.6	12091.7	30.0
3	21.6	.00	3.3	4918.2	4907.3	10.9	12102.7	30.0
4	81.6	.01	12.3	4877.3	4836.4	40.9	12143.6	30.0
5	82.9	.01	12.3	4806.4	4765.5	40.9	12184.5	350.0
6	84.6	.01	11.6	4415.5	4377.1	38.4	12222.9	30.0
7	18.0	.00	2.4	4347.1	4339.1	8.1	12230.9	4339.1

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FIGURE 182

SUMMARY OF EXAMPLE CASE RESULTS

CASE	OPPORTUNITY	LAUNCH SITE	IMPACT AREA	SHUTTLE ORBIT #	SHUTTLE ORBIT INCLINATION(DEG)	TRANSTAGE TOTAL ΔV (FPS)	% OFFLOAD
1	FIRST	KSC	KMR	4	57	15772	20
	COMBINED BURN			4		13338	42
2	FIRST	KSC	KMR	4	57	5416	84
3	FIRST	KSC	KMR	4	57	6047	78
4	KSC-1ST	KSC	PF	5	57	9369	65
	-2ND			6		8087	72
	VAFB-1ST	VAFB		12	72	4639	87
	-2ND			13		3928	89
	-3RD			14		4427	87
5	KSC-1ST	KSC	PF	5	57	10144	53
	-2ND			6		8806	62
	VAFB-1ST	VAFB		12	72	5505	81
	-2ND			13		4664	85
	-3RD			14		5147	82
6	COMBINED BURN	VAFB	MECK	3	90	11810	46

FIGURE 183

17.0 COST

This ROM cost analysis includes cost estimates of Shuttle borne checkout and servicing equipment for DoD payloads and for any hardware or software item or service peculiar to Shuttle launched entry technology missions as compared to ballistically launched missions. The various unique items required for DoD payloads can be categorized as Shuttle, Shuttle/PDS interface, PDS booster, RV, and ground station requirements. Cost estimates are not presented for Shuttle launches, the PDS booster, the pallets for securing the PDS to the Shuttle structural interface, the RV spin separation system (it is virtually identical to a ground launch piece of hardware).

The significant equipment and modifications required for DoD missions are listed in Figure 184. The modifications to the Shuttle payload interrogator are recommended to improve the Shuttle to PDS telemetry link after PDS deployment. By removing the 32 KBPS voice processing equipment, 48 KBPS telemetry data can be processed. This modification will reduce the cost of the Payload Interrogator and is not included as a cost item in this section.

As stated in Section 12 there are three options for pre-deployment PDS checkout. These are:

(1) Low data rate checkout with existing Shuttle equipment

(2) Transmittal of PDS data to ground via the scientific data channels

(3) Installation of a DoD dedicated checkout console

The first and second options are implemented without significant cost to DoD payloads. However, option 1 significantly reduces the amount of data that can be processed. Option 2 removes payload control completely from Shuttle crew control. The checkout console of option 3 requires some component development, hardware purchasing, system integration and checkout, and programming. The equipment includes a bit synchronizer, data formatter, data comparator, data processor, CRT display, control console and command encoder as described in Section 12, Figure 119. The data comparator, data processor, and control console are new equipment consisting of microprocessors like the Intel 8080 and appropriate circuitry and hardware for console operation. The bit synchronizer, data formatter, and command encoder are off-the-shelf items. The CRT display must be space rated and is the most expensive piece of hardware in the console. The ROM estimate for the development of the console is $900K. The unit cost without programming is estimated at $150K per unit based upon Skylab related equipment costs. One unit will suffice for the DoD missions with a spare for ground testing and reprogramming. For each DoD mission,

SUMMARY OF SHUTTLE UNIQUE COST ITEMS

1. SHUTTLE PAYLOAD INTERROGATOR
   - o REMOVE VOICE CAPABILITY AND INCREASE DATA RATE TO 48 KBPS
2. SHUTTLE/PDS INTERFACE
   - o CHECKOUT CONSOLE
3. PDS
   - o 50 WATT, 1 FT DIAMETER STEERABLE, PARABOLIC DISH ANTENNA FOR TDRSS USE
4. RV
   - o COOLANT SYSTEM
   - o ACOUSTIC DESIGN LEVELS OF 135 db
   - o MORE RELIABLE COMPONENT DESIGN TO MINIMIZE GROUND LAUNCH DELAYS AND PRE-DEPLOYMENT CHECKOUT
5. GROUND STATIONS
   - o EQUIP SELECTED SCF STATIONS WITH USB TYPE COMMAND AND VOICE RECEIVERS

FIGURE 184

reprogramming and verification of the software will be required. This recurring cost will be as much as $100K per mission depending upon mission complexity.

The major cost item recommended for the PDS is a 50 watt transmitter with a 1 ft diameter steerable, parabolic dish antenna for TDRS data relay. This would be needed if PDS data is to be relayed to ground stations during PDS burns and maneuvers. This antenna could also serve for PDS/Shuttle communications during initial PDS deployment and plane change burn. Development of this system is required for the steering system, transmitter, and antenna. The ROM development costs of these units are estimated to be $400K with a unit cost of $50K. The $50K does represent a recurring cost since each PDS would require this system.

There are three recommendations for RV design changes identified in Figure 184. The coolant system design changes were costed based upon the radiator with phase change material concept of Figure 126 in Section 14. A coolant system is required because the RV radiation cooling in the Shuttle payload bay is not adequate to dissipate the heat generated during pre-deployment checkout. Typical coolant system ROM development cost for space application is $200K and the unit cost is $25K. Unit costs will be very sensitive to the location and distribution of electronic components in the RV. If they can be concentrated in one area, the electronic component heat exchanger system in the RV will be minimized. This would require a change in RV design philosophy.

The other two RV requirements for acoustic design levels and reliability also impact design philosophy. Although they were not costed, these recommendations will impact RV cost. The need for highly reliable designs can be traded for the amount of pre-deployment payload checkout, and anticipated Shuttle launch delays due to component replacement. Shuttle launched DoD payloads may be in orbit for 14 Shuttle orbits or 21 hours as compared to 30 minutes for a ground launch. Thus component lifetimes must be increased to meet these long duration missions and minimize aborted PDS deployment.

The final item called out in Figure 184 is a receiver compatibility change at the Air Force SCF stations to accommodate the NASA Unified S-Band and voice data. These modifications are more a matter of facility policy and will not contribute significantly to mission costs.

Figure 185 summarizes the three significant cost items for Shuttle launched DoD entry technology experiments, i.e., checkout console, PDS antenna, and RV cooling system. The total development cost is $1.5M and the unit costs $325K. The checkout console unit is reusable with reprogramming for each DoD mission and its recurring cost is $100K. These ROM cost estimates can be used with estimates of Shuttle launch and PDS booster costs to compare the total Shuttle mission costs with comparable ground launches.

COST SUMMARY FOR DOD PAYLOADS

ITEM	DEVELOPMENT $K	UNIT $K	RECURRING $K
o CHECKOUT CONSOLE	900	250	100
o PDS TRANSMITTER	400	50	50
o COOLANT SYSTEM	200	25	25
TOTAL	1500	325	175

FIGURE 185

18. CONCLUSIONS

The major conclusion of this study is that Shuttle can provide unique simulation capability for DoD reentry vehicle experiments. The areas of uniqueness are listed in Figure 186. They emphasize the fact that the Shuttle is a mobile launch platform, is at earth orbital velocity, has a high weight payload capability, and can provide payload overflight at many desirable impact areas.

A second conclusion is that the KREMS radars at KMR can be used to track and record reentries from Shuttle deployed experiments. This fact was established in Section 5 in which a southwest approach corridor to KMR was identified and provides full KREMS coverage to impact. As a consequence, the same type, amount, and quality of data can be acquired as is currently acquired from a VAFB to KMR launch into the northeast corridor at KMR. The southwest corridor is achievable from either a KSC or VAFB launch.

The analysis of Section 7 identified the Shuttle unique capability to provide high velocity shallow flight path angle reentries. Because the Shuttle is in a circular orbit with velocities near 25 kft/sec, it takes little booster energy to deorbit a payload from the Shuttle orbit at these velocities and shallow flight path angles. On the other hand, low velocity and/or steep reentry flight path angles require very large booster energies comparable to ground launch systems.

A preferred payload deorbit procedure was developed in Section 7 to provide maximum exoatmospheric payload trajectory simulation as well as minimal time between PDS deployment from Shuttle and payload impact. This strategy involves deploying the PDS from Shuttle; the PDS then performs a plane change maneuver to target it for the impact point; a deorbit burn is then made to place the PDS on a ballistic reentry trajectory; spacing burns follow to provide the desired payload spacing at the pierce point.

Because of the potential for a large number of burns and payloads, the PDS function is best served by a liquid propellant booster. The preferred PDS then consists of a booster, spin separation and attachment system, and the RV's as described in Section 11. Use of an existing booster and proven spin separation system will minimize development cost and risk of the PDS. In special cases requiring only one large deorbit burn a rocket motor booster could suffice.

For the high velocity shallow flight path angle reentry, Transtage class boosters are oversized as identified in Section 9. Considerably smaller boosters made up of Shuttle RCS components more efficiently perform the function of the PDS booster.

UNIQUE SHUTTLE REENTRY SIMULATION CAPABILITY

- SIMULATION OF 1500-7100 NMI TRAJECTORY
- SIMULATION OF HIGH VELOCITY (25 KFT/SEC) TRAJECTORIES
- SIMULATION OF FLAT REENTRIES (LOW ENTRY ANGLES)
- MULTIPLE PAYLOADS
  - RV'S
  - BUSES
  - SITE DEFENSE RADAR TARGETS
- IMPACT AREA SELECTION FLEXIBILITY
  - WEATHER EFFECTS
  - TERMINAL GUIDANCE OVER LAND
  - RANGE SAFETY PROBLEMS REDUCED

FIGURE 186

Accuracy of the pierce point does suffer as described in Section 10 because of tracking errors in Shuttle location and velocity. The downrange error in Shuttle velocity can result in pierce point downrange dispersions as large as 6.4 NMI per ft/sec velocity error. This dispersion can be tolerated by displacing the impact point the appropriate distance from populated areas to avoid range safety problems. Otherwise, more frequent Shuttle tracking updates, for instance immediately prior to PDS deployment, will reduce this error to a tolerable value.

Launching of DoD payloads from Shuttle imposes few constraints on payload communication systems. If available Shuttle equipment is to be used for payload checkout, then the maximum data rate from each payload should not exceed 64 KBPS and preferably be less than 16 KBPS. Where higher payload data rates are necessary, checkout will require either a separate onboard checkout console or the data must be transmitted to the ground via one of Shuttle's wide band data links. In the latter case, ground personnel will evaluate the payload status and relay the information to the crew. Transmission of data directly to ground from separated payloads requires standard (5 watts) transmitters and flush-mounted, omni-directional antennas. Communication via TDRS, however, will require high power (50 watts) transmitters and steerable, directional antennas. Command uplinks are straightforward except for some inconvenience that may be caused by differences in USB and SGLS techniques. Shuttle's ability to communicate with a separated payload is limited to about 20 NMI but it can radar track to 300 NMI, if the payload is equipped with a transponder.

Complete range safety studies are required before the range safety problems of Shuttle payload deployment can be assessed. In many situations, PDS burns are made with no ground coverage, and payloads are reentered over areas which have not been considered in previous range safety analyses. These considerations must be included in detailed range safety analyses.

Specific mission analyses of Section 16 established firmly the capability of Shuttle to deliver DoD payloads at KMR for either the KREMS radars or Site Defense Radar and at Poker Flat. In fact, Poker Flat targetting can be achieved from either a KSC or VAFB with multiple consecutive deorbit opportunities and minimum size PDS configurations.

The absolute cost of using Shuttle launches compared to ground launches is not defined in this study. In fact, overland flights into Poker Flat appear as a unique capability of Shuttle and, therefore, a comparison is unwarranted. Shuttle cos over and above traditional ground launches will involve primarily the communications aspects of the missions, i.e., checkout console, satellite or ship coverage during PDS burns, and special PDS equipment to interface with Shuttle data links.

This study has demonstrated that Shuttle can perform the DoD missions to the depth presented. The next section describes recommendations to further define the Shuttle role in DoD reentry testing.

19.0 RECOMMENDATIONS

Recommendations for additional studies and Shuttle, PDS, RV and support equipment changes have been identified in this study.

Six areas requiring more detailed study are KMR range safety, KREMS ALTAIR-TRADEX interaction, Poker Flat range development, Minibus design, TDRSS usage and onboard checkout. The KMR range safety study should address the range safety considerations for the southwest corridor at KMR. This would include the analyses to define range safety requirements for deorbiting payloads. The tracking coverage requirements should be defined from this study to enable detailed mission planning to be accomplished.

Investigation of the KREMS west-southwest corridor is required to define any problems involved with the TRADEX radiating at ALTAIR. This zone was considered closed during the study. If the TRADEX-ALTAIR interaction was found not to be a problem, the number of orbit opportunities at KMR would increase and the plane change requirements would be reduced.

More detailed analysis of the Poker Flat range is required. Range safety, weather conditions, support equipment, geography all need better definition. A detailed mission study for delivering PGRV to Poker Flat is recommended to fully define the utility of the Poker Flat range and what modifications and instrumentation are necessary to support DoD missions.

Because many of the DoD missions require a small deorbit booster of the Minibus class, design studies should be initiated to develop a design for use in the 1980's. This design study could be worked as part of the Poker Flat range development study. Output of the study would include a design, development schedule and life cycle costs for the Minibus concept.

The costs, procedures, and limitations of using the TDRSS to provide tracking, telemetry, and command links needs detailed investigation. By defining the costs of equipment and services for using this system, comparisons with ship or Shuttle tracking could be made and a preferred PDS tracking network established.

Detailed definition of checkout console hardware and software for pre-deployment checkout of the PDS is required. This information would serve for comparison with using existing Shuttle payload displays or sending the PDS data to ground stations for checkout. These comparisons will define the merits of a dedicated checkout console.

In addition to these study recommendations, there are many recommendations with respect to equipment modifications, improvements, and development. For instance, implementing the following recommendations will improve Shuttle's capability to launch DoD payloads.

a. PDS to Shuttle RF link
   1. Increase data rate from 16 KBPS to 48 KBPS or higher.
   2. Improve Payload Interrogator receive capability by at least 6 db.
b. PDS/Payload Telemetry System
   1. Provide a separate PCM output for pre-deployment checkout. This output to be within Shuttle bit rate limits.
c. PDS Command System
   1. Provide a USB compatible command receiver for either STDN or TDRS commands.
   2. Equip selected SCF stations with USB type command encoders.
d. PDS Tracking
   1. Provide a Ku-band transponder for Shuttle tracking.
e. RV Tracking
   1. Provide a C-band transponder for ground tracking.
f. PDS Checkout
   1. Provide a Shuttle checkout console for DoD payloads.

The PDS should be equipped with STDN compatible equipment. When Kwajalein is the impact point, the STDN station at Orroral, Australia will be the main tracking and command link, with C-band skin tracking from KMR and S-band telemetry reception. The PDS will have sufficient altitude that coverage from the two stations will overlap in some cases. Similarly, when Poker Flat is the impact point, the STDN station at Fairbanks will be the main tracking and command link with S-band skin tracking and S-band telemetry reception at Poker Flat. The PDS should have a Ku-band transponder so that Shuttle can track the PDS during plane change burn. Most orbital paths into Poker Flat will pass between Hawaii and Guam, with little coverage from either SGLS station. Where either station can track the PDS, telemetry and tracking are possible, but a command up-link may not be compatible since SGLS uses a different command code format than STDN.

A reconsideration of the design philosophy for RV's is also recommended. Component reliability must be improved to minimize checkout requirements and delayed Shuttle launches due to RV equipment failures. In addition, changes in subsystem cooling designs and acoustic vibration limits will make the RV's more compatible with the Shuttle environment. These considerations could significantly reduce the amount of pre-deployment checkout required.

Implementation of the above recommendations in the next few years will greatly reduce interface problems in the 1980's when DoD entry technology testing from Shuttle may be a reality. Early detailed studies and equipment development will provi'e an orderly transition to the use of Shuttle for some of the unique applications identified in this study.

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APPENDIX A

STRATEGY A & B IMPULSIVE ΔV, FLIGHT TIME AND RANGE COMPUTATION

Given: V_E = reentry velocity (inertial)

γ_E = reentry flight path angle

h_E = reentry altitude

h_S = Shuttle orbit altitude

Find: ΔV = deorbit velocity change required

Solve: Use reference (18) approach.

1. Compute deorbit trajectory eccentricity, e, and energy, E,

o Define deorbit trajectory eccentricity:

$$e = -\sqrt{\left(\frac{r_E V_E^2}{K} - 1\right)^2 \cos^2\gamma_E + \sin^2\gamma_E} \qquad \text{(A1)}$$

where $r_E = h_E + R_E$

R_E = earth radius

K = gravitational constant

Note $e < 0$ for subcircular ellipse

o Define deorbit trajectory energy:

$$E = \frac{V_E^2}{2} - \frac{K}{r_E} \qquad \text{(A2)}$$

2. Compute velocity, V_S, and flight path angle, γ_S, at Shuttle orbit altitude.

o Define deorbit velocity at Shuttle orbit altitude:

$$V_S = \sqrt{2\left(E + \frac{K}{r_S}\right)} \qquad \text{(A3)}$$

o Define deorbit flight path angle at Shuttle orbit altitude:

$$\cos\gamma_S = \sqrt{\frac{e^2 - 1}{\left(\frac{r_S V_S^2}{K} - 1\right)^2 - 1}} \qquad \text{(A4)}$$

where $r_S = h_S + R_E$

3. ΔV at Shuttle orbit is then computed

o Define Shuttle orbital velocity, V_o (assume circular orbit)

$$V_o = \sqrt{K / r_S} \qquad \text{(A5)}$$

o Then

$$\Delta V = \sqrt{V_o^2 + V_S^2 - 2 V_o V_S \cos\gamma_S} \qquad \text{(A6)}$$

where V_o is obtained from eq. A5

V_S is obtained from eq. A3

$\cos\gamma_S$ is obtained from eq. A4

4. Deorbit time computation

o Calculate semi-major axis a

$$a = -K/2E \tag{A7}$$

o Calculate position angles θ_E, θ_S, at reentry and Shuttle deorbit, respectively, using the following equation:

$$\tan\theta = \frac{(rV^2/K)\sin\gamma\cos\gamma}{(rV^2/K)\cos^2\gamma - 1} \tag{A8}$$

Note: for

$e < 0$ & $\gamma_E < 0$

$\theta_E < 0$

for strategy A

$\theta_S \geq 0$

for strategy B

$\theta_E < \theta_S \leq 0$

o The time from deorbit point to pierce is determined by evaluating the following equation at θ_E and θ_S :

$$t(\theta) = \frac{a^{3/2}}{\sqrt{K}}\left[2\tan^{-1}\left(\sqrt{\frac{1-e}{1+e}}\tan\frac{1}{2}\theta\right) - \frac{e\sqrt{1-e^2}\sin\theta}{1+e\cos\theta}\right] \tag{A9}$$

o Total trajectory time is given by

$$t_{total} = t(\theta_E) - t(\theta_S) \tag{A10}$$

where θ_E and θ_S are given by eq. A8

$t(\theta_E)$ and $t(\theta_S)$ by eq. A9

5. Range to pierce,

o The ground range to pierce is given by

$$R = R_E\left(\theta_S - \theta_E\right) \tag{A11}$$

APPENDIX B

STRATEGY C IMPULSIVE ΔV COMPUTATION

Given: V_E = reentry velocity

γ_E = reentry flight path angle

h_E = reentry altitude

h_s = Shuttle orbit altitude

Find: ΔV_1 = impulsive ΔV for first Hohmann transfer burn

ΔV_2 = impulsive ΔV for second Hohmann transfer burn

Solve: Use Reference (18) approach

1. Compute deorbit trajectory eccentricity, e and energy, E , as shown in step 1 of Appendix A.

2. Compute apogee altitude, h_a and velocity, V_a .

$$h_a = -\frac{(e+1)K}{2E} - R_E = r_a - R_E \tag{B1}$$

$$V_a = \sqrt{2\left(E + \frac{K}{r_a}\right)} \tag{B2}$$

where K = gravitational constant

3. Compute Hohmann transfer orbit eccentricity, e_H ,

$$e_H = \frac{r_a - r_s}{r_a + r_s} \tag{B3}$$

where $r_s = h_s + R_E$

R_E = earth radius

4. Compute Hohmann transfer orbit velocity V_{H_1} and V_{H_2} ,

$$V_{H_1} = \sqrt{\frac{(e_H^2 + 1)\ K}{r_s}} \tag{B4}$$

$$V_{H_2} = \sqrt{\frac{(e_H^2 + 1)\ K}{r_a}} \tag{B5}$$

5. Compute ΔV_1 at Shuttle orbit

ΔV_2 at apogee

- Define Shuttle orbital velocity, V_o (assumed circular orbit)

$$V_o = \sqrt{K/r_s} \tag{B6}$$

- Then

$$\Delta V_1 = V_{H_1} - V_o \tag{B7}$$

$$\Delta V_2 = V_{H_2} - V_o$$

$$\Delta V = |\Delta V_1| + |\Delta V_2| \tag{B8}$$

where V_{H_1} is given by eq. B4

V_0 is given by eq. B6

V_{H_2} is given by eq. B5

V_a is given by eq. B2

6. Total deorbit time computation

o Hohmann transfer transit time is given by equation A9 of Appendix A evaluated for θ = 180 deg, i.e., half an elliptic orbit period. This gives

$$t_H = \frac{\pi a^{3/2}}{\sqrt{K}} \tag{B9}$$

where

$$a = (r_a + r_s)/2 \tag{B10}$$

o Deorbit time is given by evaluating equation A8 of Appendix A at entry to obtain θ_E

o Then solving equation A9 of Appendix A results in $t(\theta_E)$

o The total deorbit time is then

$$t = t_H + t(\theta_E) \tag{B11}$$

where t_H is given by eq. B9

$t(\theta_E)$ is given by eq. A9 of Appendix A

7. Range to pierce, R

The ground range to pierce is given by

$$R = R_E \left(\pi + |\theta_E| \right) \tag{B12}$$

APPENDIX C

METHODOLOGY FOR DETERMINING PROPELLANT AND PAYLOAD FOR DEPLOYMENT OF MULTIPLE PAYLOADS

Define:

1 $m_{B_{o_i}}$ = mass remaining at burnout after ith burn

2 n_i = ratio of inert weight of booster to propellant weight for ith burn (mass fraction)

3 P_i = mass of propellant burned for ith burn

4 m_{o_i} = miscellaneous mass other than payload and propulsion for ith stage

5 m_{D_i} = payload mass deployed after ith burn

6 ΔV_i = impulsive ΔV applied during ith burn

7 I_i = specific impulse for ith burn

8 g = reference gravitational constant = 32.174 ft/sec^2

9 δ_i = 0 if motor restart, 1 if drop away stage

Neglecting gravity terms:

$$\Delta V_i = I_i g \ \ln\left(\frac{m_{B_{o_i}} + P_i}{m_{B_{o_i}}}\right)$$

Solving for $m_{B_{o_i}}$:

$$m_{B_{o_i}} = P_i/[\exp\,(\Delta V_i/I_i g)-1] \tag{B1}$$

The burnout mass can also be expressed by

$$m_{B_{o_i}} = \underbrace{\sum_{j=1}^{N} m_{o_j}(1-\delta_j)}_{\text{misc mass remaining}} + \underbrace{\sum_{j=1}^{N} m_{D_j}}_{\text{payload mass remaining}} + \underbrace{\sum_{j=1}^{N} n_j P_j\,(1-\delta_j)}_{\text{propulsion inert mass remaining}} + \underbrace{\sum_{j=i+1}^{N} P_j}_{\text{propellant mass remaining}} \tag{B2}$$

Equations 1 and 2 constitute a set of N linear equations with N unknowns, P_i. If the propellant system is fixed, then an additional equation is required:

$$\sum_{j=1}^{N} P_j = P_o \tag{B3}$$

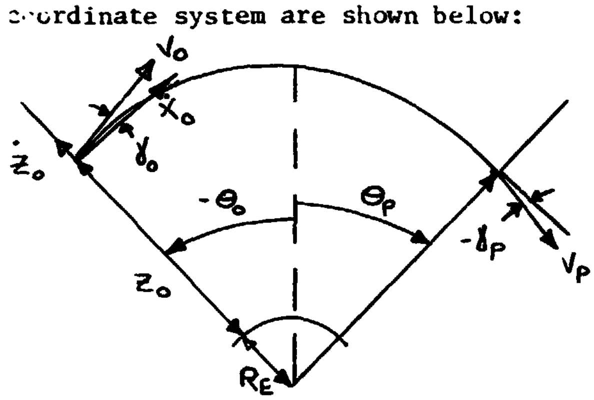
If $M_{Dj} = M_D$, i.e., all payloads are of equal weight, then Eq.B1-3 represent a system of N+1 equations in N+1 unknowns, P_i, M_D. Note that m_{oj} can be a linear function of M_D and the system of equations remains linear.

APPENDIX D

SENSITIVITY ANALYSIS

Presented is an orbital mechanics sensitivity analysis used to evaluate range sensitivity at the pierce point to perturbations in the orbit initial state vector (r_o, V_o, γ_o). Pierce point altitude was constrained to that value used in defining the unperturbed range. An appropriate set of flight mechanics equations for energy and eccentricity were used as a basis for the perturbation equations. The sensitivity of pierce point conditions are related to deorbit point altitude and velocity errors through the perturbation equations.

The geometry and coordinate system are shown below:



The defining orbital mechanics equations are (Reference 18):

o Energy

$$E = \frac{V^2}{2} - \frac{K}{r} \tag{D1}$$

which can also be expressed as

$$V = \sqrt{2\left(E + K/r\right)} \tag{D2}$$

o Eccentricity

$$e^2 = \left[\left(rV^2/K\right) - 1\right]^2 \cos^2\gamma + \sin^2\gamma \tag{D3}$$

which can also be expressed as

$$\cos^2\gamma = \left(e^2 - 1\right) \Big/ \left[\left(\frac{rV^2}{K} - 1\right)^2 - 1\right] \tag{D4}$$

o Angular Displacement

$$\tan\theta = \frac{\left(rV^2/K\right)\sin\gamma\cos\gamma}{\left(rV^2/K\right)\cos^2\gamma - 1} \tag{D5}$$

and $r_p = \text{constant}$ (D6)

The desired sensitivities: $\partial\theta/\partial\gamma_o$, $\partial\theta/\partial r_o$, $\partial\theta/\partial V_o$ which define the sensitivity of the angular displacement between launch and pierce to initial γ_o, r_o, V_o, are given by:

$$\frac{\partial \theta}{\partial \gamma_o} \qquad \frac{\partial}{\partial \gamma_o}\left[\theta_p - \theta_o\right]$$

$$\frac{\partial \theta}{\partial V_o} \qquad \frac{\partial}{\partial V_o}\left[\theta_p - \theta_o\right] \tag{D7}$$

$$\frac{\partial \theta}{\partial r_o} \qquad \frac{\partial}{\partial r_o}\left[\theta_p - \theta_o\right]$$

The terms $\frac{\partial \theta_o}{\partial \gamma_o}$, $\frac{\partial \theta_o}{\partial V_o}$, and $\frac{\partial \theta_o}{\partial r_o}$ are evaluated by straightforward differentiation of the eq. D5 with unperturbed state vector (V_o, γ_o, r_o) values used in evaluating the derivatives. This term in effect evaluates the perturbation of the apse line due to perturbed (V_o, γ_o, r_o). The terms $\frac{\partial \theta_p}{\partial \gamma_o}$, $\frac{\partial \theta_p}{\partial r_o}$, and $\frac{\partial \theta_p}{\partial V_o}$ are more difficult to obtain for they define the pierce point perturbation due to the initial state vector perturbation. The parameters e and E defined in eq. D1 and D3 are used to provide the transformation between these perturbed states at 0 and p.

From D5 $\quad \theta = \theta\,(r, V, \gamma)$

thus

$$\frac{\partial \theta_p}{\partial \gamma_o} = \frac{\partial \theta_p}{\partial r_p}\frac{\partial r_p}{\partial \gamma_o} + \frac{\partial \theta_p}{\partial V_p}\frac{\partial V_p}{\partial \gamma_o} + \frac{\partial \theta_p}{\partial \gamma_p}\frac{\partial \gamma_p}{\partial \gamma_o} \tag{D8}$$

but

$$r_p = \text{constant}, \quad \frac{\partial r_p}{\partial \gamma_o} = 0 \tag{D9}$$

and eq. D8 becomes

$$\frac{\partial \theta_p}{\partial \gamma_o} = \frac{\partial \theta_p}{\partial V_p}\frac{\partial V_p}{\partial \gamma_o} + \frac{\partial \theta_p}{\partial \gamma_p}\frac{\partial \gamma_p}{\partial \gamma_o} \tag{D10}$$

Now using the energy relationship of eq. D2, the $\partial V_p / \partial \gamma_o$ becomes

$$\frac{\partial V_p}{\partial \gamma_o} = \frac{\partial V_p}{\partial E}\frac{\partial E}{\partial \gamma_o} + \frac{\partial V_p}{\partial r_p}\frac{\partial r_p}{\partial \gamma_o} \tag{D11}$$

Note that $\quad E = E\,(V_o, r_o), \quad \frac{\partial E}{\partial \gamma_o} = 0$

also $\quad r_p = \text{constant}, \quad \frac{\partial r_p}{\partial \gamma_o} = 0$

therefore

$$\frac{\partial V_p}{\partial \gamma_o} = 0 \tag{D12}$$

The derivative $\frac{\partial \gamma_p}{\partial \gamma_o}$ of eq. D10 can be evaluated using eq. D4 to give

$$\frac{\partial \gamma_p}{\partial \gamma_o} = \frac{\partial \gamma_p}{\partial e}\frac{\partial e}{\partial \gamma_o} + \frac{\partial \gamma_p}{\partial V_p}\frac{\partial V_p}{\partial \gamma_o} + \frac{\partial \gamma_p}{\partial r_p}\frac{\partial r_p}{\partial \gamma_o} \tag{D13}$$

Again note that $\quad \frac{\partial r_p}{\partial \gamma_o} = 0$

and $\partial V_p / \partial \gamma_o = 0$ from eq. D12

Therefore,

$$\frac{\partial \gamma_p}{\partial \gamma_o} = \frac{\partial \gamma_p}{\partial e} \frac{\partial e}{\partial \gamma_o} \tag{D14}$$

Note that both derivatives in eq. D14 can be calculated from eq. D3 where

$$e = e(r, V, \gamma)$$

Equation D10 can be rewritten in the form

$$\boxed{\frac{\partial \theta_p}{\partial \gamma_o} = \frac{\partial \theta_p}{\partial \gamma_p} \frac{\partial \gamma_p}{\partial e} \frac{\partial e}{\partial \gamma_o}} \tag{D15}$$

The three derivatives are readily calculated using eq. D5, D4 and D3 respectively.

The next derivative of interest from eq. D7 is the $\partial \theta_p / \partial V_o$

From eq. D5,

$$\frac{\partial \theta_p}{\partial V_o} = \frac{\partial \theta_p}{\partial V_p} \frac{\partial V_p}{\partial V_o} + \frac{\partial \theta_p}{\partial \gamma_p} \frac{\partial \gamma_p}{\partial V_o} + \frac{\partial \theta_p}{\partial r_p} \cancelto{0}{\frac{\partial r_p}{\partial V_o}} \tag{D16}$$

The $\frac{\partial V_p}{\partial V_o}$ is given by differentiating eq. D2 to give

$$\frac{\partial V_p}{\partial V_o} = \frac{\partial V_p}{\partial E} \frac{\partial E}{\partial V_o} + \frac{\partial V_p}{\partial r_p} \cancelto{0}{\frac{\partial r_p}{\partial V_o}} \tag{D17}$$

The $\frac{\partial \gamma_p}{\partial V_o}$ is given by differentiating eq. D4 to give

$$\frac{\partial \gamma_p}{\partial V_o} = \frac{\partial \gamma_p}{\partial V_p} \frac{\partial V_p}{\partial V_o} + \frac{\partial \gamma_p}{\partial e} \frac{\partial e}{\partial V_o} + \frac{\partial \gamma_p}{\partial r_p} \cancelto{0}{\frac{\partial r_p}{\partial V_o}} \tag{D18}$$

Combining D17 and D18 with D16 gives

$$\boxed{\frac{\partial \theta_p}{\partial V_o} = \frac{\partial \theta_p}{\partial V_p} \left[\frac{\partial V_p}{\partial E} \frac{\partial E}{\partial V_o} \right] + \frac{\partial \theta_p}{\partial \gamma_p} \left[\frac{\partial \gamma_p}{\partial V_p} \frac{\partial V_p}{\partial V_o} + \frac{\partial \gamma_p}{\partial e} \frac{\partial e}{\partial V_o} \right]} \tag{D19}$$

where all derivatives can be evaluated from eq. D2, D3, D4 and D5. Finally, the derivative $\partial \theta_p / \partial r_o$ of eq. D7 can be evaluated using eq. D5 to give

$$\frac{\partial \theta_p}{\partial r_o} = \frac{\partial \theta_p}{\partial V_p} \frac{\partial V_p}{\partial r_o} + \frac{\partial \theta_p}{\partial \gamma_p} \frac{\partial \gamma_p}{\partial r_o} + \frac{\partial \theta_p}{\partial r_p} \cancelto{0}{\frac{\partial r_p}{\partial r_o}} \tag{D20}$$

Looking at $\frac{\partial V_p}{\partial r_o}$ from eq. D1

$$\frac{\partial V_p}{\partial r_o} = \frac{\partial V_p}{\partial E} \frac{\partial E}{\partial r_o} + \frac{\partial V_p}{\partial r_p} \cancelto{0}{\frac{\partial r_p}{\partial r_o}} = \frac{\partial V_p}{\partial E} \frac{\partial E}{\partial r_o} \tag{D21}$$

Looking at $\frac{\partial \gamma_p}{\partial r_o}$ from eq. D4

$$\frac{\partial \gamma_p}{\partial r_o} = \frac{\partial \gamma_p}{\partial e} \frac{\partial e}{\partial r_o} + \frac{\partial \gamma_p}{\partial V_p} \frac{\partial V_p}{\partial r_o} + \frac{\partial \gamma_p}{\partial r_p} \cancelto{0}{\frac{\partial r_p}{\partial r_o}} \tag{D22}$$

where $\frac{\partial e}{\partial r_o}$ is evaluated from D3 and $\frac{\partial V_p}{\partial r_o}$ from D21.

Combining eq. D21 and D22 with D20 gives

$$\boxed{\frac{\partial \theta_p}{\partial r_o} = \frac{\partial \theta_p}{\partial r_p}\left[\frac{\partial V_p}{\partial E}\frac{\partial E}{\partial V_o}\right] + \frac{\partial \theta}{\partial \gamma_p}\left[\frac{\partial \gamma_p}{\partial e}\frac{\partial e}{\partial r_o} + \frac{\partial \gamma_p}{\partial V_p}\frac{\partial V_p}{\partial r_o}\right]} \tag{D23}$$

The partial derivatives necessary for calculating $\frac{\partial \theta_p}{\partial \gamma_o}$, $\frac{\partial \theta_p}{\partial V_o}$, $\frac{\partial \theta_p}{\partial r_o}$ are given in eq. D15, D19 and D23. The derivatives for $\frac{\partial \theta_o}{\partial r_o}$, $\frac{\partial \theta_o}{\partial V_o}$, $\frac{\partial \theta_o}{\partial \gamma_o}$ are given by differentation of eq. D5.

Differentiation of eq. D5 yields

$$\frac{\partial \theta_o}{\partial \gamma_o} = \frac{r_o V_o^2}{K}\left\{\frac{\cos 2\gamma + 2\tan\theta_o \sin\gamma_o \cos\gamma_o}{\left[\left(1+\tan^2\theta_o\right)\left(\frac{r_o V_o^2}{K}\cos^2\gamma_o - 1\right)\right]}\right\} \tag{D24}$$

$$\frac{\partial \theta_o}{\partial V_o} = \frac{2 r_o V_o^2}{K} \frac{\sin\gamma_o \cos\gamma_o \left\{1 - \left[\frac{r_o V_o^2}{K}\cos^2\gamma_o \Big/ \left(\frac{r_o V_o^2}{K}\cos^2\gamma - 1\right)\right]\right\}}{\left[\left(\frac{r_o V_o^2}{K}\cos^2\gamma_o - 1\right)\left(1+\tan^2\theta_o\right) V_o\right]} \tag{D25}$$

$$\frac{\partial \theta_o}{\partial r_o} = \frac{\partial \theta_o}{\partial V_o}\frac{V_o}{2 r_o} \tag{D26}$$

The derivatives required to solve eqs. D15, D19 and D23 are from differentiation of eq. D1

$$\frac{\partial E}{\partial V_o} = V_o \tag{D27}$$

$$\frac{\partial E}{\partial r_o} = K/r_o^2 \tag{D28}$$

from differentiation of eq. D3

$$\frac{\partial e}{\partial V_o} = \frac{2 r_o V_o^2}{K}\left(\frac{r_o V_o^2}{K} - 1\right)\frac{\cos^2\gamma_o}{E V_o} \tag{D29}$$

$$\frac{\partial e}{\partial r_o} = \frac{\partial e}{\partial V_o}\frac{V_o}{2 r_o} \tag{D30}$$

$$\frac{\partial e}{\partial \gamma_o} = \frac{\sin\gamma_o \cos\gamma_o}{E}\left[1 - \left(\frac{r_o V_o^2}{K} - 1\right)^2\right] \tag{D31}$$

from differentiation of eq. D4

$$\frac{\partial \gamma_p}{\partial V_p} = \frac{2 r_p V_p^2}{K}\left(\frac{r_p V_p^2}{K} - 1\right) \Big/ \left\{V_p\left[\left(\frac{r_p V_p^2}{K} - 1\right)^2 - 1\right]\tan\gamma_p\right\} \tag{D32}$$

Equation D24 through D32 are evaluated at either the initial point, o, or pierce point, p, to provide the derivatives required in eq. D7. Two other derivatives required are in eq. D19

$$\frac{\partial V_P}{\partial V_o} = \frac{V_o}{V_P} \tag{D33}$$

and in eq. D23

$$\frac{\partial V_P}{\partial r_o} = \frac{K}{V_P\, r_o^2} \tag{D34}$$

which are obtained by differentiating eq. D1.

The derivatives given by eq. D7, i.e.,

$$\partial\theta/\partial\gamma_o \qquad \partial\theta/\partial r_o \qquad \partial\theta/\partial V_o$$

can now be evaluated by substituting eqs. D24 through D34 into eqs. D7, D15, D19 and D23. The resulting sensitivity derivatives can be related to velocity and position errors at the deorbit point as follows:

$$\begin{aligned}
&\partial\theta/\partial x_o = \partial\theta/\partial y_o = 0\ , \quad \frac{\partial\theta}{\partial z_o} = \frac{\partial\theta}{\partial r_o}\frac{\partial r_o}{\partial z_o} \\
&\frac{\partial\theta}{\partial \dot{x}_o} = \frac{\partial\theta}{\partial V_o}\frac{\partial V_o}{\partial \dot{x}_o} + \frac{\partial\theta}{\partial \gamma_o}\frac{\partial \gamma_o}{\partial \dot{x}_o} \\
&\frac{\partial\theta}{\partial \dot{y}_o} = \frac{\partial\theta}{\partial V_o}\frac{\partial V_o}{\partial \dot{y}_o} + \frac{\partial\theta}{\partial \gamma_o}\frac{\partial \gamma_o}{\partial \dot{y}_o} \\
&\frac{\partial\theta}{\partial \dot{z}_o} = \frac{\partial\theta}{\partial V_o}\frac{\partial V_o}{\partial \dot{z}_o} + \frac{\partial\theta}{\partial \gamma_o}\frac{\partial \gamma_o}{\partial \dot{z}_o}
\end{aligned} \tag{D35}$$

where X_o, Y_o, Z_o are the initial position coordinates and

$\dot{X}_o$, $\dot{Y}_o$, $\dot{Z}_o$ are the initial velocity components.

The derivatives $\frac{\partial r_o}{\partial z_o}$, $\frac{\partial V_o}{\partial \dot{x}_o}$, $\frac{\partial V_o}{\partial \dot{y}_o}$, $\frac{\partial V_o}{\partial \dot{z}_o}$, etc. are given by differentiating the following equations:

1. $r_o = R_E + z_o$ (D36)

$$\boxed{\partial r_o/\partial z_o = 1}$$

2. $V_o^2 = \dot{x}_o^2 + \dot{z}_o^2$ (D37)

$$\boxed{\frac{\partial V_o}{\partial \dot{x}_o} = \frac{\dot{x}_o}{V_o} = \cos\gamma_o}$$

3. $\tan\gamma_o = \dot{z}_o/\dot{x}_o$ (D38)

$$\boxed{\frac{\partial \gamma_o}{\partial \dot{x}_o} = \frac{-\sin\gamma_o}{[1+\cos^2\gamma_o]V_o}}$$

4. $V_o^2 = \dot{x}_o^2 + \dot{z}_o^2$ (D39)

$$\boxed{\frac{\partial V_o}{\partial \dot{y}_o} = 0}$$

5. $\tan \gamma_o = \dfrac{\dot{Z}_o}{\dot{X}_o}$

$$\boxed{\frac{\partial \gamma_o}{\partial \dot{Y}_o} = 0} \qquad \text{(D40)}$$

6. $V_o^2 = \dot{X}_o^2 + \dot{Z}_o^2$

$$\boxed{\frac{\partial V_o}{\partial \dot{Z}_o} = \frac{\dot{Z}_o}{V_o} = \sin \gamma_o} \qquad \text{(D41)}$$

7. $\tan \gamma_o = \dfrac{\dot{Z}_o}{\dot{X}_o}$

$$\boxed{\frac{\partial \gamma_o}{\partial \dot{Z}_o} = \frac{\cos^2 \gamma_o}{[1 + \cos^2 \gamma_o]\, V \sin \gamma_o}} \qquad \text{(D42)}$$

By substituting eq. D36 through D42 into the eqs. of D35 with the other derivatives evaluated through eq. D7 the sensitivity of pierce point angular displacement to initial position and velocity errors can be determined.