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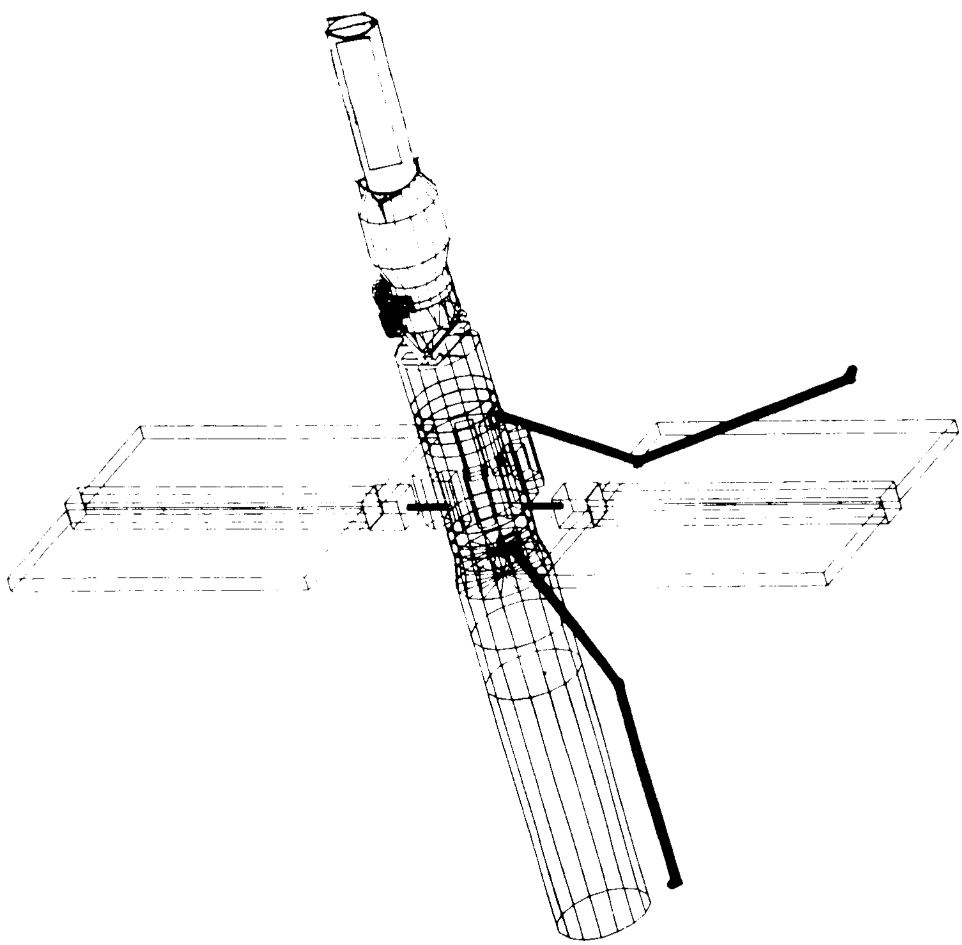
CENTAUR OPERATIONS AT THE SPACE STATION COST AND TRANSPORTATION ANALYSIS



GENERAL DYNAMICS
Space Systems Division

CONTRACT NO. NAS3-24900
REPORT NO. CR-182128
GDSS-SP-88-006

CENTAUR OPERATIONS AT THE SPACE STATION COST AND TRANSPORTATION ANALYSIS

FINAL REPORT

10 AUGUST 1988

Prepared for
NASA - Lewis Research Center
Cleveland, Ohio

Prepared by
GENERAL DYNAMICS SPACE SYSTEMS DIVISION
San Diego, California

FOREWORD/COST DISCLAIMER

The cost estimates herein are for planning and comparison purposes only and do not constitute a commitment on the part of General Dynamics.

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ACKNOWLEDGMENTS

Without the help of R. Sedlund, J. Che, D. Chiarrapa, C. Lowry, C. Phillips, C. Huynh, and J. Maloney, this analysis could not have been performed.

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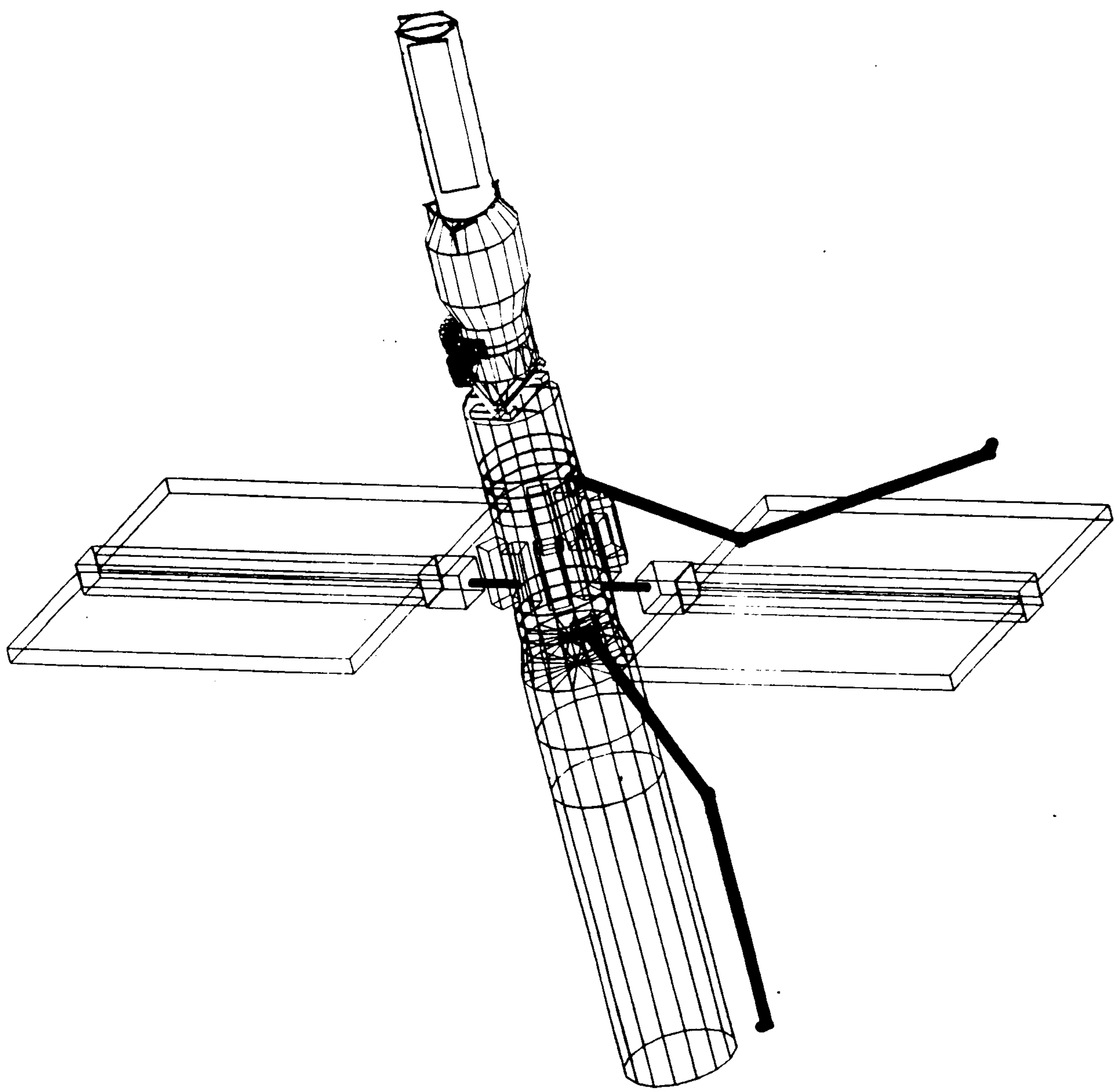
SECTION 1
SUMMARY



A study was conducted to expand on the analysis of Technology Demonstration Mission (TDM) concepts generated in 1986 by NASA Report No. CR-179593, "Centaur Operations at the Space Station." TDMs are experiments and exercises that would utilize the General Dynamics Space Systems Division (GDSS) Centaur G-Prime upper stage to advance technologies required for Space Transfer Vehicle (STV) accommodations and operations at the Space Station. The current study begun in 1987 performed an initial evaluation of the cost to NASA for TDM implementation and termination. It also analyzed the potential for creating a commercial COMmunication SATellite (COMSAT) launch program utilizing Centaur and the TDM hardware.

Titan/Centaur is the only planned operational version of the Centaur G-Prime upper stage. The study added modifications to evolve it to a space-based Titan/Centaur (SBTC) for use in analyses.

Major study results were as follows:

a. The payload capability of SBTC from the Space Station to geosynchronous orbit was nearly double what is currently (1987) predicted for ground-based Titan/Centaur launching.

b. Commercial satellite launches from the Space Station exhibited a cost equivalence, or in some cases, a cost advantage over current ground launching when used in a "topping off" mode for a ground-launched SBTC.

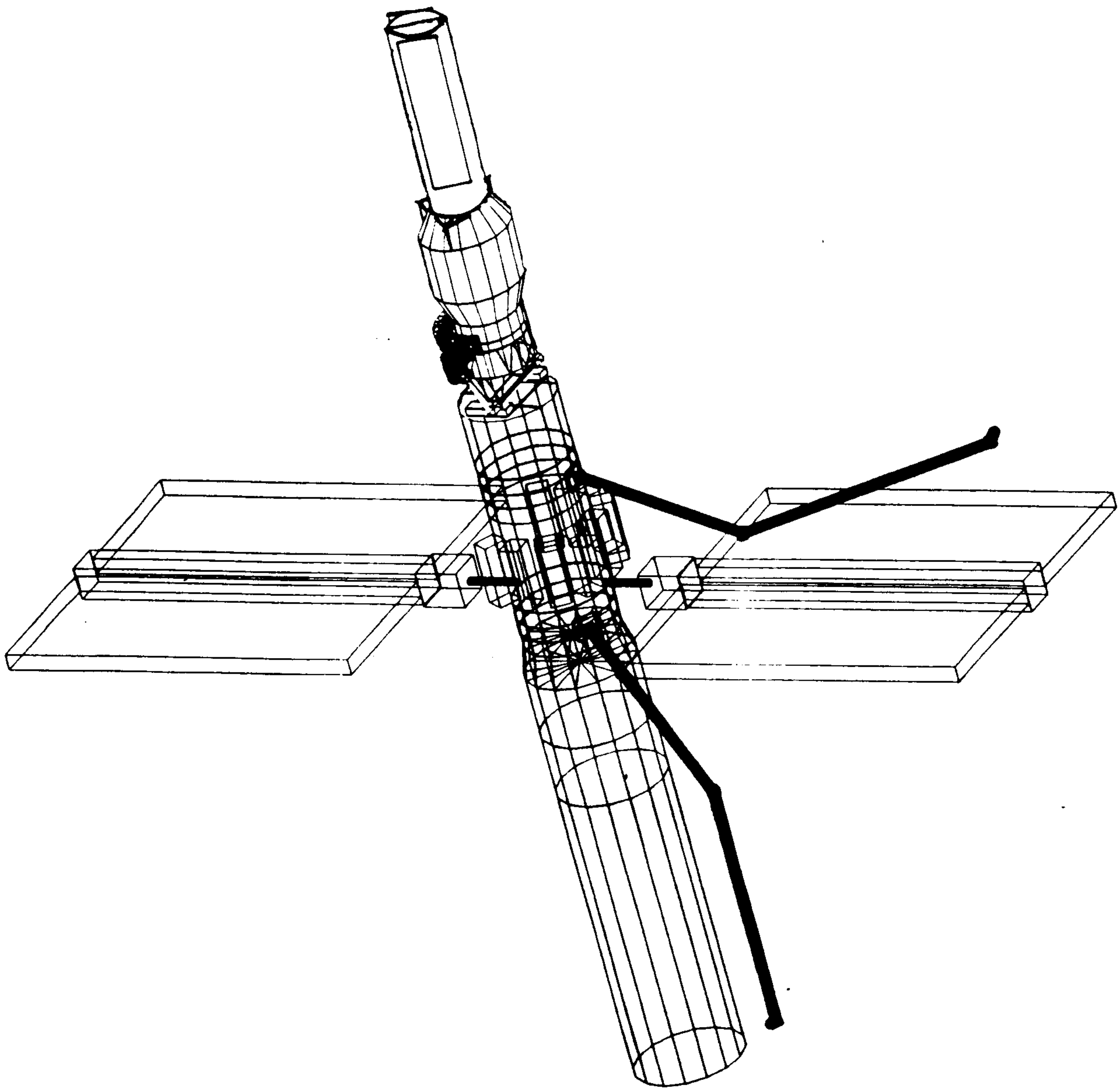
c. The "topping off" mode appears to be most advantageous when it is an "enabling" component for scenarios deploying multiple heavy payloads.

d. Overall costs for SBTC TDMs, utilizing operational hardware and a reuse design philosophy, was comparable to planned STV technology development, which utilizes dummy pieces of structure and tankage.

e. The SBTC TDMs offer significantly higher STV fidelity than currently planned accommodations technology development. This includes an actual payload launch after TDM completion, which is not a feature of STV dummy demonstrations.

It was concluded that SBTC TDMs would be valuable to NASA because they provide more realistic and cost-effective simulations for technology development than current planning for about the same cost. It was also concluded that an SBTC augment (topping off) COMSAT launch program would be valuable to NASA since it produces a definite cost and capability advantage for heavy multiple payload launches.

SECTION 2
INTRODUCTION



Centaur Operations at the Space Station (COSS) Study was performed for NASA/Lewis Research Center (NASA/LeRC). It had two parts: Phase I and Phase II, using the same Contract No. NAS3-24900. Both present predesign concepts for new programs. Phase I would pave the way for STV at the Space Station. It developed missions to demonstrate the technology to store, maintain, and launch STVs from the Space Station. At program completion, remaining assets are assimilated into Space Station and STV development. Phase II conducted cost and transportation analysis. Specifically, it determines the cost and value to NASA of Phase I. Additionally, it postulates the outcome if Phase I assets were not assimilated, but instead became the basis of a Space Station based expendable launch program. The launch program architecture is established. Its capabilities and costs are then compared to an equivalent ground based launch program.

2.1 BACKGROUND

The COSS study began in September 1986. Phase I work was completed in February 1987, and results are in NASA Final Report No. CR179593 (GDSS-SP-87-003). NASA/LeRC then allowed GDSS to perform follow-on work in a second phase beginning 1 September 1987. Phase II added two additional tasks to the COSS contract and was completed on approximately 3 June 1988.

2.1.1 THE PHASE I STUDY. The goal of COSS Phase I (COSS I) was to pave the way for STV at the Space Station using the Space Transportation System/Centaur (STS/Centaur) upper stage rocket. To accomplish this goal, COSS I had two objectives: first, to predesign these Technology Demonstration Missions (TDMs) to demonstrate the technology to store, maintain, and launch STVs from the Space Station, and second to document Space Station structural or software scarring required by TDMs into the official Space Station data base.

Two TDMs were predesigned which defined five experiments and exercises. Figure 2-1 shows that the Accommodations TDM would demonstrate STV berthing, vehicle checkout/maintenance/servicing, and payload integration tasks at the Space Station. This would take place in a Space Station hangar especially designed for the TDM. Figure 2-2 shows that the Operations TDM would be performed on a Co-Orbiting Platform (COP) designed for the TDMs. The COP would be positioned in the same orbit as the Space Station, but 100 n.mi. in front of it. The Operations TDM would demonstrate cryogenic propellant fill/drain, and launch an actual COMSAT mission, as illustrated in Figure 2-3.

TDM tasks would be repeatedly executed for 9 months to gain experience, and to perform evaluations and modifications. The COSS I program would then end with an actual Centaur launch from the COP, deploying one or more real, but unspecified, payloads. Centaur would not be recovered. TDM hardware, including the COP, would be assimilated into Space Station accommodations, and into an off-station STV servicing platform to be subsequently constructed. This ending would optimize the cost effectiveness of COSS I resources. It may also provide some return on program investment from payload customer revenues.

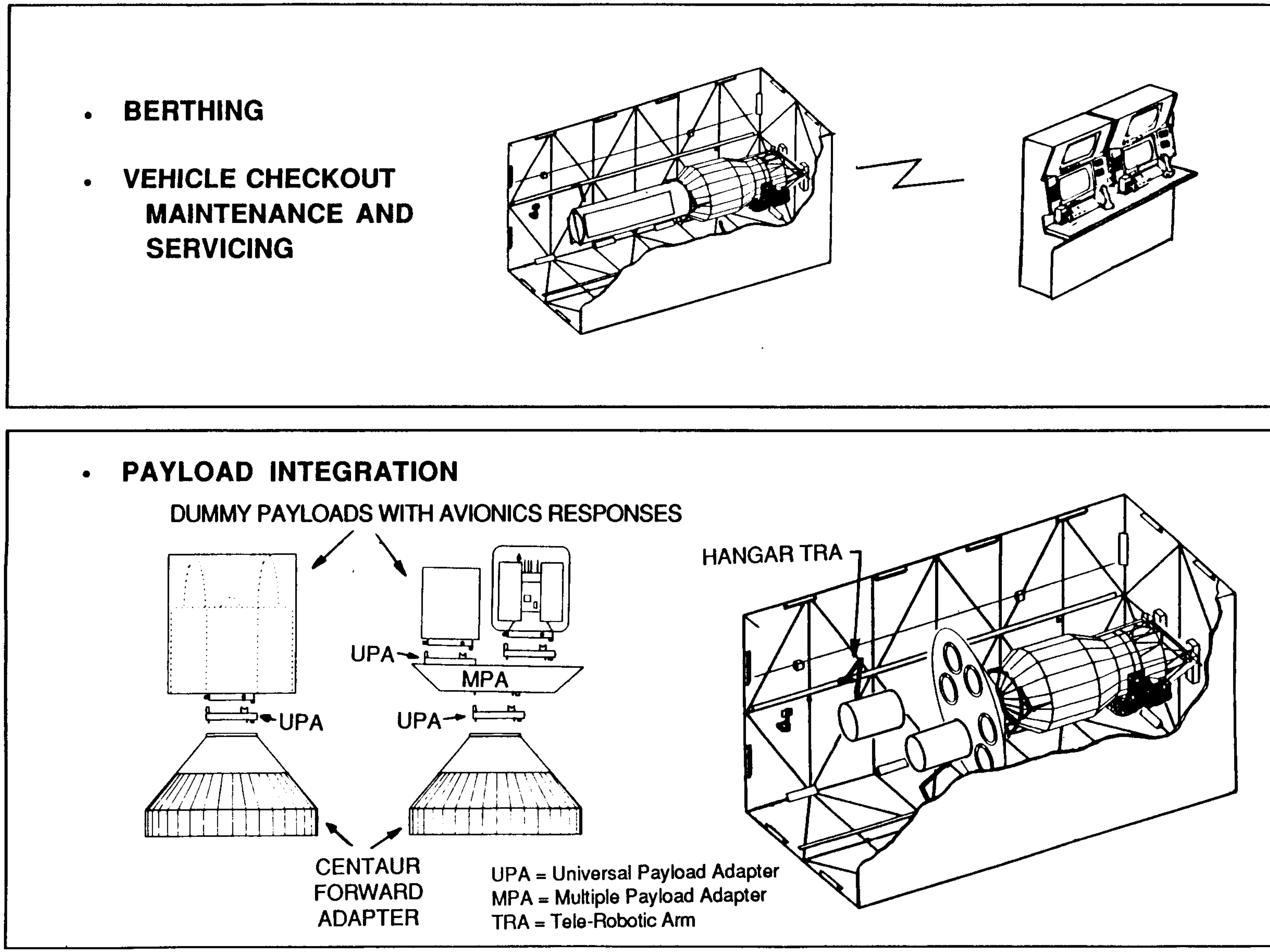
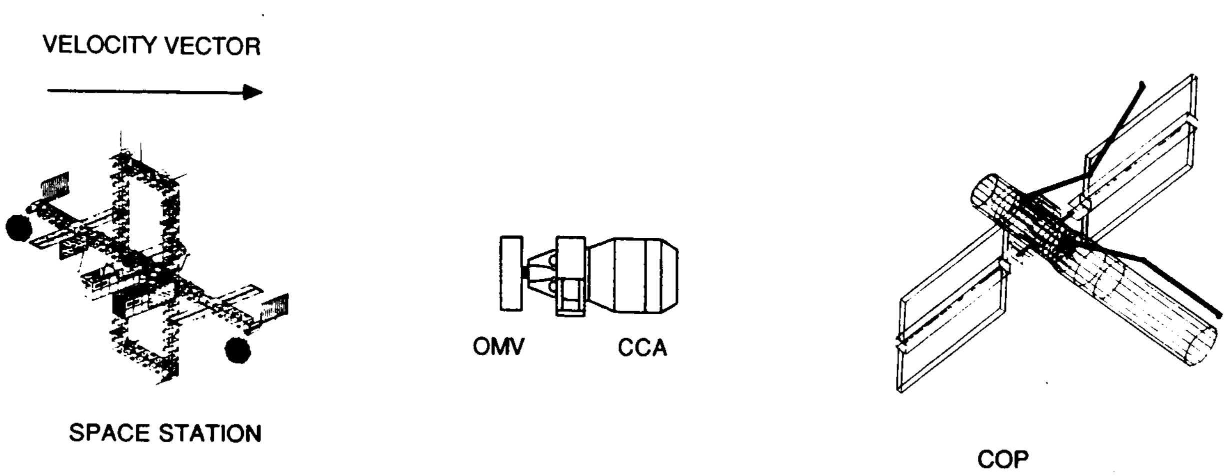


Figure 2-1. COSS Accommodations TDM Will Develop Three Technologies



NOTE:

CCA = CENTAUR / CISS ASSEMBLY
CISS = CENTAUR INTEGRATED SUPPORT STRUCTURE
OMV = ORBITAL MANEUVERING VEHICLE
COP = CO-ORBITING PLATFORM

Figure 2-2. Operations TDM Done at COP, 100 n.mi. in Front of Station

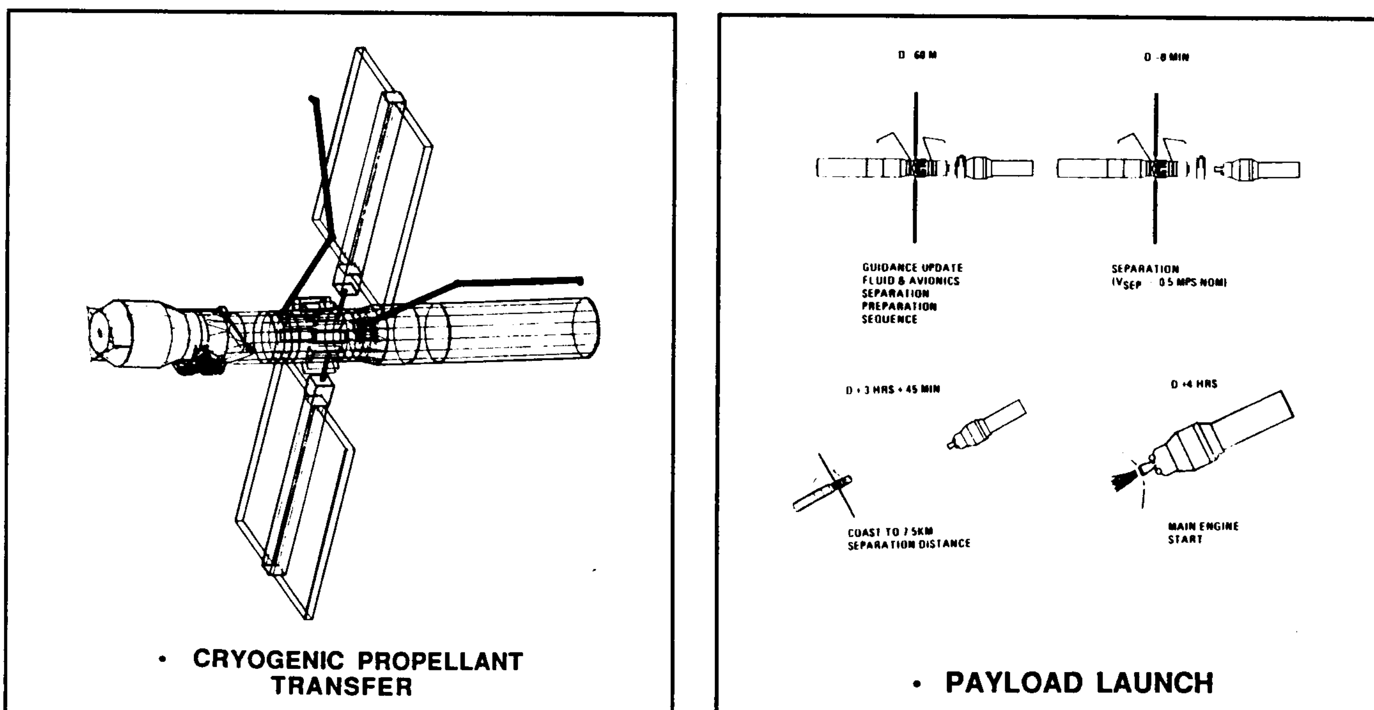


Figure 2-3. The Operations TDM Will Develop Two Technologies

The launch aspect of COSS TDMs drew particular attention. This was because preliminary calculations indicated that the STS/Centaur payload capability from the Space Station far exceeded what it could perform as an upper stage to a ground-based booster or the Shuttle. NASA/LeRC wanted to know: 1) whether the benefits to STV development resulting from COSS TDMs was worth their cost, and 2) would the additional payload capability of a Centaur deployment from Space Station justify a Space Station based expendable space transportation program for launching commercial COMSATs. This prompted NASA/LeRC to fund the current Phase II study.

2.2 PHASE II OBJECTIVES

COSS II objectives were to: 1) define the operations required to launch commercial COMSATs using expendable Centaur, 2) determine the cost effectiveness of such a space transportation program, and 3) compare the costs, advantages, and disadvantages of COSS TDMs and similar TDMs that are part of current STV program managed by Marshall Space Flight Center (MSFC).

2.3 SCOPE

The scope of analyses for defining operations and cost effectiveness of the space-based expendable launch program was limited to the span of years 1998 and 2002. Additional criteria were as follows:

- Logistics by both current and heavy launch vehicles was allowed
- DOD payloads were excluded except for GPS
- COMSAT launch cost effectiveness was determined by comparing equivalent space and ground launch costs

The scope for analyzing costs, advantages/disadvantages of COSS and STV TDMs was limited to in-space operations. It was taken that:

- COSS TDM operations begin with the first space element arrival, the hangar, and end with resource re-allocation to STV
- STV TDM operations begin with the first space element arrival, the STV simulator structure, and end with the conclusion of the propellant transfer TDM
- No precursor ground development is costed

Where they did not exist, details and costs of the STV test plan were created or estimated by our study. Results were approved by the GDSS OTV Turnaround Study manager (contract NAS8-36924 DR-3), and reviewed by NASA/MSFC to ensure their accuracy.

2.4 APPROACH

The first step of our approach was to replace the STS/Centaur vehicle with a Space-Based Titan/Centaur (SBTC) for TDMs and launch operations. A "quick-look" in Phase I determined that the COSS vehicle should be changed from a STS/Centaur taken out of a 12-yr storage, to a 1997 production Titan/Centaur. This would avoid the reliability and obsolescence questions of long-term storage of the only two STS/Centaurs ever to be made.

Our approach to evaluating SBTC commercial COMSAT launches is illustrated in Figure 2-4. Its major elements are to:

- Quantify SBTC payload performance from Space Station deployment
- Determine payload mission model commensurate with SBTC capabilities
- Construct a manifest of reasonable payload recommendations
- Develop vehicles, payloads, and propellant supply logistics
- Compare the total costs for space versus ground launch of the manifest
- Examine other benefits, advantages, and disadvantages for COSS II
- Formulate conclusions and recommendations for SBTC commercial use

Our approach for TDM program cost analysis is also illustrated in Figure 2-4. Its major elements are to:

- Conduct STV cost and data research
- Create Work Breakdown Structures (WBSs) for COSS and STV TDMs
- Create test plans based on WBSs
- Develop appropriate cost models and generate costs at the WBS level
- Examine other benefits, advantages, and disadvantages for COSS TDMs
- Formulate conclusions and recommendations for the COSS TDM program

To compactly describe TDM program operations for the COSS program, a 20-min color video animation was produced as part of the study contract. It starts with SBTC delivery for the accommodations TDM, and ends with launch demonstration in the operations TDM.

To implement our approaches, two tasks were added to the contract. The first, Task 5, provided for analysis of Centaur performance boundaries, mission models, TDM modifications, and other analyses leading to the commercial launch concept. Task 5 also supported the transition from STS/Centaur to SBTC, and the production of the video animation. Task 6 supported value determinations. It allowed for WBS, test plans, and detailed cost models necessary for program cost analysis of both the TDM and space-based expendable launch program concepts.

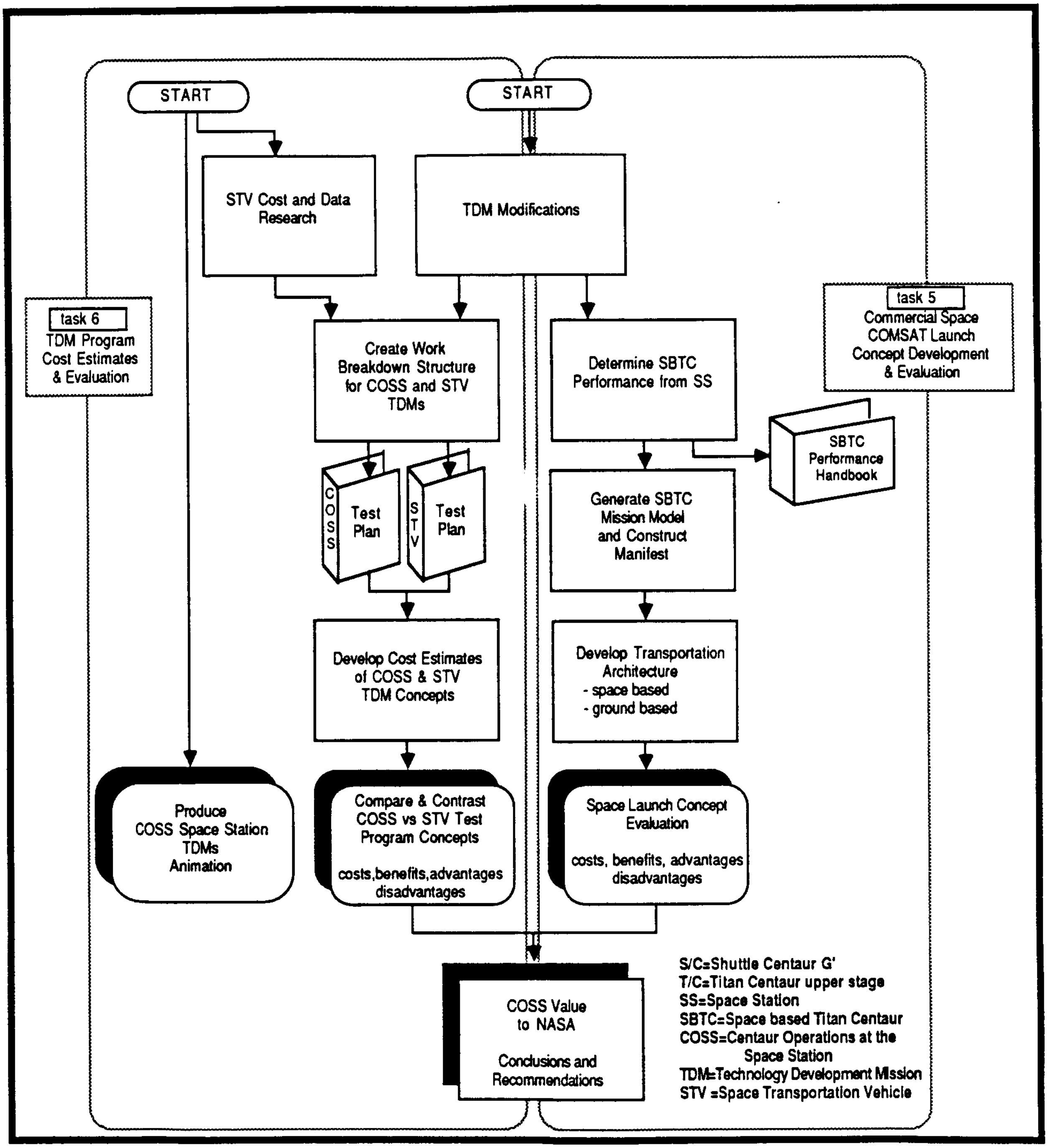


Figure 2-4. Task 5 Constructed and Evaluated COMSAT Space Launch, while Task 6 Costed and Evaluated the TDM Program

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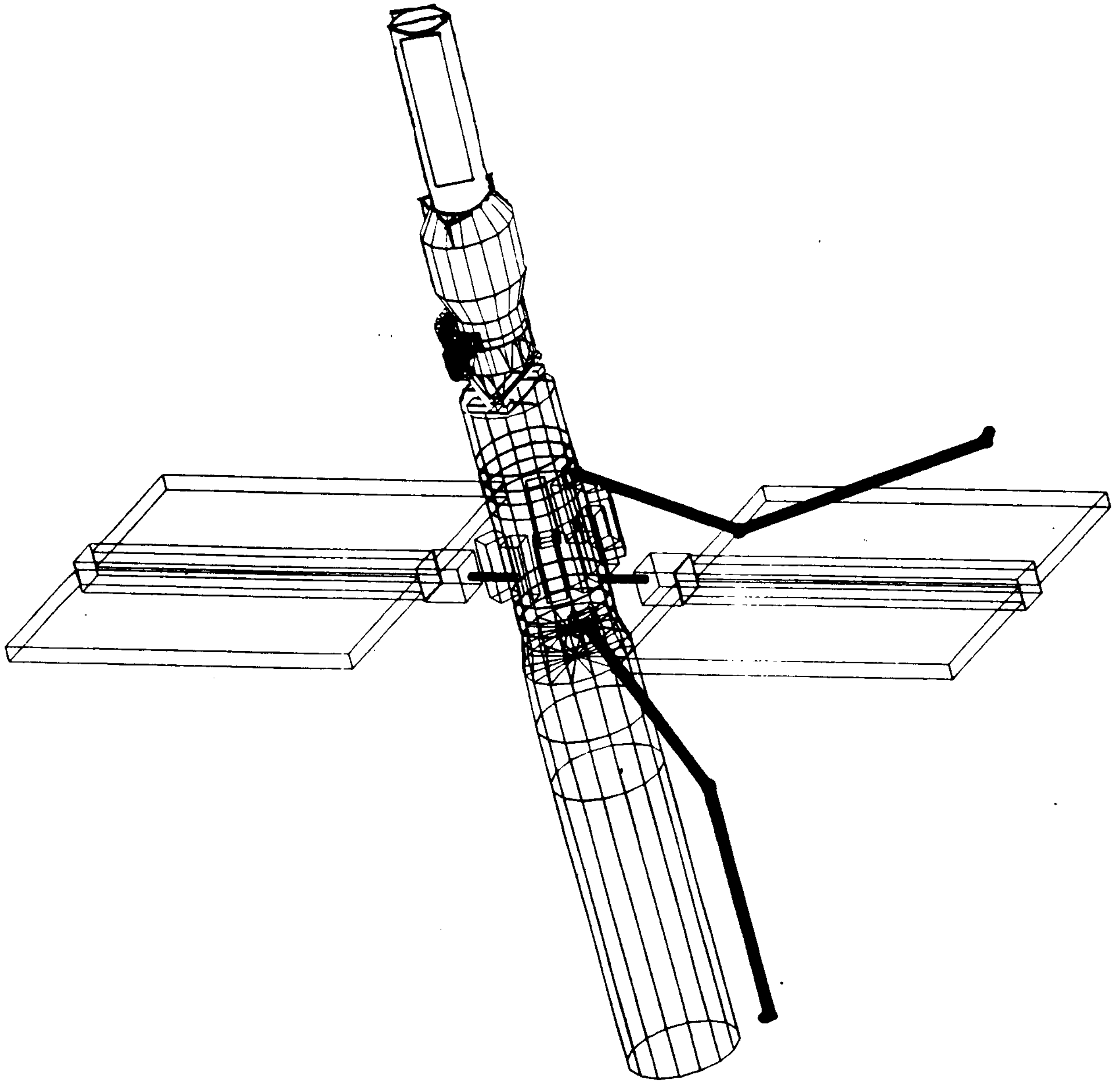
SECTION 3
TASK 5 — SPACE OPERATIONS FOR COMMERCIAL APPLICATIONS

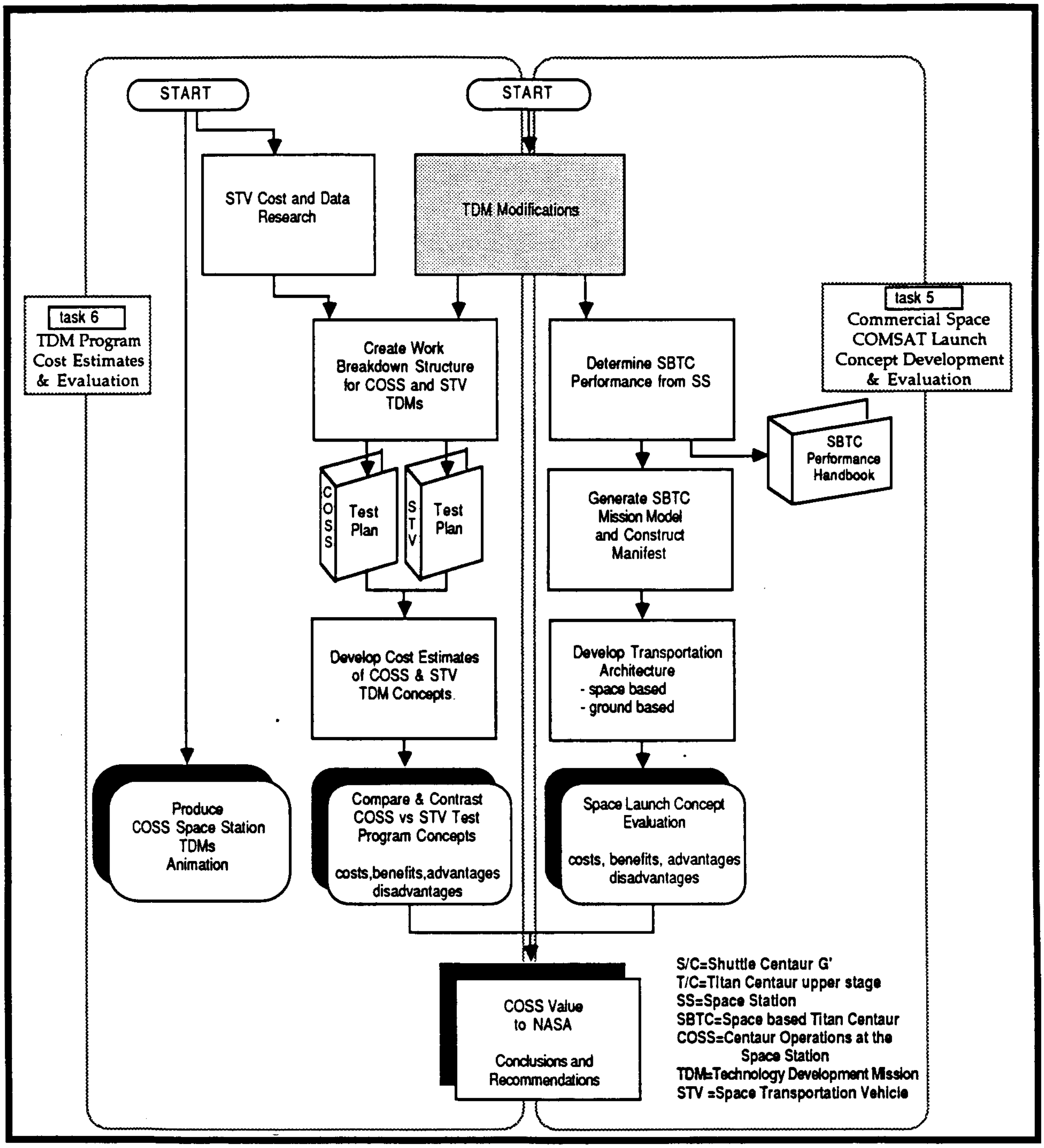


This section defines a space-based COMSAT launch concept. Its cost effectiveness is then evaluated against ground-based COMSAT transportation systems.

The launch experiment of COSS TDMs drew attention. This was because preliminary calculations indicated that Centaur G-Prime payload capability from the Space Station far exceeded what it could perform as an upper stage to a ground-based booster or the Shuttle. NASA/LeRC wanted to know whether this extra SBTC capability, and the TDM assets which could already be in place, would justify a Space Station based expendable transportation program for launching COMSATs. GDSS was contracted to define and evaluate this transportation concept.

The first steps in defining the concept were to analyze required changes and updates to COSS TDMs. Next the performance data base for SBTC was expanded, to include parametric information on dual payload launches, and limited point data on three and four payload launches. We then constructed a rough mission model and used the performance data to construct a sample manifest. We compared the costs of launching the sample manifest with: 1) the space-based transportation concept, and 2) with ground launch systems. Conclusions, recommendations, and suggestions for optimization were made based on study data.

3.1 TDM MODIFICATIONS



3.1 TDM MODIFICATIONS

TDMs are exercises and drills to develop and demonstrate the technology to store, maintain, and launch STVs from the Space Station. The follow-on COSS (COSS II) TDMs use a Titan/Centaur (T/C) vehicle modified into a Space-Based Titan/Centaur (SBTC) configuration as an STV simulator.

3.1.1 SPACE-BASING TITAN/CENTAUR. Initial intent was to use an STS/Centaur as its space-based TDM test bed. Storage costs and concerns for obsolescence motivated a shift in baseline to the Titan/Centaur (T/C) upper stage. The switch to T/C necessitated the addition of 1699 lb of additional hardware to space base the vehicle. The changes are summarized in Table 3-1. This new vehicle also becomes the baseline in postulating a commercial space launch operations program.

Table 3-1. Changes to the Titan/Centaur Upper Stage for Space Basing Will Add 1699 lb

Space Based Titan/Centaur Weight Summary		
Titan/Centaur Dry Weight (with full RCS & GHe)	3055 kgs	(6720 lbs)
Δ Modified Forward Support Structure	323 kgs	(711 lbs)
Δ Modified Aft Adapter	-26 kgs	(-58 lbs)
Δ Modified Fluid, Electrical Lines & Interfaces	375 kgs	(826 lbs)
Add Liquid Acquisition Devices (both tanks)	93 kgs	(205 lbs)
Add O-g Mass Guages (both tanks)	2 kgs	(5 lbs)
Add Jet Pulse Mixer	5 kgs	(10 lbs)
TOTAL SPACE-BASED TITAN/CENTAUR	3827 kgs	(8419 lbs)

Δ Weight = Element weight added - T/C element weight removed

The T/C upper stage is normally launched atop the Titan IV booster vehicle, and fits within the 200 in. diameter payload fairing as shown in Figure 3-1. T/C must therefore be modified for Space Station basing. Modifications are driven by the need to transport the T/C to the Space Station in the Orbiter cargo bay, the requirement to fill/drain propellants while docked in a zero-gravity environment, and the need to interface with support equipment at the Space Station. These problems had been solved for STS/Centaur by allowing it to remain attached to its already constructed Centaur Integrated Support System (CISS). Rather than design a new structure, it was decided to reroute T/C plumbing and cables to fit the STS/Centaur CISS.

3.1.1.1 Structural Modifications. The T/C is attached to its launch vehicle at its aft end using a 25.5 in. long metallic cylindrical adapter. Forward attachment is with six tangentially mounted support struts which tie the forward adapter to the payload fairing as shown in Figure 3-2. Since the SBTC will be transported to the station via the Orbiter, a different supporting structure is required. The selected method is to utilize the STS/Centaur CISS to support the SBTC both while in the Orbiter and while at the Space Station. This will require that the T/C aft adapter be replaced with a CISS-compatible STS/Centaur 35.6 cm (14 in. thick) aft adapter to support the rear of the vehicle, and

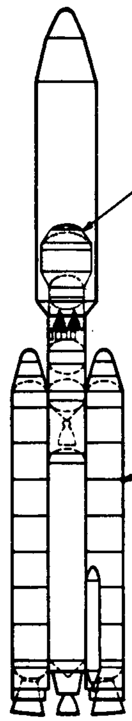


Figure 3-1. The Standard Titan/Centaur Upper Stage Vehicle Is Delivered to Orbit by the Titan IV Booster

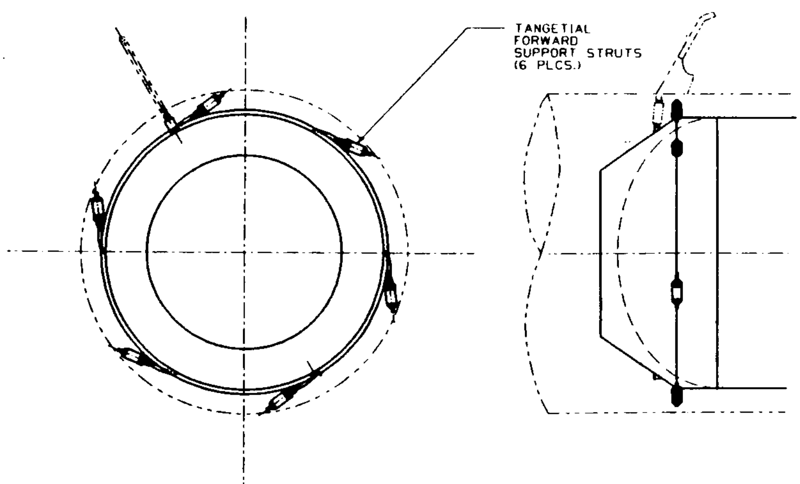


Figure 3-2. The Standard Titan/Centaur Support Structure is Made for the Titan Fairing

provide a separation interface for mission deployment. Because both aft adapters are 120 in. diameter, the substitution is not considered a major change.

Figure 3-3 shows the new forward support configuration. At the forward end of the SBTC, the tangential struts (forward bearing reactors) must be replaced with STS/Centaur trunnion and keel support structures to allow mating with Orbiter cargo bay attachment fittings. The addition of the trunnions requires some modifications to the equipment module frustrum - attach fittings for the tangential struts are removed and fittings compatible with the trunnions must be added.

3.1.1.2 Fluid Systems Modifications. Since all lines on the standard T/C are mounted at locations specifically tailored to interface with launch pad umbilicals, these locations and plumbing routings do not correspond well with those necessary for interfacing with the CISS. Figure 3-4 shows differences between T/C and SBTC fluid line routings. Note that all T/C lines run radially away from the vehicle and interface with pad umbilical lines that penetrate the payload fairing. To attach to the CISS, all fluid lines must be routed to the disconnect panels on the CISS. To do this, the S/C interface panels are installed to the aft end of the SBTC and all vehicle fluid lines will be routed to them. This allows for no changes to the CISS. The LH_2 tank fill and drain duct on the T/C is removed and a new line running from the tank penetration location to the appropriate disconnect panel replaces it. For the LH_2 tank vent line, a rerouting of lines is not possible since this requires a line routing along the LH_2 tank sidewall and would protrude from the Orbiter cargo bay envelope. The line must therefore be removed and the tank penetration sealed off and replaced with an S/C line routing as used on that vehicle. For the LO_2 tank fill and drain line, a simple line replacement can be used, routing the line from the CISS disconnect panel to the T/C penetration. Because the LO_2 tank vent line location would interfere with the CISS structure, this line is removed and the penetration plugged so that an S/C-type line routing can be used. For tank helium pressurant lines, the aft T-4 umbilical panel location on the T/C aft bulkhead will be retained and lines will be routed

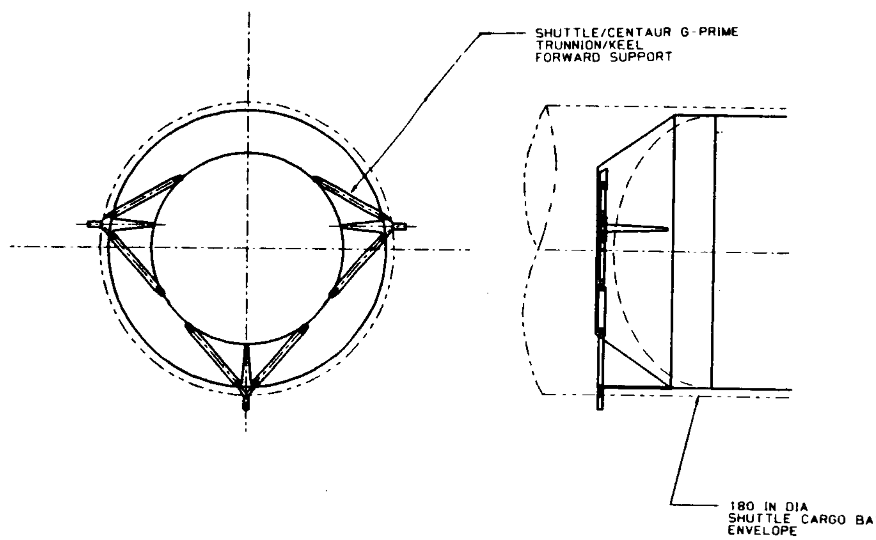


Figure 3-3. To Transport the Titan/Centaur in the Shuttle, the Shuttle/Centaur Forward Support Structure is Required

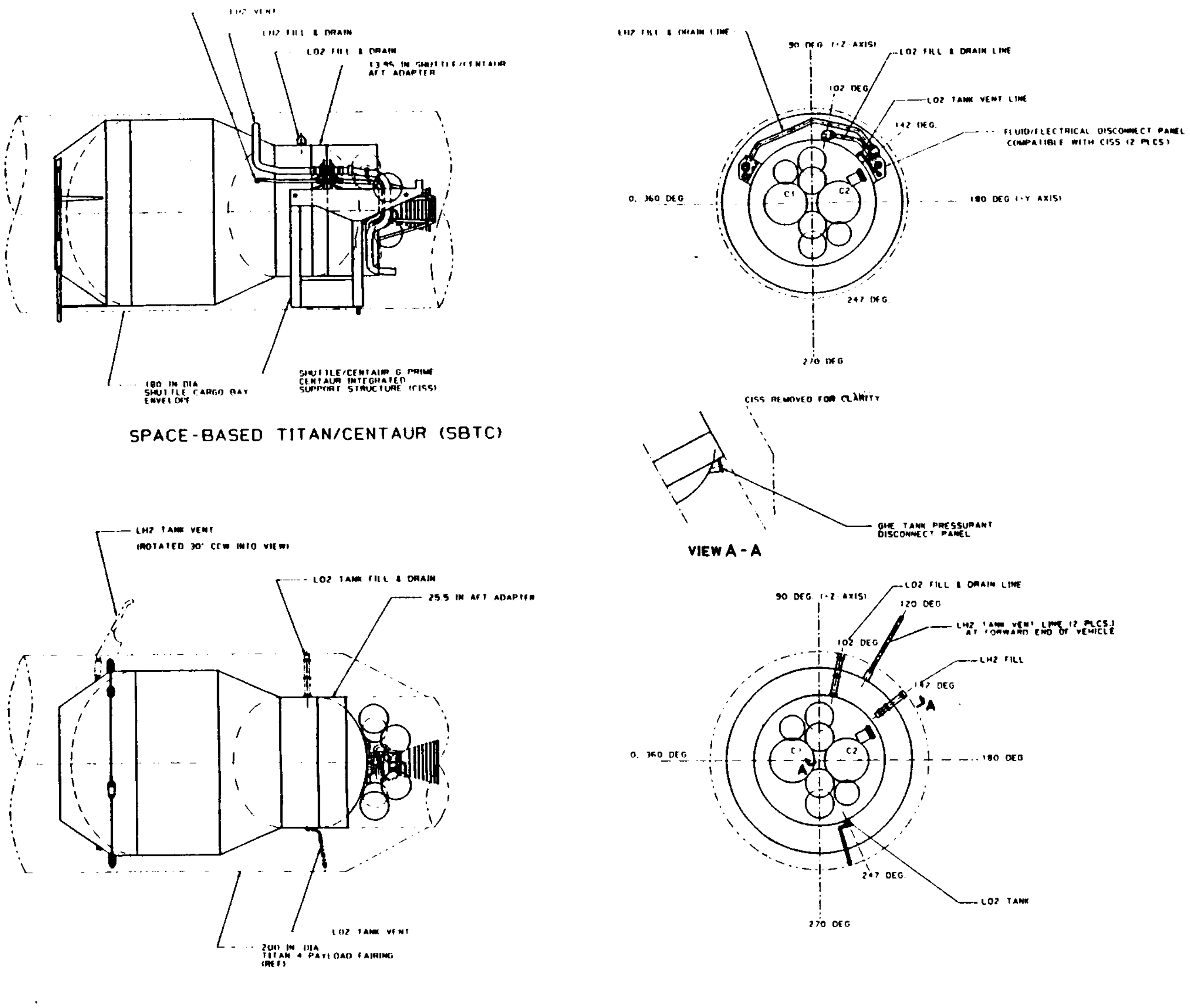
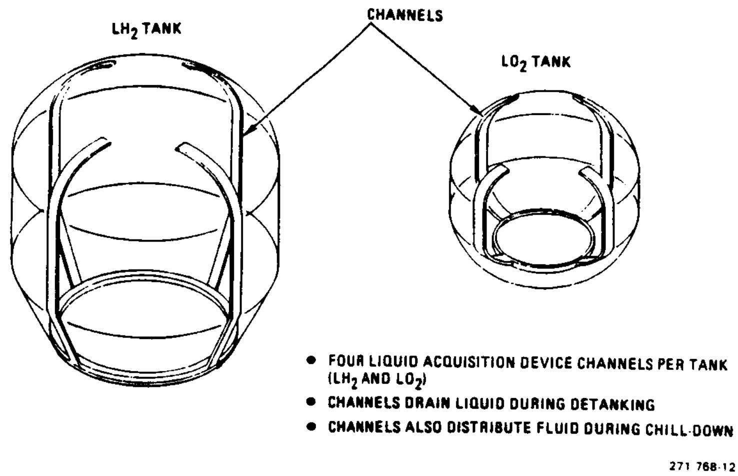


Figure 3-4. Titan/Centaur Fluid Line Routings Were Modified to Allow for Attachment to the Shuttle/Centaur CISS

to the interface at the aft disconnect panels. Finally, the electrical and instrumentation monitoring lines will be rerouted to locations at the upper portion of the aft adapter in order to mate with the CISS.

Many of the internal tank modifications identified in COSS for the space-based STS/Centaur will be required for SBTC and are shown in Figure 3-5. Zero-gravity mass gauges being developed by NASA/Johnson Space Center (JSC) must be installed in both tanks to measure fluid quantities during tanking and detanking. Liquid Acquisition Devices (LAD) in the form of a channel-type total liquid communication system are required for zero-gravity fill and drain in both tanks. These also provide efficient tank chilldown with a minimum liquid loss. Installed in the LH_2 tank is the S/C-developed Thermodynamic Vent System (TVS) which is required to allow a no-vent fill and liquid-free venting. Installed in the LO_2 tank is a mixer to increase agitation and



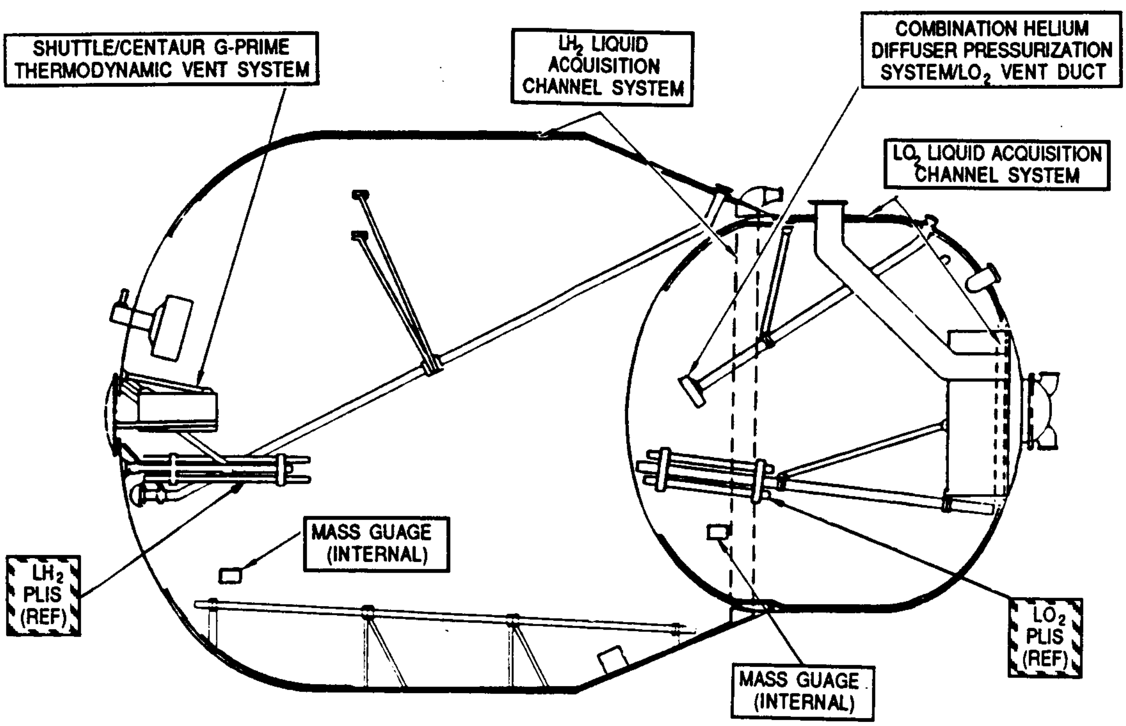


Figure 3-5. Internal Modifications Similar to Those Required for the Space-Based Shuttle/Centaur Are Necessary to Space Base the Titan/Centaur Upper Stage

allow for heat dissipation into the LH_2 tank so that an LO_2 TVS is not required. Since all fill and drain operations are conducted in a zero-gravity environment, the T/C Propellant Level Indicating System (PLIS) which would be used for ground fill would be inoperative and will be removed. Also removed is the T/C LH_2 Chilldown System ducting which is not required for the space-based operations.

Space-basing the T/C will not require any modifications to the avionics (except for software changes), since by 1991/1992, all T/Cs will be fitted with advanced avionics adequate to meet mission requirements.

3.1.2 CENTAUR HANGAR MODIFICATIONS. Three major changes to the COSS Centaur Hangar have been identified as being necessary to perform operations at the Space Station.

First, the hangar has been shortened by 8m (26.3 ft). The early Centaur Hangar was 10m (32.8 ft) high x 10m wide x 20m (65.6 ft) long. When reviewing station operations and the COSS operations animation, it was found that the station Mobile Remote Manipulating System (MRMS) arm's reach was not sufficient to allow a hand-off to the hangar Telerobotic Arm (TRA) without interfering with the upper hangar wall. A shorter hangar facilitates hand-off to the hangar TRA, and since the hangar length was initially sized to enclose both a payload and the Centaur/CISS Assembly (CCA), the only effect will be to expose part of the payload. Based on discussions with Ford Aerospace this should not affect payload operations, since payload spacecraft would be stored while on-station at the Satellite Processing Facility and would remain at the Centaur Hangar for a relatively short time.

Secondly, an aft door was added to provide for simplified Orbital Maneuvering Vehicle (OMV) mating operations. The aft opening wall now hinges 180 degrees outward. This allows the aft face of the CCA to be accessed during OMV mating without removing the CCA from its hangar. This also allows the Centaur to be rigidly fixed to the hangar during the OMV mating process. Modifications to the hangar to provide for the hinged aft wall require additional structure to frame the aft "door" as well as hinges and a drive motor. Figure 3-6 shows the aft-hinged door.

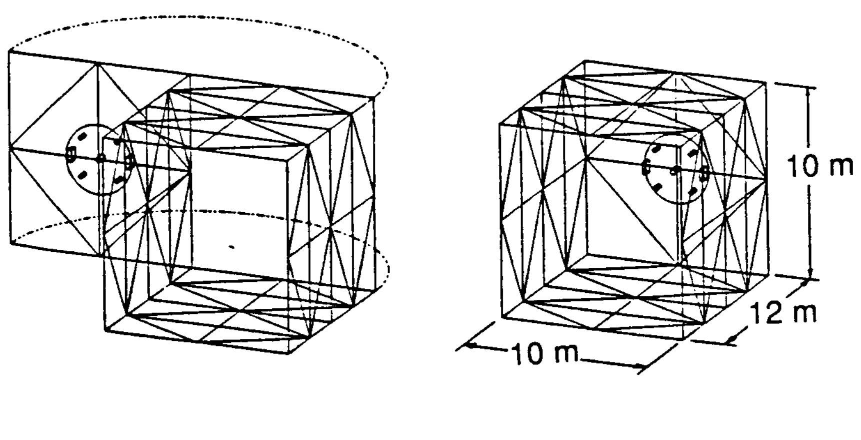


Figure 3-6. The Entire Aft Wall of the Centaur Hangar Hinges Out of the Way to Allow OMV Mating to the CCA

Third, to support the CCA and payload while the aft hangar wall opens and the OMV is mated, a Centaur Support Structure was added to the hangar. These two mechanisms rotate down from the hangar ceiling to grasp the forward three trunnions, one keel, and two longeron, of the CISS prior to the aft wall opening. These will be sized to provide support during the OMV mating operations at the CCA aft interface. Figure 3-7 shows the Centaur Support Structure attached to the hangar. The total weight impacts of all hangar changes on the station is shown in Table 3-2.

3.1.3 SPACE STATION AND CISS SCAR MODIFICATIONS. No changes are required to the station or the CISS other than those already discussed in the COSS Final Report.

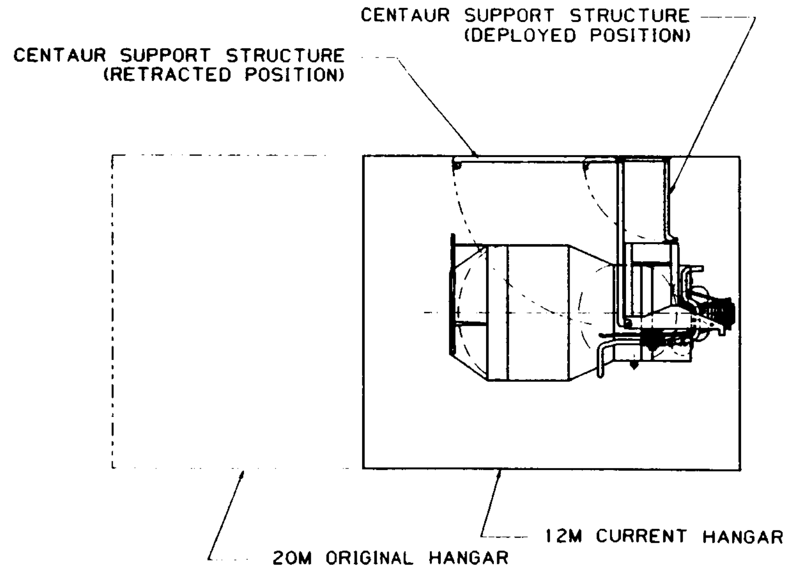


Figure 3-7. Hinging Centaur Support Structure to Hangar Reacts OMV Berthing and Mating Loads

Table 3-2. COSS Identified Changes to the Centaur Hangar Will Decrease the Total Weight by 3452 lb

COSS I vs COSS II
Hangar Weight Summaries

ITEM	COSS I [kg (lbs)]	COSS II [kg (lbs)]
Truss Structure, Aft Door(1)	4900 (2220)	3360 (1526)
Misc. Structure (TRA tracks, MFR attachments, etc.)	1430 (650)	660 (300)
Tele-Robotic Arm	2530 (1150)	2530 (1150)
Insulation/Debris Shield	10,630 (4820)	6380 (2892)
Electronics, Wiring	1100 (500)	660 (300)
Harnessing, Cabling	550 (250)	330 (150)
TOTAL	21,140 (9590)	13,930 (6318)

NOTES: EVA Tool Kit weights not included
(1)-aft door required for CSOD hangar only

3.1.4 OMV TRANSFER OPTIMIZATION. Three methods were evaluated for the OMV transfer maneuver from the Space Station to the Co-Orbiting Platform (COP) located 185.2 km (100 n.mi.) away. They simulated OMV engines executing: 1) two radial burns, 2) two tangential burns, and 3) a four-tangential-burn Hohman Transfer. The four-burn transfer was chosen as the best compromise between transfer time and fuel economy.

Figure 3-8 illustrates the appropriate differential equations and their general solutions for describing transfers between two nearby orbiting vehicles. Figure 3-9 illustrates the three methods and compares the time and ΔV requirements for each, independent of payload mass. The first method shown uses two radial, inward directed, thruster burns of equal duration. As illustrated in Figure 3-9, the response to the first inward burn ($\dot{z}_0$) one-half orbit later is a forward displacement equal to 4 $\dot{z}_0/w$ and an upward velocity equal in magnitude to the inward burn. A second equal inward velocity then restores a circular co-orbiting condition. The maximum altitude change downward occurs after a quarter orbit and is $\dot{z}_0/w$ or one-quarter the range. For a 185.2-km (100-n.mi.) range, the total ΔV requirement is the 112.3 m/s (368.5 fps). The time requirement is inherently one-half orbit.

Figure 3-10 shows the transfer time versus ΔV curve obtained from equations in Figure 3-8 for methods two and three. The slight "knee" in the curve was arbitrarily selected as the analytical point for both methods.

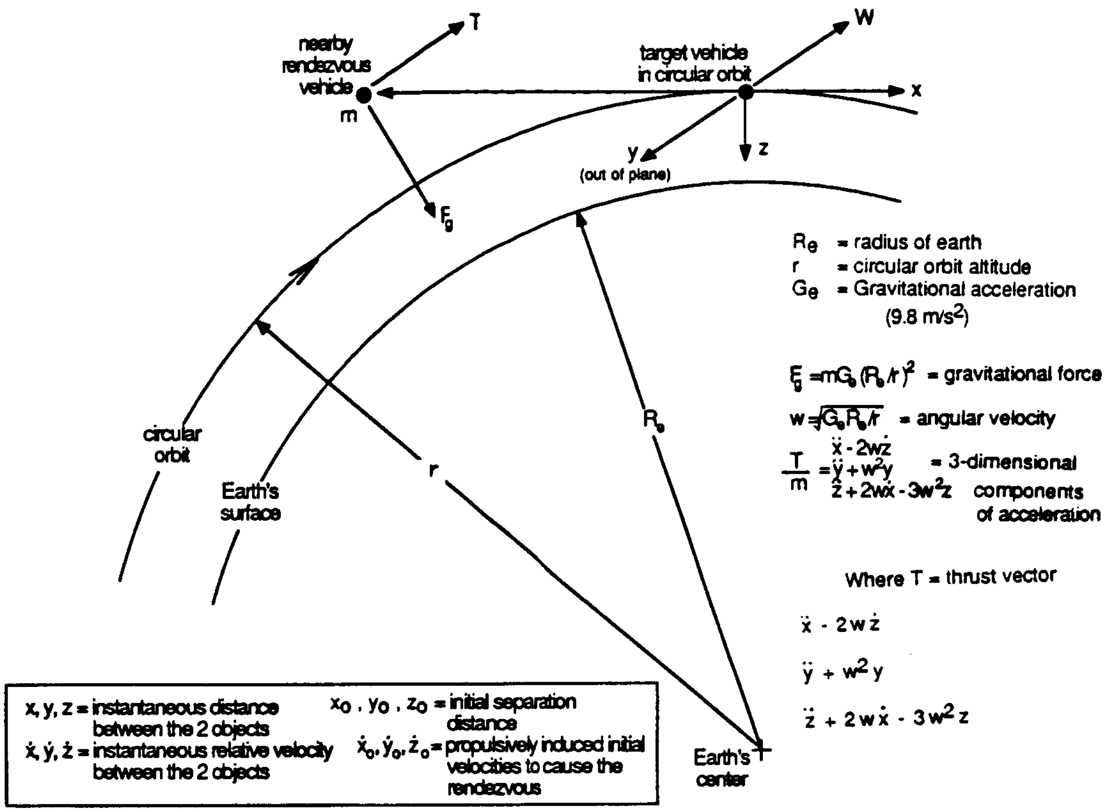
The second method, also illustrated on Figure 3-9, uses two tangential burns. The first, a retroburn, causes a slightly elliptical orbit whose reduced period gradually allows the OMV apogee to occur at the COP, where a recircularization burn, equal to the initial burn, is applied to cause the relative velocity to be zero. ΔV requirements are small, being only 6.04 m/s including 10% added for transfer orbit corrections. The transfer time can be reduced by increasing the ΔV, so long as transfers are limited to an integral number of orbits.

The third method uses a four-burn transfer. The first and second burn cause a Hohmann transfer to a lower altitude circular orbit. The slightly reduced period of the lower orbit causes the OMV to move toward the COP. Upon reaching the target, a Hohmann transfer is again executed to elevate the OMV back into the COP orbit. With the same ΔV used for the second method, the transfer time is slightly increased. Again, the ΔV requirement shown adds 10% for orbit corrections.

The return trip (shown in dashed line) has the same ΔV requirements. They are, however, applied in the opposite direction. The OMV returns via an increased altitude trajectory.

The four-burn method was chosen as optimum since its well-behaved trajectory eases guidance requirements for corrective action. Its two-way transfer time is well within the 40-hr battery life of the OMV, and the ΔV requirements are low. While not currently required, a decreased transfer time is available with an increased transfer ΔV. At this point, further refinement requires consideration of OMV characteristics, CCA, and payload weights.

Table 3-3 lists the OMV data used in this analysis. It was taken from the NASA/MSFC OMV User's Guide, October 1987 and the TRW Alternate System Design Concepts (Phase B) Study, August 1985.

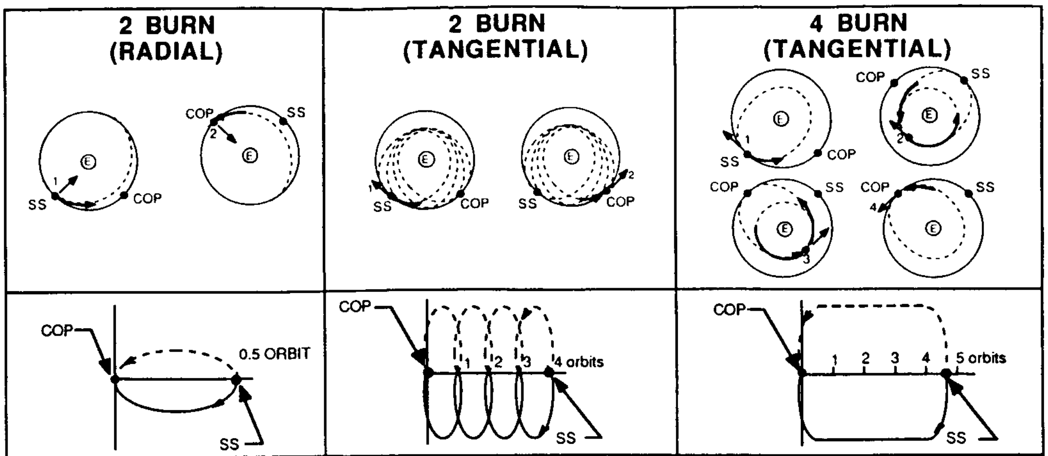


Let T=0 and assume a x_0, y_0, z_0 and $\dot{x}_0$, $\dot{y}_0$, $\dot{z}_0$ and determine relative distance and velocity sometime later. Also note that y-equation is uncoupled.

$$\dot{x} = \dot{x}_0(4\cos wt - 3) + 6wz_0(1 - \cos wt) + 2\dot{z}_0 \sin wt$$
$$\dot{y} = -y_0 w \sin wt + \dot{y}_0 \cos wt$$
$$\dot{z} = -2\dot{x}_0 \sin wt + 3z_0 w \sin wt + \dot{z}_0 \cos wt$$

$$x = (4\dot{x}_0/w)\sin wt - 3\dot{x}_0 t + 6wz_0(t - \sin wt/w) - (2\dot{z}_0/w)(\cos wt - 1) + x_0$$
$$y = y_0 \cos wt + (\dot{y}_0/w)\sin wt$$
$$z = (2\dot{x}_0/w)(\cos wt - 1) + z_0(4 - 3\cos wt) + (\dot{z}_0/w)\sin wt$$

Figure 3-8. These Equations Were Used to Develop Space Station to COP Transfer and Rendezvous Trajectories and Time and ΔV Requirements



	2 BURN (RADIAL)	2 BURN (TANGENTIAL)	4 BURN (TANGENTIAL)
TIME (hrs.)	.788 (0.5 orbits)	6.3 (4 orbits)	7.24 (4.6 orbits)
ΔV	112.3 m/s (368.5 fps)	6.04 m/s (19.8 fps)	6.04 m/s (19.8 fps)

Figure 3-9. Three Methods for CCA Transfer Were Evaluated During the Study

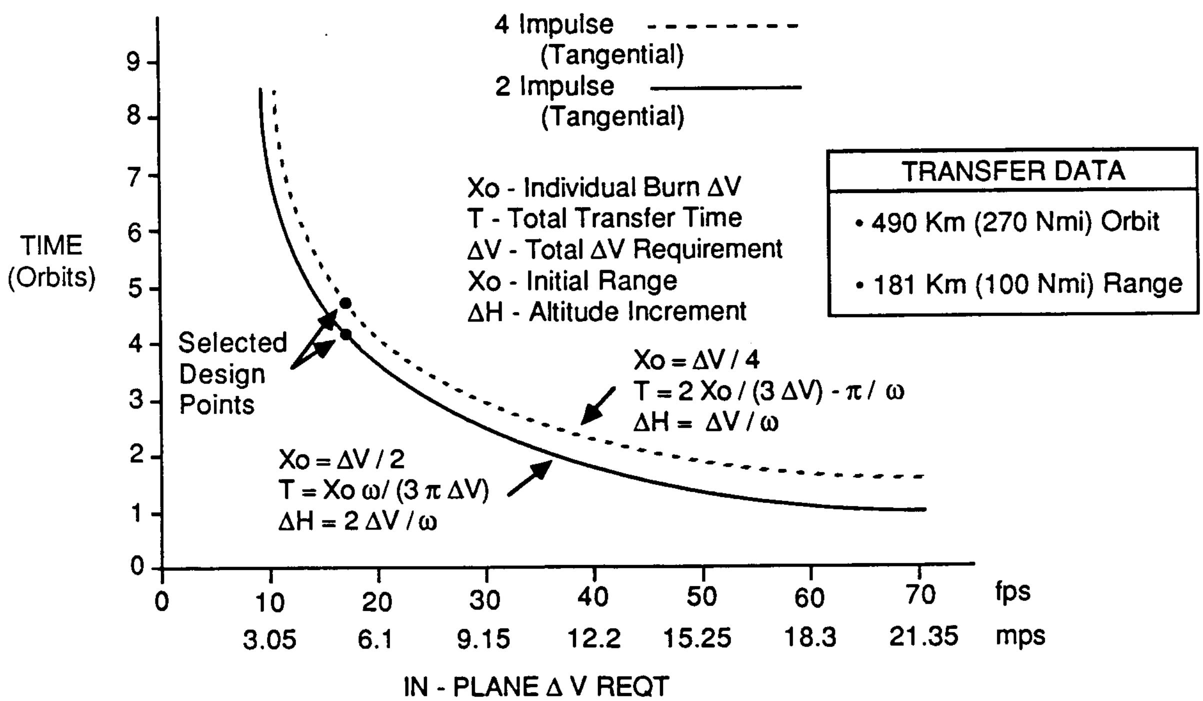


Figure 3-10. Rendezvous Time Is Inversely Proportional to the ΔV Requirement

Table 3-3. The Latest OMV Performance Characteristics Were Obtained for Use in the Analysis

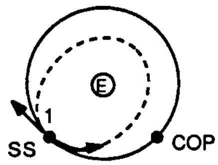
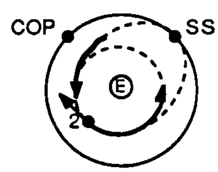
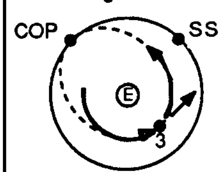
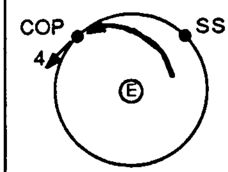
OMV empty weight	3040 kg (6702 lbs)		
Propulsion System			
Cold Gas	22.6 N	(5 lbs)	thrust / engine
	74.8 Kg	(165 lbs)	propellant
			66 sec specific impulse
RCS	53.0 N	(12 lbs)	thrust / engine
	544 Kg	(1200 lbs)	propellant
			220 sec specific impulse
Main	57.8 to 577.8 N	(13 to 130 lbs)	thrust / engine
	4082 Kg	(9000 lbs)	propellant
			288 sec specific impulse

The Remote Manipulator System (RMS) would be the active element in the docking operations. The OMV's role is to come into and remain within RMS range and maintain attitude control for RMS docking. We estimate this should require 4 fps ΔV, which can be satisfied with the OMV cold gas thrusters designed for proximity operations. The hydrazine-fueled RCS thrusters would be used for the four-burn transfer mission and for guidance corrections. Guidance corrections were sized at 10% of the total of the four main burn ΔV. Use of the OMV main bipropellant propulsion system is not required.

Additional data for the selected four-burn tangential transfer is given in Figure 3-11. The cold gas thrusters will provide proximity operations near the Space Station to allow the OMV to drift sufficiently before firing the hydrazine thrusters. The first engine burn occurs 1 hr after deployment. At this point, the OMV will be 2.2 km (1.18 n.mi.) away from the station. When the OMV reaches perigee, the second burn occurs. At this point, it is 4.95 km (2.67 n.mi.) below and 13.9 km (7.5 n.mi.) in front of the Space Station. After 3.6 orbits, the third burn (posigrade) occurs and sends the OMV into an elliptical orbit with an apogee at the COP's orbit. This burn occurs when it is 4.95 km (2.67 n.mi.) below and 13.9 km (7.5 n.mi.) behind the COP. The final hydrazine burn (circularization) occurs half an orbit later when the OMV is still 2.2 km (1.18 n.mi.) behind from the COP to prevent hydrazine COP contamination. The remaining distance will be covered by small cold-gas thruster firings in proximity of the COP.

Table 3-4 lists mission events for a complete roundtrip. The outbound trip event times correspond to those of Figure 3-11. The inbound trip corresponds to the outbound except for the target change from the COP to the Space Station.

Three Space Station-to-COP transfer trips are necessary during the course of TDM operations. Each was analyzed to determine its fuel requirements. The results are shown in Figure 3-12. The first trip is a transfer of the CCA to the COP for the zero-gravity Cryogenic Propellant Resupply TDM. After dropping off the CCA, the OMV then

	BURN 1 Retroburn	BURN 2 Circularization	BURN 3 Posigrade burn	BURN 4 Circularization
				
ELAPSED TIME (hrs)	1.0	1.8	7.5	8.3
DISTANCE FROM STATION ΔAltitude/ ΔDownrange	0 / 2.2km (0 / 1.2 nm)	4.95 / 13.9km (2.68 / 7.5 nm)	4.95 / 171.3 km (2.68 / 92.5 nm)	0 / 183.1 km (0 / 98.8 nm)
DISTANCE FROM COP ΔAltitude/ ΔDownrange(nm)	0 / 183.1 km (0 / 98.8 nm)	4.95 / 171.3 km (2.68 / 92.5 nm)	4.95 / 13.9 km (2.68 / 7.5 nm)	0 / 2.2km (0 / 1.2 nm)
ΔV (fps) [not including proximity operations]	1.37 m/sec (4.5 fps)	1.37 m/sec (4.5 fps)	1.37 m/sec (4.5 fps)	1.37 m/sec (4.5 fps)

NOTE: Total Δ V = [1.37 m/sec x 4 burns] x 1.10 proxops = 6.04 m/sec

Figure 3-11. The Four Tangential Burn Approach Gives a Well-Behaved, Efficient Transfer Rendezvous

immediately returns to the station. The second trip is to retrieve the CCA. The OMV returns alone and brings the empty Centaur and CISS back to the station. A third trip is exemplified by the transfer of the CCA, Multiple Payload Adapter (MPA), and payload(s) for the launch in the Operations TDM.

Preliminary planning assumes the heaviest payload would be the FACC Evolutionary Communications Platform (ECP). The transfer equations (Figure 3-8) for the chosen four-burn transfer were redone to include actual SBTC and payload masses, and proximity operations. Results are shown in Table 3-12. It can be seen that the propellant requirement is well below the total OMV capacity. If required, the transfer time could be reduced with an increase in mono-propellant requirements defined earlier. The current two-way transfer time of 18.6 hr is well below the OMV battery limit of 40 hr. However it may still be desirable to reduce the transfer time. About 11.3 hr of the 18.6 hr is directly associated with the transfer. A possible mission improvement would be to half the 11.3-hr time by doubling the hydrazine requirement. The hydrazine requirement is still well within OMV capacity and the total roundtrip requirement would be reduced to about 13 hr.

3.1.5 <u>PAYLOAD ADAPTER ANALYSIS AND CONCEPTS</u>. The development of a common payload interface is considered crucial to the efficient use of an STV to deliver a variety of payloads. There is presently no standard interface between launch vehicles. Even on the same launch vehicle, many payload-peculiar modifications are required. For STV space operations to have maximum flexibility, satellite manufacturers would be encouraged to adopt standard interface on future satellite designs. The following describes the procedure used to develop STV interface concepts which could be tested by COSS TDMs.

Table 3-4. Only the OMV Cold Gas and Reaction Control Thrusters Are Required for Transfer to and from the COP

Event	Time Req'd. (hrs) event	total	Fuel Source
Deploy from SS	0.5	0.5	cold gas
Coast to clear SS	0.5	1.0	" "
Outbound burn 1	0.02	1.02	mono-propellant
Hohmann 1/2 orbit coast (descent)	0.786	1.806	
Burn 2	0.02	1.826	
Coast to COP(1)	5.67	7.296	
Burn 3	0.02	7.516	
Hohmann 1/2 orbit coast (ascent)	0.786	8.302	
Burn 4	0.02	8.322	
Remote piloted COP approach	0.5	8.822	cold gas
RMS recovery of OMS	0.5	9.322	" "
OMV disengage/coast to clear COP	1.0	10.322	cold gas
Return Burn 1	0.02	10.342	mono-propellant
Hohmann 1/2 orbit coast (ascent)	0.786	11.128	
Burn 2	0.02	11.148	
Coast to SS(1)	5.67	16.818	
Burn 3	0.02	16.838	
Hohmann 1/2 orbit coast (descent)	0.786	17.624	
Burn 4	0.02	17.644	
Remote piloted SS approach	0.5	18.144	cold gas
RMS recovery	0.5	18.644	" "

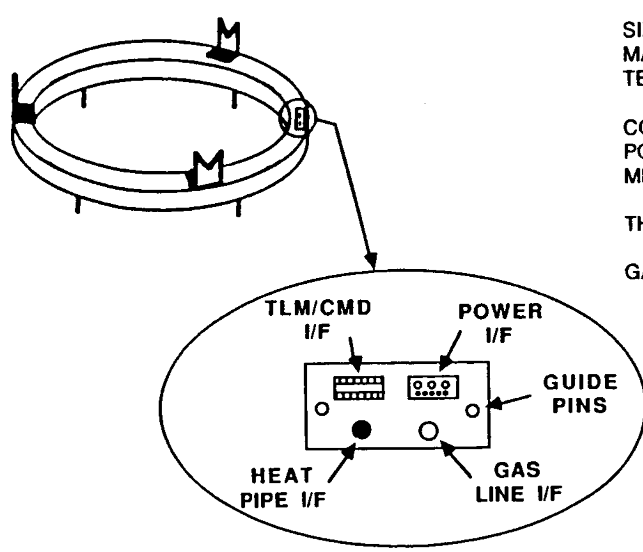
(1) Includes radar search and track for rendezvous burns

3.1.5.1 Universal Payload Adapter. An investigation of payload requirements and existing interfaces provided the basis for a derived Universal Payload Adapter (UPA) with a standard interface. This interface provides for the potential fluids, avionics/electrical and thermal requirements as derived from information gathered on future spacecraft and Space Station needs.

Our UPA design is shown in Figure 3-13 along with its maximum services values. The UPA will physically be 1.27m (50 in.) in diameter with a mass of 43.2 kg (95.2 lb). This adapter will attach to the front of the Centaur Transition Section for single payload deliveries (Figure 3-14). The interface must be able to provide power and electrical signals as its primary function. For versatility, our design accommodates, as an optional service, fluid and thermal interfaces. A brief assessment of each of these follows.

MISSION		WT.	PROPELLANTS CONSUMED COLD GAS N2	PROPELLANTS CONSUMED MONO-PROPELLANT	TOTAL TIME (hrs)
Propellant Transfer Experiment Delivery	TO COP: OMV+CCA	10,355 kg (22,828 lb)	19.5 kg (43 lb)	32.2 kg (71 lb)	9.32
	TO STATION: OMV	4,094 kg (9,025 lb)	7.7 kg (17 lb)	12.7 kg (28 lb)	9.32
Propellant Transfer Experiment Return	TO COP: OMV	4,145 kg (9,139 lb)	7.8 kg (17.2 lb)	12.9 kg (28.4 lb)	9.32
	TO STATION: OMV+CCA	10,334 kg (22,782 lb)	19.5 kg (42.9 lb)	32.1 kg (70.8 lb)	9.32
(3) FS-1300 Satellite Delivery	TO COP: OMV+CCA+ECP	15,304 kg (33,738 lb)	28.8 kg (63.6 lb)	47.6 kg (104.9 lb)	9.32
	TO STATION: OMV+CISS	7,230 kg (15,939 lb)	13.6 kg (30 lb)	22.5 kg (49.6 lb)	9.32
Evolutionary Comm. Platform Delivery	TO COP: OMV+CCA +MPA+(3) FS-1300	12,644 kg (27,875 lb)	23.8 kg (52.6 lb)	39.3 kg (86.7 lb)	9.32
	TO STATION: OMV+CISS	7,243 kg (15,968 lb)	13.6 kg (30 lb)	22.5 kg (49.5 lb)	9.32

Figure 3-12. Transfer Propellant Requirements are Well Below OMV Capability



SIZE	1.27 M DIA
MASS	43.2 KG
TELEMETRY	2.4 KBPS, PYRO CONTROL SEPARATION BREAKWIRES
COMMAND	.8 KBPS DISCRETE UPLINK
POWER	1.5 KWATT
MECHANICAL	3 LATCH OMV DESIGN, POSITIVE CONTROL, SPRING EJECTION
THERMAL	HEAT PIPE FITTING, ORU REQD FOR OPT SERVICE
GAS	.6 CM LINE, OPT SERVICE, SPACE ON MPA FOR MODULAR ATTACHMENT.

LEGEND

TLM	TELEMETRY
CMD	COMMANDING
I/F	INTERFACE

Figure 3-13. The UPA Will Provide a Common Interface Between the SBTC and Payloads

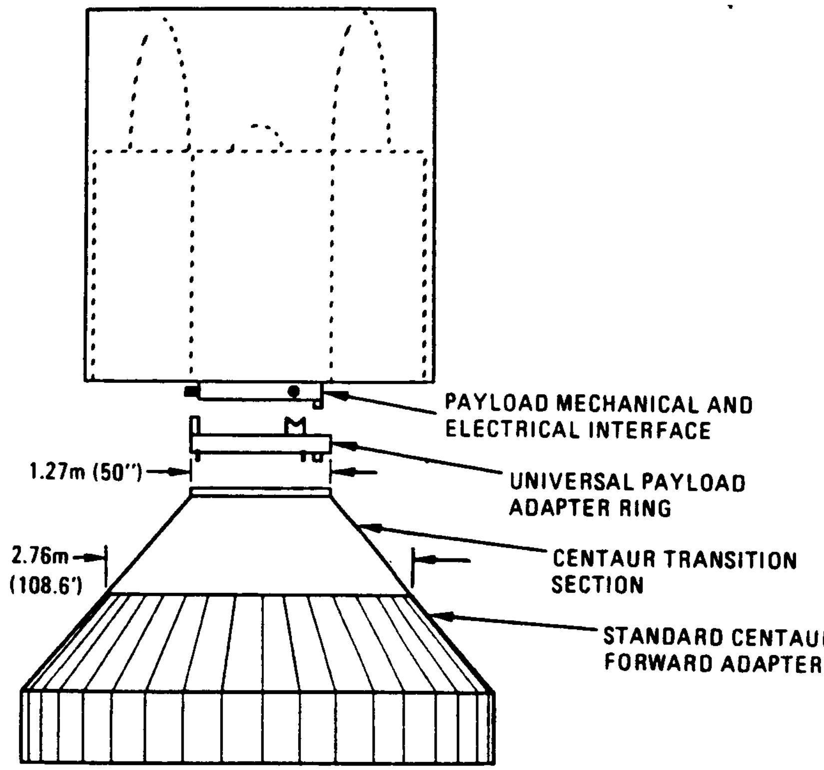


Figure 3-14. UPA Latches are Motor Driven and Have a Payload Ejection Spring

Avionics/Power/Electrical. Each of the satellites investigated required power from an external source during transfers and delivery. The requirement ranged from 100 to 1000W depending on the systems powered up and the expected heater requirements. As a result, a UPA interface capability of 1.5 kW was chosen to accommodate greater future needs. The provisions for telemetry were desirable but a loss of telemetry was not considered critical. Spacecraft were deemed sufficiently dormant prior to delivery and appendage deployment (if applicable), that little data was actually required to be certain of health. Commanding was an important function, but few commands were required. Most spacecraft will require only one command at a pre-determined time before deployment and then will autonomously control all sequences, including firing of the separation pyros. Other commands, such as uplink control of heaters or other systems, were not considered a significant function.

Thermal. UPAs will provide an interface with the satellite for an optional heat pipe dissipation system.

Mechanical. A three-latch mechanical interface will be used based on the OMV interface and hardware. The latches will have positive control (motor driven) and will provide spring ejection of the satellite at deployment to impart a small separation velocity (Figure 3-14). Guide pins on the electrical connector will assist with proper alignment and ensure interface integrity. Zero-force insertion electrical and power connectors will preclude pin jams and separation friction. The action of pulling in and locking the satellite with the three holddown latches will cause the electrical connectors to grip the pins. Similarly, release of the latches will cause the connectors to release the pins.

Fluid. A gas line interface will be provided to allow a purge system to be employed. The Multiple Payload Adapter (MPA) can be scarred to accommodate gas bottles and a plumbing system, but will not typically provide a purge gas service. A 0.63-cm (0.25-in.) line will provide the desired flow rates.

Space Station. The most important standard for vehicles operating in and around the Space Station will be the MRMS and OMV payload interface standard (see Table 3-5). The complement of electrical and mechanical connectors and capabilities provided by these interfaces will be the only utilities available during transfers using these systems. Other research into Ariane interfaces, Commercial Atlas/Centaur interface plans, and discussions with Ford Aerospace Communications Corporation and Hughes Space and Communications Corporation provided additional insight into future satellite requirements and design plans. A U.S. Air Force (USAF) Space Division report on Spacecraft Partitioning and Interface Standardization (see Bibiography) of satellite systems provided additional information on industry goals and discussion of potential standardization approaches. All this information was used in deriving the types and service values for our UPA.

3.1.5.2 Multiple Payload Adapter. Our MPA concept is shown in the Figure 3-15. When attached to the SBTC it will allow for multiple payload delivery. Payload attachment locations were picked after developing SBTC performance capabilities (see Section 3.2). Although the design can accommodate up to six payloads, the limiting practical case, due to propellant boil-off constraints, was the potential to deliver five GPS satellites. This combination determined a 2.2m (87.2-in.) radius UPA attach centerline. Based on spacing requirements for five 1.3m (50-in.)-diameter UPAs, a UPA diameter of 5.8m (19 ft) is required. This diameter does not allow for single-piece cargo bay delivery, and thus will require assembly at the Station.

The exploded view in Figure 3-15 shows that the MPA has four major elements. The forward interface panel allows for attachment of up to six UPAs to allow for multiple payload delivery. Each of the six fixed interfaces are common. In addition to providing structural attachment for the payloads, the MPA also provides signal multiplexing for commanding and telemetry for the payloads carried per a pre-programmed sequence loaded before the flight. The central utility cableway routes the utilities to a main bus and down to the Centaur vehicle. The six compression panels carry the main thrust and bending loads from the payloads to the vehicle and the aft interface panel mates to the Centaur through a transition section as described in the COSS Final Report. The weight summary for the MPA is given in Table 3-6.

Multiple payload delivery is complicated by the fact that, as each payload is deployed from the MPA, a new center-of-gravity (CG) location results. This off-axis CG shift maximizes at a point in time just prior to final payload release. Analyses were done on payload configurations, working from maximum payload capacity to final payload release. The worst case being the final deployment of an FS-1300 satellite (1540 kg at 1.5m above interface). The composite CG location shown is for an empty Centaur (but includes RCS propellant). As can be seen in Figure 3-16, to thrust through the CG, the

Table 3-5. The OMV and MRMS Interface Requirements Were Considered in our UPA Design

Commanding:	160 bps	256 commands
Telemetry:	800 bps	

<u>OMV Peculiar Options</u>

If No OMV Commanding or Telemetry Required

Commands:	1 kbps TDRS
	2 kbps GSTDN
Telemetry:	14 kbps TDRS Multiple Access
	28 kbps TDRS Single Access

G N & C	Provide OMV attitude and State Vector to payload
Power	Five kwhrs at no greater than 1 kw/hr without power augmentation kit (1.8 kw/hr, 52.2 kwhrs.)
Thermal	No active thermal control is provided. Thermal isolation of payload from OMV is required.
Mechanical	Standard Grapple Fixture (Three point docking adapter with positive control latches and spring ejection on OMV.)

main engine gimbal requirement is 6.88 degrees. Since the Centaur RL-10 engines can gimbal without mechanical interference up to eight degrees (although the present Centaur is programmed to stop the engines at three degrees), no difficulties should be encountered with this off-axis distance. This angle results in a loss of only 0.7% of the engines' thrust. Note that structural and dynamic analyses would be required to analyze these higher than normal gimbal angles.

3.1.6 <u>SPACECRAFT HANDLING AND PROTECTION DURING INTEGRATION AND LAUNCH</u>. The handling of satellites for the COSS II program, from preparation for mating to the Centaur until their release in the proper orbit, required investigation of following four major areas:

- Control of movement to prevent damage
- Physical protection of the spacecraft and its equipment
- Provisions for Communication/Telemetry Required
- Providing Thermal Management.

The accommodation of the spacecraft in these four areas ensures that the integrity of the satellite will be maintained until it becomes operational on orbit.

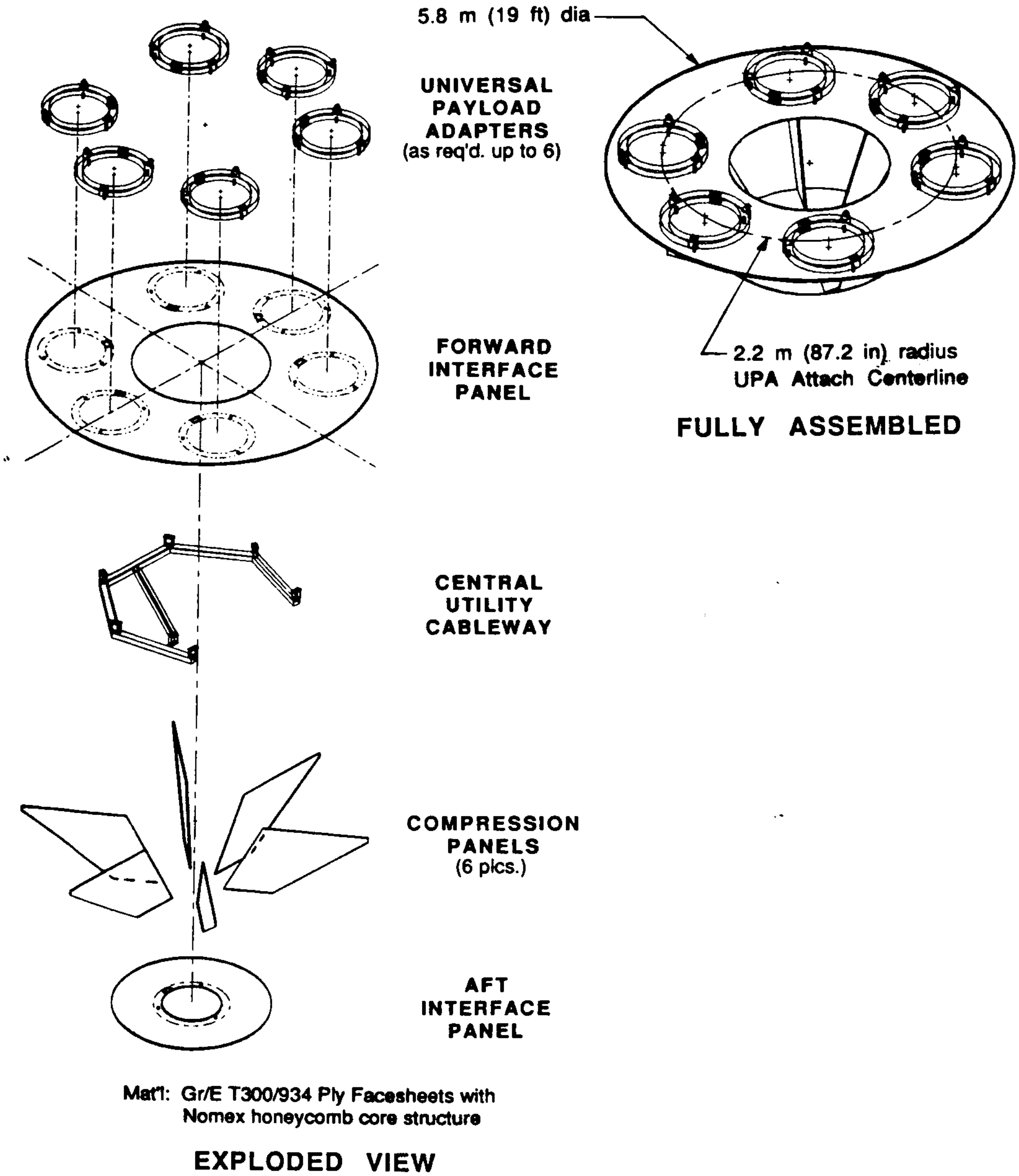


Figure 3-15. Our MPA Concept Will Allow SBTC to Deploy Two to Six Payloads

Table 3-6. The Estimated Weight of our MPA Concept is 725 lb

Structure	105 kg	(230 lbs)
Mechanisms	164 kg	(360 lbs)
Wiring	18 kg	(40 lbs)
Contingency	43 kg	(95 lbs)
TOTAL	330 kg	(725 lbs)

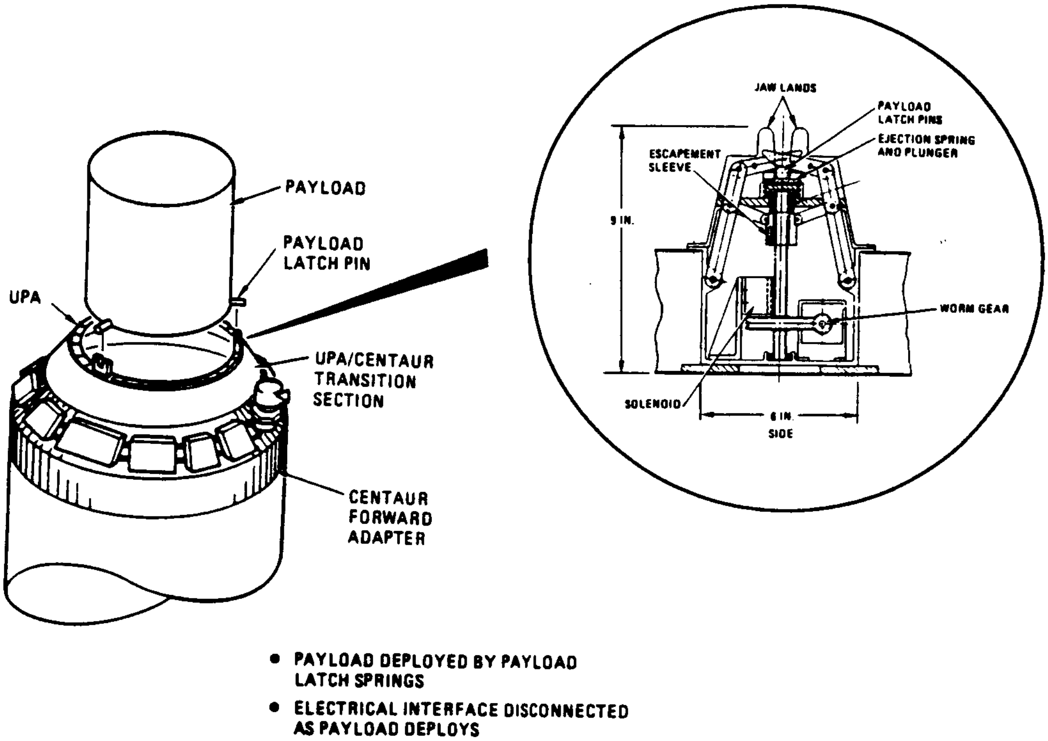


Figure 3-16. For All Payload Manifesting Recommendations of Our Study, Off-Axis CGs Can Be Accounted for With Main Engine Gimballing

3.1.6.1 Control of Movement. The satellite designs evaluated for this study were for the 1995 timeframe and as such contained the grapple fixtures required for movement by the Space Station MRMS and hangar TRA. Movement of the satellites from the Satellite Processing Facility (SPF) to the Centaur hangar will be carried out using the MRMS remotely controlled from the Space Station control room. For single satellite launch cases, the satellite alone will be transferred. For multiple satellite launches, the satellites will be integrated with the MPA in the SPF, then the loaded MPA would be transported to the Centaur Hangar. Movement of the MRMS with a load is limited to approximately 0.6 meters per minute. At this rate, the move from the SPF to the Hangar will take about 1 hr which allows monitoring the movements to protect against contact with other surfaces. Additionally, remote television viewing, bumper guards and software motion stops will ensure the satellite does not contact any Space Station or hangar structure.

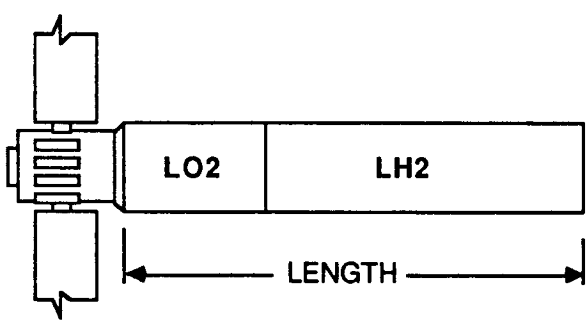
3.1.6.2 Satellite Protection. The satellites will be protected while in storage at the Space Station by the SPF which will provide the necessary resources. This includes a covering for micrometeoroid and atomic oxygen protection, passive thermal control, power, and telemetry services. Once the satellite is removed, though, this protection will not be available. The micrometeoroid and atomic oxygen protection is not considered a problem due to the short duration of exposure (less than two weeks). Passive protection of satellite sensors (e.g., star trackers, earth sensors) will be accommodated by design of MRMS movements, OMV/CCA transfer procedures to the COP, and COP pointing and operations during tanking for launch. The satellite manufacturers will likewise be encouraged to provide active protection with sunshields and deployable covers over sensitive sensors. Contamination will be minimized through operational design of the spacecraft handling procedures and provision for an optional helium purge capability. The Space Station will provide helium for this purge both in the SPF and while attached to the Centaur. The MPA will be scarred to accept an optional helium purge system and the UPA provides a purge gas interface to the satellites.

3.1.6.3 Signal Provisions. The satellites will require continuous support of power, telemetry and commanding which the SPF will provide. During the transfer of a satellite, or the MPA and multiple satellites, these resources will be provided via the MRMS electrical interface. Very limited power and telemetry capability exists, especially in handling multiple satellites, but will allow health monitoring during the transfer and insight into the Satellite thermal condition. Once mated to the Centaur, the CCA will provide the necessary resources via the interface to the Space Station. Similarly, the OMV/CCA will provide telemetry, commanding, and power during the transfer to the COP with the CCA/COP providing these upon mating at the COP. At each step it will be crucial to know the satellite health state so that corrective action can be taken as soon as possible. Limited uplink commanding will be available to assist in providing active thermal control as required. The status and safety of pyro initiators will be verified via telemetry to the Space Station and ground prior to transferring the Centaur to the COP. Information on the health of the separation breakwires will be confirmed prior to activating the satellite for final checkout and launch and spacecraft arming for flight will be commanded while the Centaur is tanked at the COP and final countdown has begun.

3.1.6.4 Thermal Management. Thermal management of the satellites will be one of the most critical aspects of ensuring satellite health during its period of storage and preparation for flight. The SPF will provide the necessary resources to thermally protect the satellite while it is in storage. Once removed from the hangar in preparation for flight, a combination of active and passive thermal management will be employed. Passive thermal control will consist of designing the satellite so that critical elements are insulated and planning the satellite transfers to minimize direct solar exposure to any one area. Additionally, telemetry monitoring will allow insight into satellite temperatures. The approach of an avionics unit or instrument to its high- or low-temperature redline can be corrected by re-orienting the satellite or by active control. Active control will be the responsibility of the satellite manufacturer to provide heaters in areas where low-temperature concerns exist. Power, telemetry, and commanding will exist to allow the satellite user to discretely manage the satellite thermal state. The MPA will provide scarring for an optional heat pipe dissipation system and the UPAs will provide an interface with the satellite.

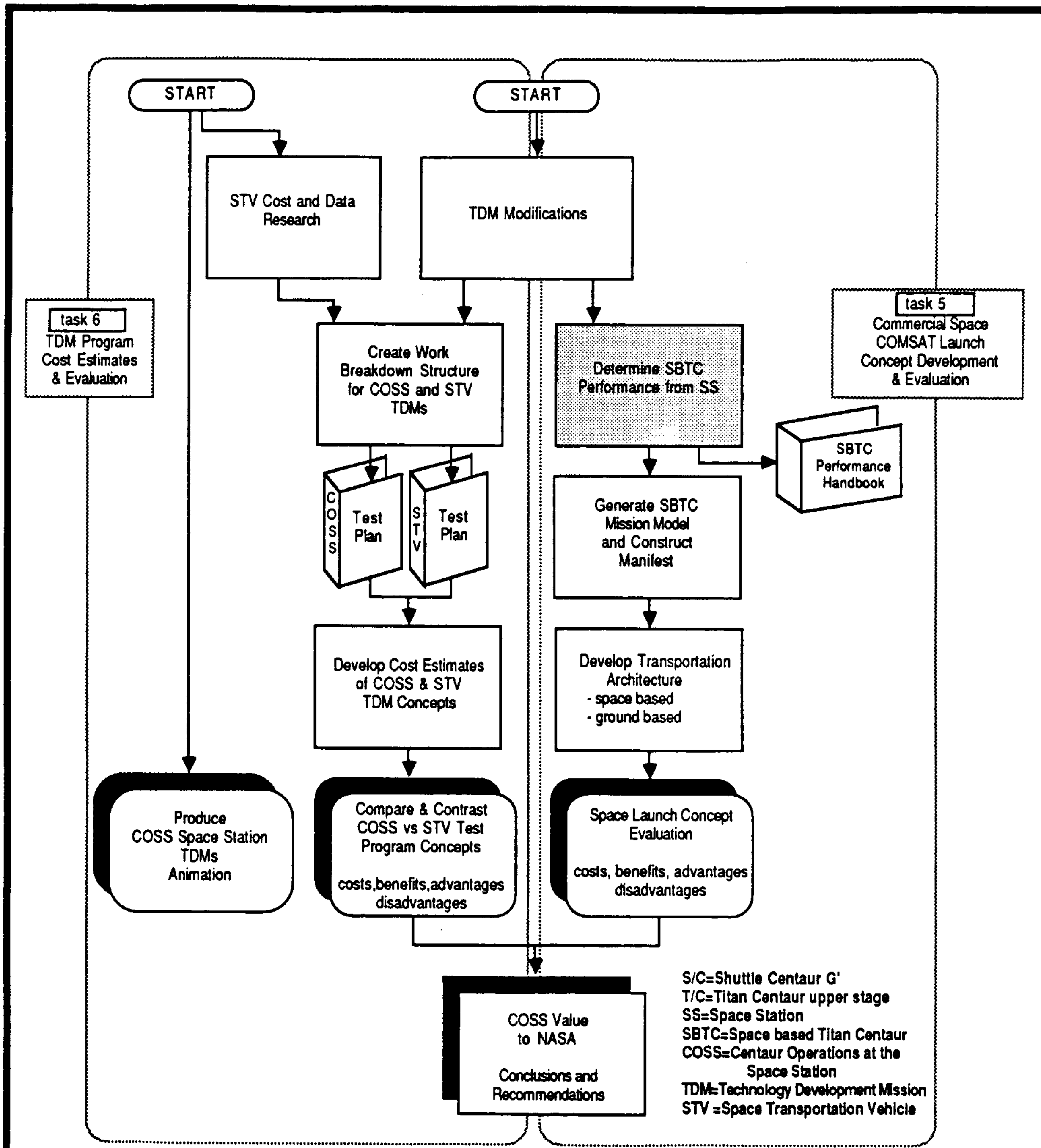
3.1.7 CO-ORBITING PLATFORM CAPACITY OPTIONS. The original COP tank sizes and capability were based on a single Centaur's tanking requirements to support the TDMs, and to perform a single actual mission. Using a combination of Shuttle and Titan IV launches to deliver the propellants resulted in a COP capacity of about 27,000 kg (60,000 lb). For routine COMSAT delivery operations, two additional concepts for COP tank capabilities have been evaluated. The ALS E, ALS/FBB, and STS-C launch vehicles, to become available by the mid 1990s, will allow for larger COP tanksets, servicing two or three SBTC flights without refueling the COP. Table 3-7 compares the original test program and additional operational program concepts. The 45,450 kg (100 klb) propellant depot was chosen as the nominal baseline for commercial space operations analysis. The operation of the COP would not be affected by the size of the propellant tanks attached. The COP could be initially configured for the 55 klb fueling TDM test program concept, then be switched to the 100 klb baseline if a commercial transportation program becomes operational.

Table 3-7. The Size and Mass of the COP Will Depend on the Concept Chosen for Delivery



DEPOT CONCEPTS	ORIGINAL	CONCEPT 1	CONCEPT 2
PROPELLANT MASS	27,270 kgs (60 klbs)	45,450 kgs (100 klbs)	63,630 kgs (140 klbs)
LENGTH	13.5 m (44.3 ft)	16 m (52.5 ft)	18 m (59.1 ft)
TOTAL MASS (Structure & Prop)	41,480 KGS (91.3 klbs)	68,750 kgs (151.3 klbs)	89,430 kgs (196.7 klbs)
DELIVERY VEHICLE	SHUTTLE & TITAN IV	STS-C OR ALS E	ALS/FBB
NUMBER SBTC FLIGHTS SUPPORTED	1	2	3

3.2 DETERMINE TITAN/CENTAUR PERFORMANCE FROM SPACE STATION



3.2 DETERMINE TITAN/CENTAUR PERFORMANCE FROM SPACE STATION

It seemed intuitively obvious that launch vehicle capabilities from the Space Station would be greater than from ground launch. To quantify this, a Space-Based Titan/Centaur (SBTC) performance analysis package was developed. This data was then used to assist in making the manifesting recommendations later in this study. It should also provide sufficient data to NASA/LeRC to allow analysis of options not given. The performance was done for single and double communications satellite concepts as well as multiple GPS satellite manifests. Assessment of plane change, inclination change, and spacing capabilities were carried out.

The performance analysis for the SBTC capabilities has been developed for the cases of:

- Single Payloads
  - Altitude Capabilities
  - Plane Change Capabilities
  - Earth Escape Capabilities

- Dual Payloads
  - Same Orbit, Different Spacing
  - Same Altitude, Different Inclination
  - Different Altitude, Same Inclination

- Multiple Payloads
  - GPS Delivery, two to five in Same Altitude and Orbit Plane
  - Number of Satellites Versus Allowable Satellite Weight: Equal Weights, GEO Only

Four computer analysis programs were used to investigate these areas. An overall flowchart, a brief description of the program architecture, and greater details on the programs (e.g., individual flow charts, variable lists) are provided in Appendix A.

3.2.1 PROPELLANT BOILOFF PREDICTIONS. The delivery of multiple satellites will require coast times for proper placement of subsequent satellite deliveries. The boiloff that will occur during these coast times is a function of solar radiation, exposure to earth albedo, altitude above the Earth, Centaur orientation, amount of propellant remaining, and the amount of insulation covering the Centaur. The complexity of these relationships was simplified in this analysis by defining a very conservative set of boiloff assumptions. The boiloff effects were accounted for by assuming a 25% or 50% boiloff of all remaining propellants during the coasts between deployments. For example, using the 25% boiloff rate assumptions, Figure 3-17 summarizes the calculated propellant lost. On the same chart, the total propellant loss is converted to an average boiloff rate (average kilograms per hour). The mission can thus be accomplished if the actual boiloff rate is equal to or less than this number. For multiple satellite deliveries, a comparison of the average resulting boiloff rate with the multilayer insulation (MLI) summary shows that only the 6-satellite delivery case requires more than 15 layers of MLI to perform the mission. Figure 3-18 then illustrates the relationships for the GPS delivery case between number of satellites delivered, boiloff rate, insulation required, and total delta velocity required. It should be noted that the boiloff rate scale is logarithmic due to the wide scale

60° SPACING, 25% BOILOFF (kgs)

	ON ORB PROP	1 st VENT	2 nd VENT	3 rd VENT	4 th VENT	5 th VENT	AVER. KGS/HR	TOTAL TIME (hrs)
2 SATS	4975	1312					312.4	7.2
3 SATS	4302	800	212				70.3	17.4
4 SATS	3629	766	359	108			31.9	41.7
5 SATS	2956	661	380	189	63		9.9	134.2
6 SATS	2283	527	341	207	110	41	2.5	491.0

120° SPACING, 25% BOILOFF (kgs)

	ON ORB PROP	1 st VENT	2 nd VENT	AVER. KGS/HR	TOTAL TIME
2 SATS	4975	1312		257.3	5.1
3 SATS	4302	800	212	36.9	27.4

MLI SUMMARY

# Layers	Boiloff (kgs/hr)	Thickness (cm)
15	9.59	1.25
30	4.82	2.54
60	1.45	5.08

Groundrules:
413 km orbit
LH2 Tank Area = 472 m ^2
LO2 Tank Area = 245 m ^2

Figure 3-17. Boiloff Effects During Phasing Have Been Investigated

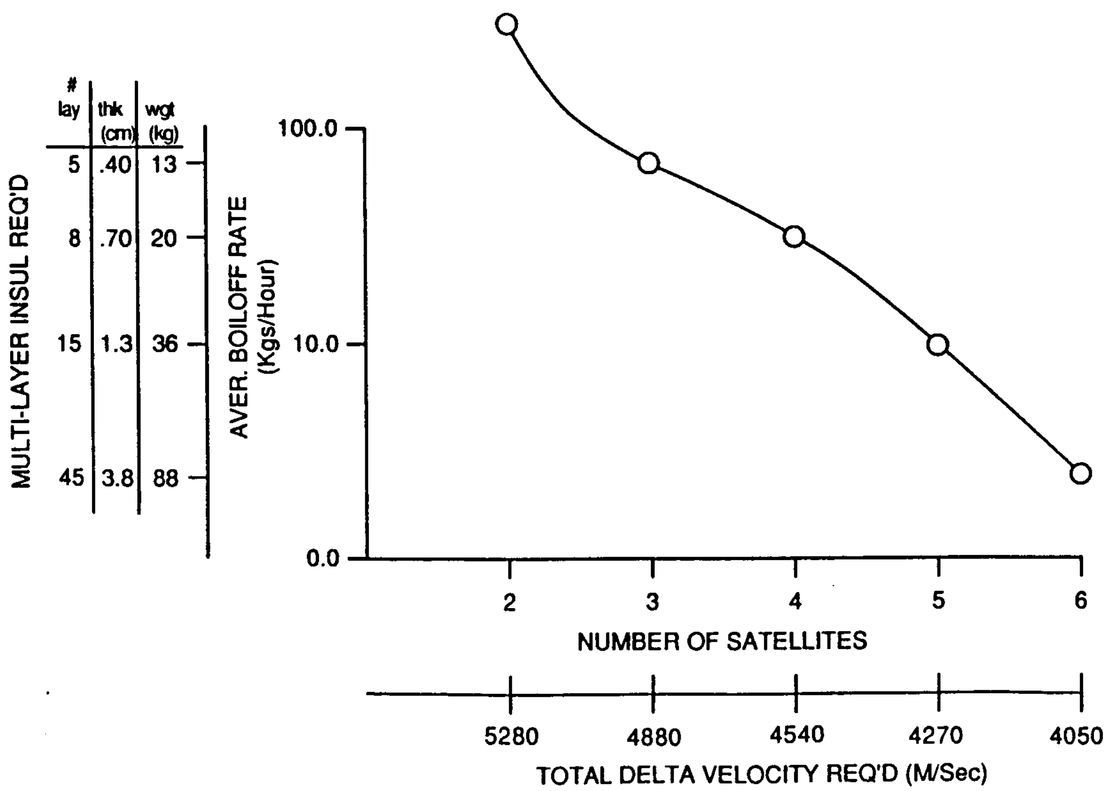


Figure 3-18. Little Insulation Is Required to Attain Very Low Boiloff Rates

variations between the two-satellite and six-satellite cases. As can be seen, even for very low boiloff rates (2.5 kg/hr), the amount of MLI required is only 3.8 cm (45 layers) for a total additional weight of 88 kg. The weight penalty is small enough to be accepted for all missions. Boiloff is therefore probably not a high concern.

3.2.2 SBTC SINGLE PAYLOAD CAPABILITY. The SBTC capability for a single payload delivery will be much larger than any currently available system. The payload delivery capability is a function of altitude and plane change requirements. This is illustrated by two performance examples in Figure 3-19 for a plane change/altitude combination. Figure 3-20 should allow the interpolation of plane change versus payload weight capability of SBTC for circular orbits between 18,520 km and GEO altitudes. The SBTC will also have the performance capability to carry out large interplanetary missions. The advantage of launching from the Station altitude over a ground-launched equivalent vehicle may be seen from the performance plot of C3 vs payload weight in Figure 3-21. The Ground-Based Titan Centaur (GBTC) and SBTC are both shown for comparison. It may also be noted that many of the launch window concerns such as weather and launch site problems are practically nonexistent from Space Station. The excess circular velocity of the SBTC will allow a large variety of final orbits.

As shown in Figure 3-22, orbital parameters may vary widely for a given characteristic velocity. SBTC will be able to take advantage of this should a mission arise requiring an unusual orbit. The complete single payload analysis is given in Appendix B.

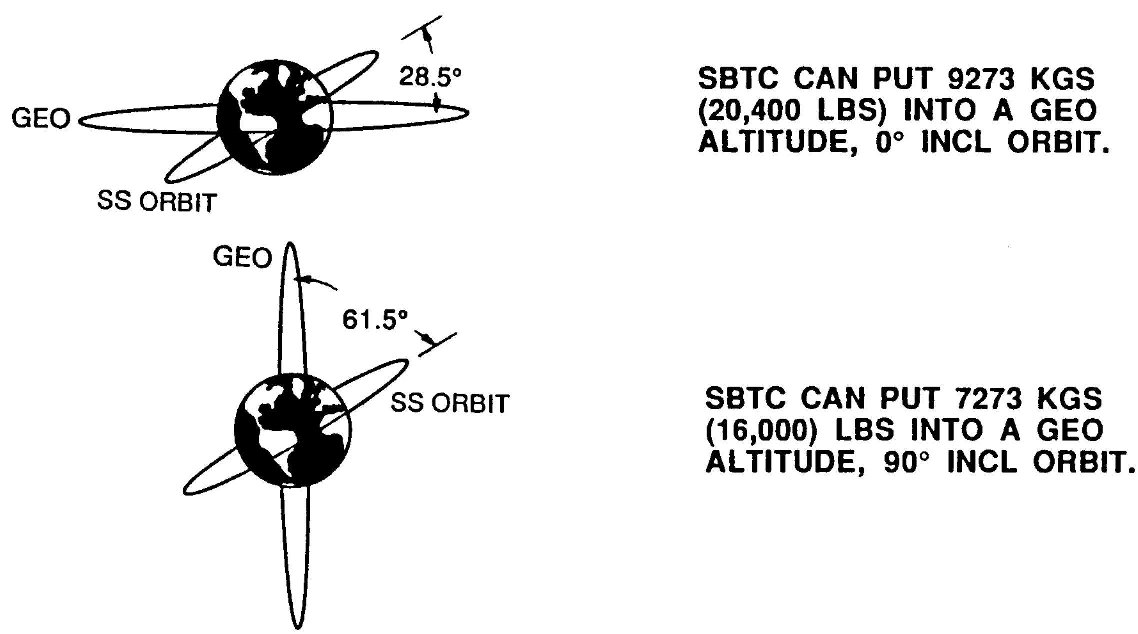


Figure 3-19. The Single Payload Maximum Weight to a Given Altitude Depends on Plane Change Required

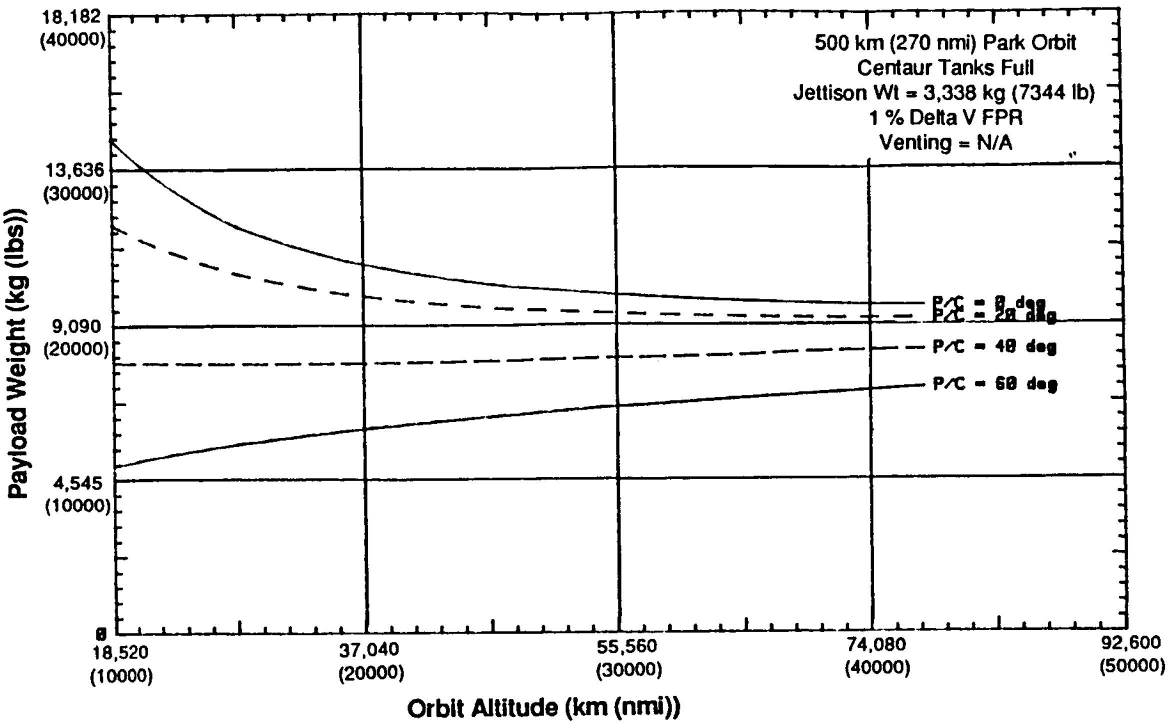


Figure 3-20. The SBTC Will Have a Robust Payload Delivery Capability to Different Orbit Altitudes

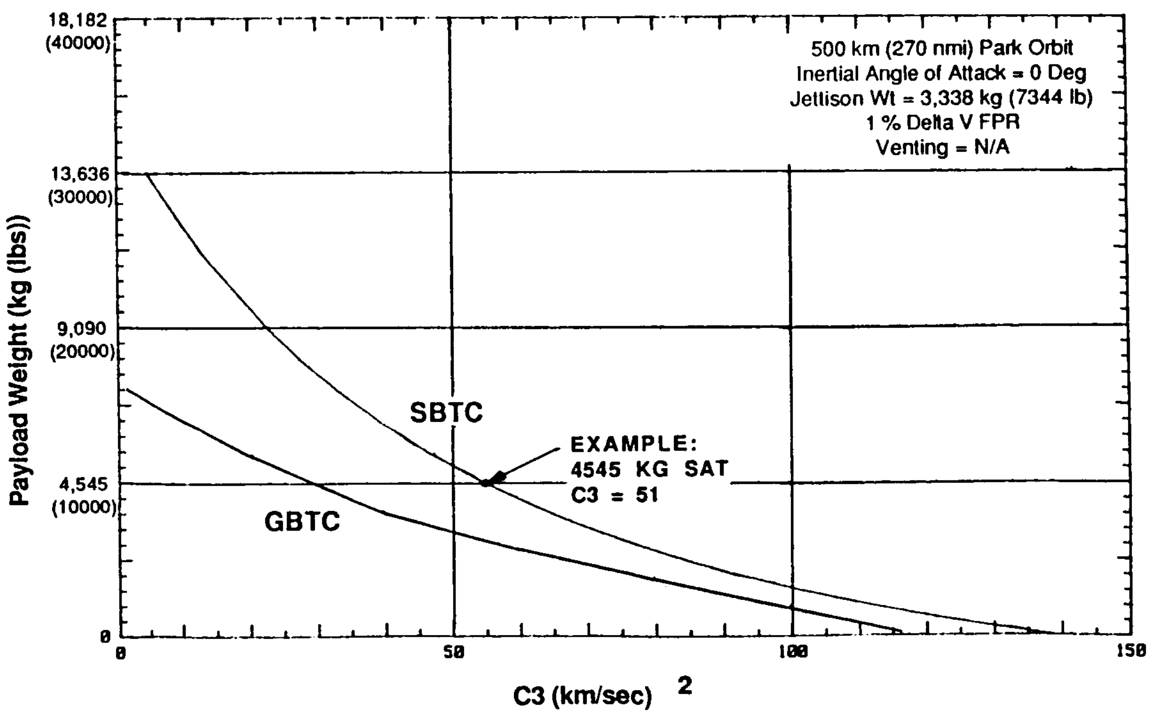


Figure 3-21. Centaur Planetary Mission Capability Increases From the Space Station

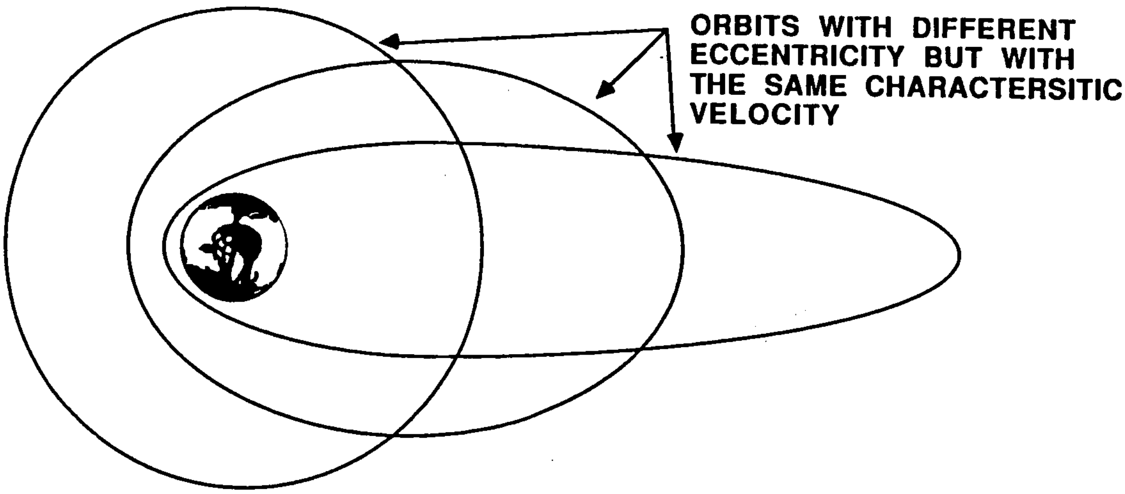
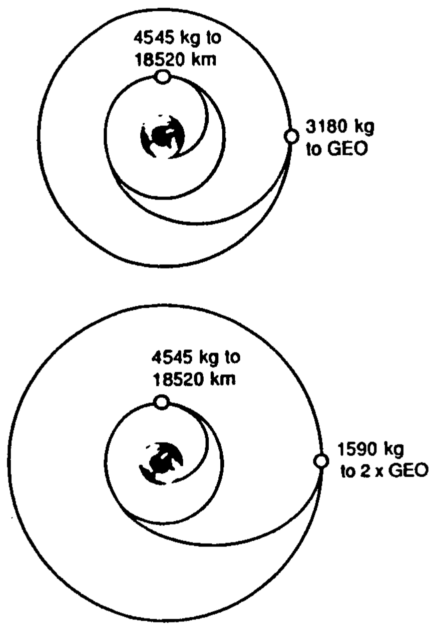


Figure 3-22. Centaur Will Be Capable of Providing a Large Variety of Orbits for a Given Satellite Weight

3.2.3 SBTC TWO SPACECRAFT DELIVERY CAPABILITY. Robust dual-payload capability is one of the benefits of the SBTC. As illustrated in Figure 3-23, the SBTC will be able to deliver different payloads to different altitudes, providing circularization for each. The performance capabilities for delivering a Global Positioning System (GPS) payload at 18,520 km, and another to GEO are shown in Figure 3-24. In addition, the SBTC can deliver different payloads to different inclinations. The performance capabilities for two times GEO altitude at varying inclination deltas are shown in Figure 3-25. These may be used to determine other mission possibilities in a manner similar to those used in the examples. Figures 3-26 and 3-27 illustrate the SBTC capability to perform an interplanetary mission after delivering a 1500-kg payload to a circularized GEO orbit.

The capability to deliver two satellites to orbit, circularize, deploy one, and then provide phasing for the second with circularization when in position is another important benefit of the SBTC. Figure 3-28 shows an example of such a case for two COMSATs to GEO. Additional dual GEO COMSAT performance capabilities are shown in Appendix B for 12-hr transfers at three altitudes (18520 km, GEO, or 2 x GEO) and two boiloff rates (25% and 50% during each coast). The plots show the second satellite capability as a function of first satellite weight and spacing required.



• THE WEIGHT OF THE FIRST SATELLITE DETERMINES THE CAPABILITY FOR THE SECOND FOR A GIVEN DELIVERY ALTITUDE DIFFERENCE.

EXAMPLE 1: IF A 4545 KG S/C IS CIRCULARIZED AT 18,520 KM (GPS ORBIT), A 3,180 KG S/C CAN BE CIRCULARIZED AT GEO.

• ALTITUDE SEPARATION BETWEEN THE S/C AFFECTS TOTAL PAYLOAD CAPABILITY.

EXAMPLE 2: IF 4545KG S/C IS CIRCULARIZED AT 18,520 KM, A 1590 KG S/C CAN BE CIRC AT 2 X GEO.

Figure 3-23. SBTC Can Deploy Two Spacecraft Which Have Different Altitude Delivery Requirements

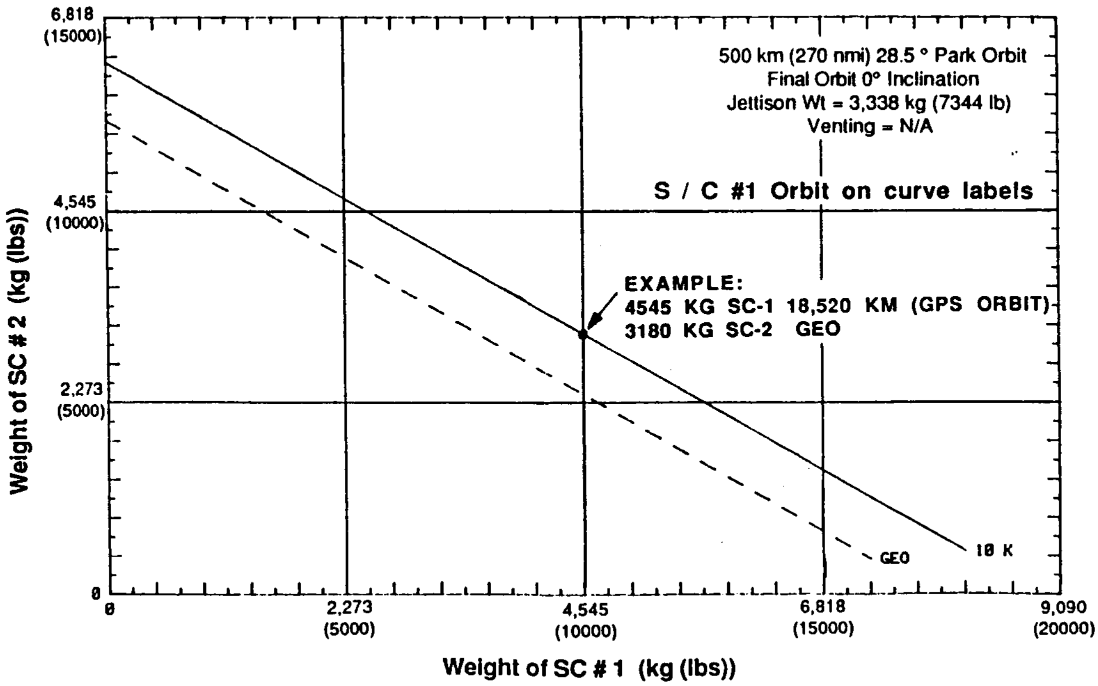


Figure 3-24. SBTC Can Deliver One Spacecraft to 18520 km and Another to GEO

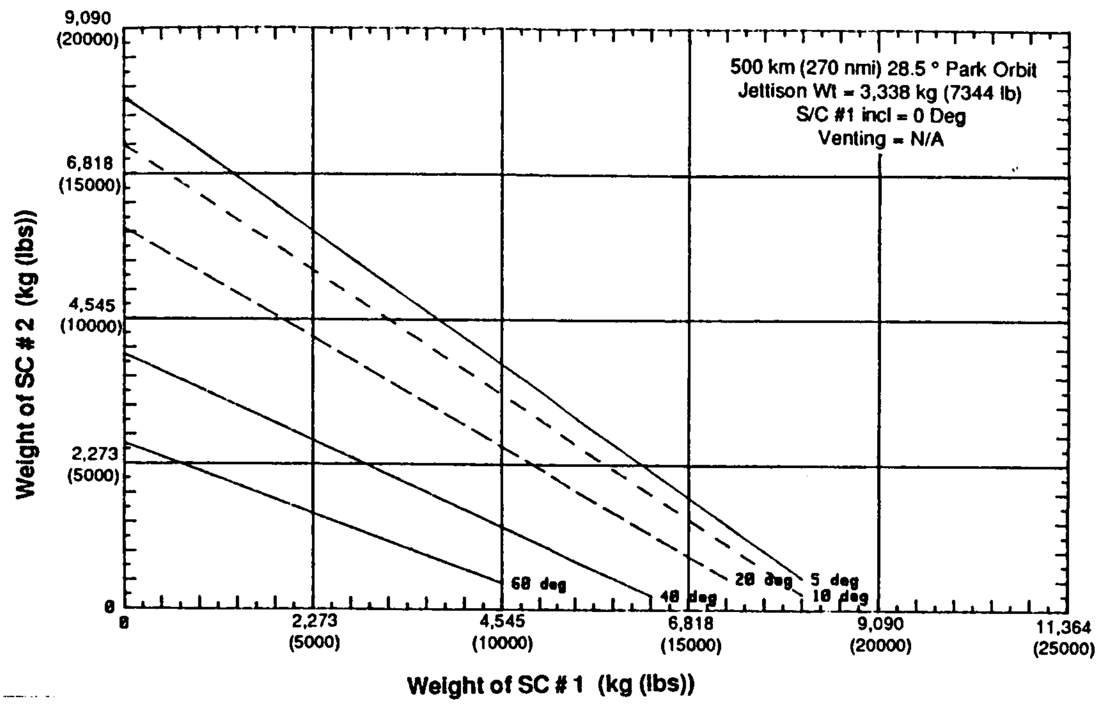
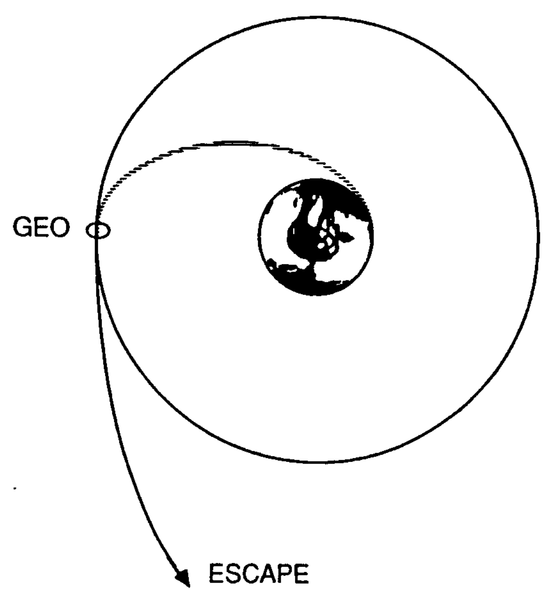


Figure 3-25. SBTC Can Deliver Two Spacecraft to 2 x GEO at Different Inclination Angles



THE CENTAUR WILL HAVE THE CAPABILITY TO PLACE A SATELLITE INTO GEOSYNCHRONOUS ORBIT AND STILL HAVE ENOUGH PERFORMANCE TO PERFORM AN ESCAPE MISSION.

FOR EXAMPLE, THE CENTAUR WOULD LAUNCH FROM THE COP, CIRCULARIZE AT 0° INCLINATION GEO ORBIT AND DEPLOY A 1500 KG SPACECRAFT. IT WOULD THEN PERFORM AN EARTH ESCAPE BURN TO PROPEL A 1,818 KG SATELLITE AT A C3 OF +10.0.

Figure 3-26. Centaur Performance From the Space Station GEO Plus Escape Delivery Mission

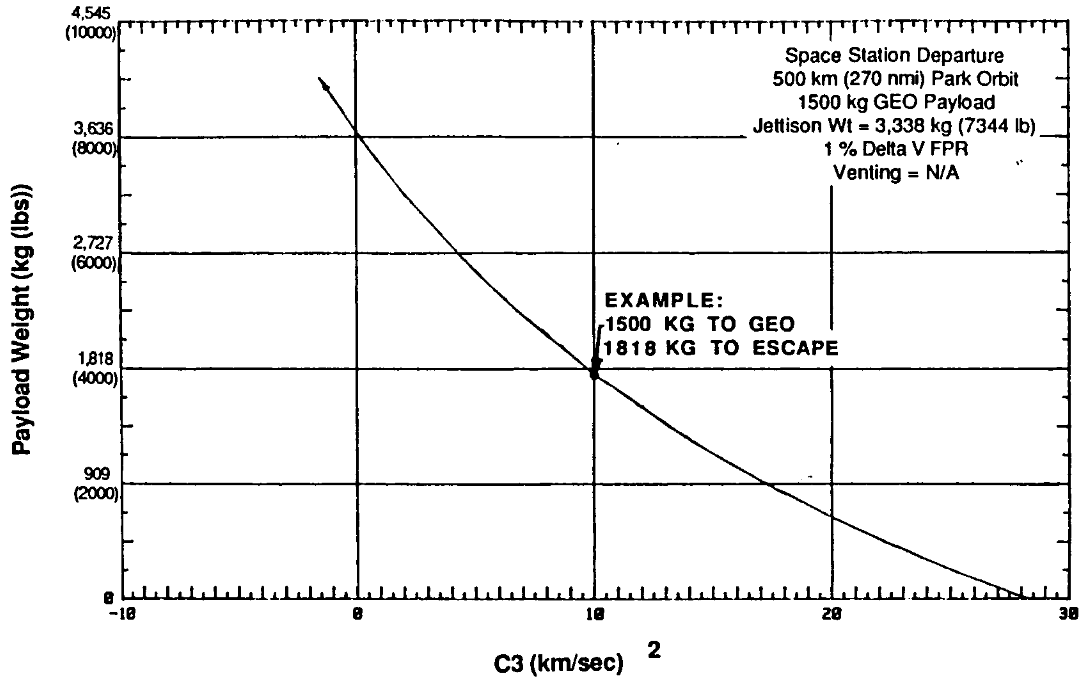
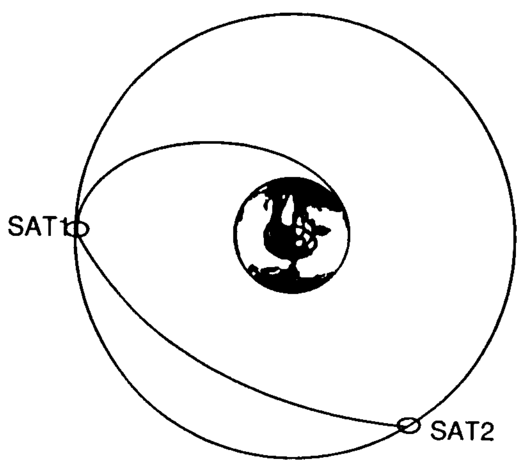


Figure 3-27. SBTC Could Perform a Planetary Mission Even After Delivery of a 1500 kg Payload to GEO

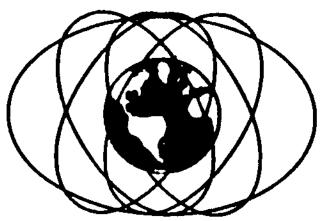


THE CENTAUR WILL HAVE THE CAPABILITY TO PLACE TWO SATELLITES INTO GEOSYNC ORBIT AT DIFFERENT PHASE ANGLES.

FOR EXAMPLE, THE CENTAUR COULD LAUNCH FROM THE COP, CIRCULARIZE AT 0° INCLINATION GEO ORBIT AND DEPLOY A 1818 KG SPACECRAFT. IT COULD THEN PERFORM A NON-HOHMAN TRANSFER TO PLACE A 2410 KG SATELLITE PHASED 120° AWAY IN 12 HRS WITH 25% PROP BOILOFF.

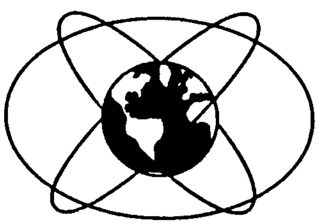
Figure 3-28. Centaur Can Deliver Two COMSATs to GEO and Provide Spacing

3.2.4 <u>SBTC MULTI-PAYLOAD DELIVERY CAPABILITY</u>. The placement of GPS satellites is another area where the SBTC performance could be valuable. The present GPS configuration calls for six planes of satellites with three satellites spaced 120 apart in each plane. The SBTC could deliver the three satellites of one plane while providing the spacing and circularization for each of the satellites (Figure 3-29 shows three satellites delivered). SBTC could also place four GPS satellites (including the active spare) in every other orbit. A proposed improved configuration for the GPS constellation calls for three orbit planes with six satellites per orbit (Figure 3-30). The SBTC could provide up to five of the satellites for one orbit, again while providing the spacing and circularization required for each.



THE PRESENT GPS CONFIGURATION CALLS FOR SIX PLANES OF SATELLITES WITH THREE SATELLITES SPACED 120° APART. THE SBTC COULD SUPPLY ALL 3 SATELLITES FOR ONE ORBIT (WITH SPACING) WITH A SINGLE LAUNCH.

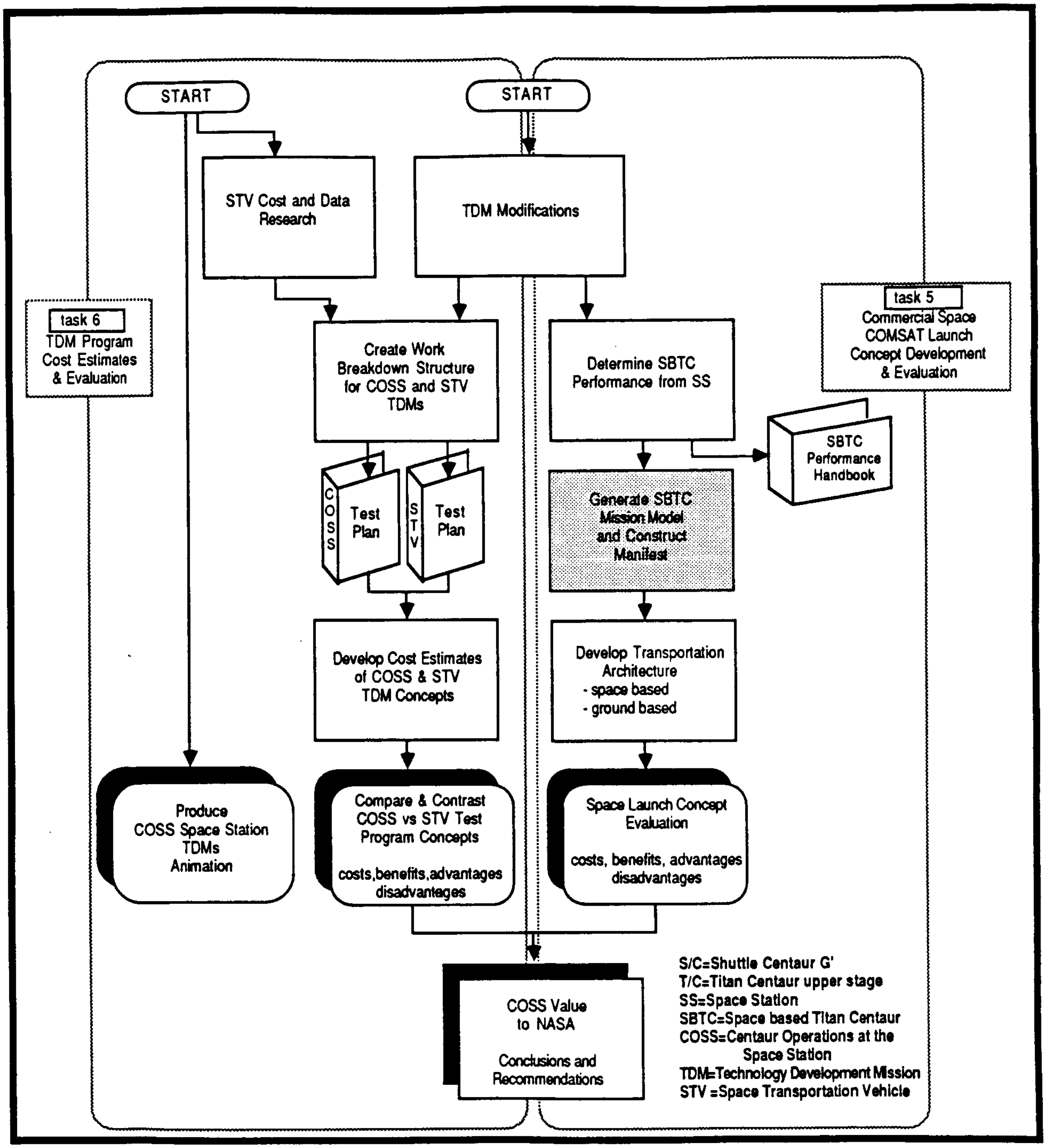
Figure 3-29. The SBTC Could Deliver Three or Four Satellites to the Current Orbits



A PROPOSED IMPROVED CONFIGURATION CONSISTS OF 3 ORBIT PLANES 120° APART WITH 6 SATELLITES AT 60° INTERVALS ON EACH. THE SBTC COULD SUPPLY UP TO 5 OF THE SATELLITES FOR ONE PLANE (WITH SPACING) WITH A SINGLE LAUNCH.

Figure 3-30. SBTC Could Deliver Up to Five Satellites for a New Constellation

3.3 GENERATE SBTC MISSION MODEL AND CONSTRUCT MANIFEST



3.3 GENERATE SBTC MISSION MODEL AND CONSTRUCT MANIFEST

This section assumes that the precursor COSS TDM program is completed, and hypothesizes routine operation begins for an ongoing COMSAT launch program.

With the T/C upper stage operational at the Space Station several real missions could be performed. Because the SBTC can substantially increase its payload capability when launched from orbit, manifesting of multiple COMSATs becomes an important factor in driving down cost.

This section presents the mission capture methodology to conduct an evaluation of SBTC space launches versus conventional ground launching of the same missions. As Figure 3-31 shows, mission informations are collected to create an SBTC mission model. Once the mission model is defined, the vehicles (both boosters and upper stages) performance and costs can be analyzed, from which a preliminary manifesting recommendation and total operations costs analysis are made. This allows direct cost comparison of SBTC-launched vehicles (with the appropriate logistics support) against ground-launched vehicles. Costs will certainly be one factor used in assessing the feasibility of commercial space operations.

3.3.1 MISSION MODEL. An SBTC mission model specifically created for the SBTC includes mission informations from four different sources. These are as follows.

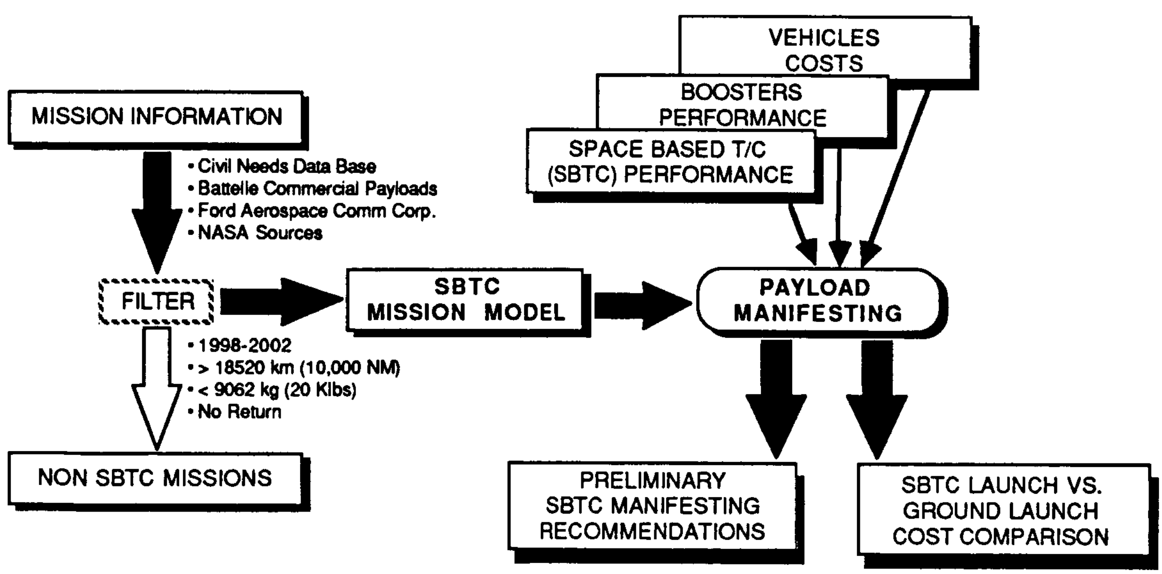


Figure 3-31. SBTC Mission Model Activity Resulted in Manifesting Recommendations and a Space Launch Versus Ground Launch Cost Comparison

3.3.1.1 The Civil Needs Data Base Version 3. This is the mission model utilized by the Space Transportation Architecture Study (STAS), which consists of NASA and civil space missions data. There are four different options in the Civil Needs Data Base (CNDB), with launch requirements ranging from "business as usual" for Option I to "ambitious growth" for Option IV. The "normal growth" Option II GEO and escape missions are used to make up part of the SBTC mission model.

3.3.1.2 The Outside Users Payload Model 1986. This model is also known as the Battelle Commercial Mission Model, which consists of commercial and international payloads. There are two options: a Low Model consisting of normal payload schedule requirements, and a High Model with more demanding launch requirements. Information from the High Model is used in the SBTC mission model.

3.3.1.3 Ford Aerospace Communications Corporation Communications Satellites. There are three Generic 1995 genre advanced COMSATs to be included in the analysis. All three would be deployed at GEO. They were provided by Ford Aerospace Communications Corporation (FACC) and are based on the current RCA FS-1300, the Hughes HS-393, and the Ford Evolutionary Communications Platform (ECP). They will hereinafter be referred to by their baselines, i.e., FS-1300, HS-393 and ECP.

Because these are advanced COMSATs, there are no launch dates assigned as yet. However, according to FACC, they will be available for deployment in the post-1995 time period.

The FS-1300 Hybrid and the HS-393 Spinner are shown in Figures 3-32 and 3-33 with informations pertaining to each satellite. These include the begin-of-life (BOL) mass, the spin table mass, stowed volume, telemetry, power, and stabilization features. The FS-1300 is three-axis stabilized, while the HS-393 is spin stabilized. The same type of information is given for the ECP in Figure 3-34; the ECP is three-axis stabilized and has a much larger mass of 7,583 kg (16,700 lb).

3.3.1.4 NASA Planetary Missions. Informations from this source are NASA planetary missions. Three missions are chosen for the mission model, including the Near Earth Asteroid Rendezvous (NEAR), the Uranus Flyby/Uranus Probe, and the Mars Surface Probe missions.

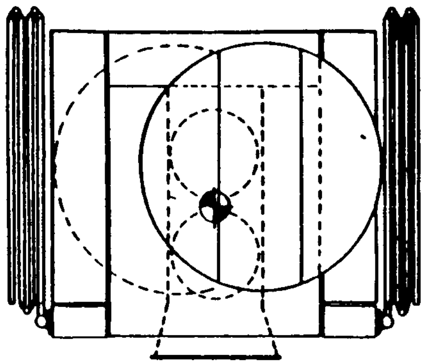
Several screening procedures are performed to select the candidate SBTC missions. The selection eliminates non-SBTC payloads using the following criteria:

- Only missions in the time period 1998 and 2002 will be considered; this is consistent with the time period expected for the FACC COMSAT.

- Missions with payload destinations below 18,520 km (10,000 n.mi GPS orbit) and payload weights above 9,062 kg (20,000 lb) are excluded, so that payload weights are consistent with SBTC performance.

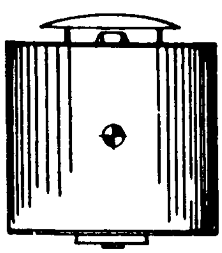
- Servicing and return required missions are also filtered out; these are not missions for which the SBTC was designed.

The resulting SBTC mission model is tabulated in Table 3-8. While there are numerous missions the SBTC can perform, those listed here include only planetary and geosynchronous destined payloads. Therefore, the list is by no means exhaustive,



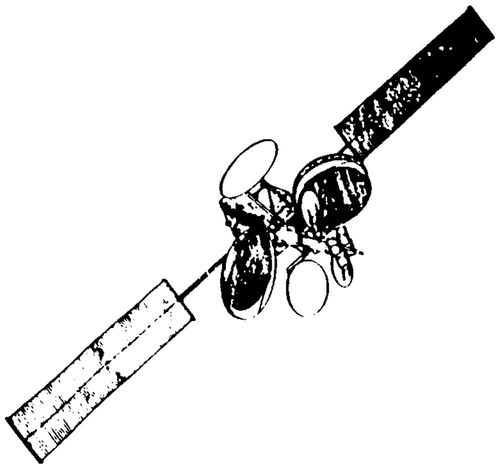
MASS (B.O.L.) (includes 2 grapple fixtures)	1540 kgs (3388 lbs)
SPIN TABLE MASS	NONE
VOLUME (stowed)	4m L x 3m W x 3m H (13'L x 9.8'W x 9.8'H)
TELEMETRY	1.2 kbps
COMMANDING	NONE
POWER (transfer)	350 watts
THERMAL	.1 rpm ROLL
PURGE GAS	NONE
STABILIZATION	3 - AXIS
LAUNCH DATE	POST 1995

Figure 3-32. The FACC FS-1300 Baseline Hybrid Communications Satellite Is Three-Axis Stabilized



MASS (B.O.L.) (includes 2 grapple fixtures)	1377 kgs (3029 lbs)
SPIN TABLE MASS	140 kgs (308 lbs)
VOLUME (stowed)	3.64 m Dia x 3.35 m H (11.8'Dia X 10.9'H)
TELEMETRY	1.2 kbps
COMMANDING	NONE
POWER (transfer)	300 watts
THERMAL	.1 rpm ROLL
PURGE GAS	NONE
STABILIZATION	SPINNER
LAUNCH DATE	POST 1995

Figure 3-33. The HS-393 Baseline Communications Satellite is Spin Stabilized



MASS (B.O.L.) (includes 2 grapple fixtures)	7583 kgs (16,682 lbs)
SPIN TABLE MASS	NONE
VOLUME (stowed)	5m L x 4.5m W x 6m H (16.3'L x 14.8'W x 19.5'H)
TELEMETRY	1.2 kbps
COMMANDING	NONE
POWER (transfer)	600 watts
THERMAL	.1 rpm ROLL
PURGE GAS	NONE
STABILIZATION	3 - AXIS
LAUNCH DATE	POST 1995

Figure 3-34. FACC ECP Is Three-Axis Stabilized

it only represents typical SBTC class payloads. For each of the missions, the payload weight, dimensions, destination, and the flight schedule requirements are given. In addition, for those missions (especially the planetary spacecraft) which might be delayed to later deployment dates for one reason or another, the original dates are also given in Table 3-8.

3.3.2 MANIFESTING OPTIONS. An extensive SBTC performance data base was developed, as discussed in Section 3.2. These analyses provide SBTC performance to deliver a single payload on a single mission, and multiple payloads to different destinations on a single mission.

The data base supported four classes of missions. The first is the business-as-usual scenario where the upper stage provides deployment energy to a single payload (Class 1, Figure 3-35). In this case, because the vehicle begins its mission from the Space Station, a payload capability of up to 9,240 kg (20,400 lb) is realized. A UPA for single payloads provides standard interface between the payload and the SBTC.

For multiple payload deployment, the MPA allows two to six spacecraft to be manifested on the same flight. For the two spacecraft case, both can be placed at GEO locations (Class 2, see Figure 3-35), or the first can be deployed at GEO, after which the second payload is given enough energy to escape (Class 3, see Figure 3-36). For either of these cases, a variety of payload weight combinations are possible. Allowable performance combinations can be interpolated from the data found in Section 3.2.4, and in Appendix B of this report. The Class 3 example lists the FACC satellites as Spacecraft No. 1 and No. 2. Both require GEO. The first weighs about 1,540 kg (2,400 lb), and the second 4,530 kg (10,000 lb). As the chart for separation capability of two satellites placed at GEO shows (Appendix B, page 17), any spacing of these weights would be well within SBTC capability.

Table 3-8. The SBTC Mission Model Was Abbreviated to GPS, GEO, and Planetary Missions Only

MISSION NAME *	P/L WT., Kg (Klbs)	L, m (ft)	D, m (ft)	MISSION ALT / INC	ORIGINAL YEAR	FLIGHT SCHEDULE '98	'99	'00	'01	'02
CRAF	5580 (12.32)	9.1 (30)	4.6 (15)	C3 = 13	1993	1				
ESA PLANETARY	1130 (2.5)	3.1 (10)	4.6 (15)	Escape					1	
CASSINI (SATURN ORB/TITAN FLYBY)	3990 (8.8)	9.1 (30)	4.6 (15)	C3 = 28	1996	1				
LUN. COMM. RELAY	1130 (2.5)	4.6 (15)	4.6 (15)	C3 = -3				1		
ISTP WIND	680 (1.5)	4.6 (15)	4.6 (15)	C3 = -3	1992		1			
LUNAR GEOSCIENCE ORBITER (LGO)	1150 (2.53)	9.1 (30)	4.6 (15)	C3 = -3	1994		1			
TDRSS	2220 (4.9)	5.9 (19)	3.0 (10)	Geosync.		1		1		1
GOES	397 (0.875)	2.4 (8)	4.6 (15)	Geosync.				1	1	
KA BAND	1360 (3.0)	4.6 (15)	4.6 (15)	Geosync.		1		1		
KU BAND	997 (2.2)	4.6 (15)	4.6 (15)	Geosync.		1				
MOBILE SAT B	3990 (8.8)	6.1 (20)	3.9 (13)	Geosync.	1997	1				
GMS -4, -5, -6		3.9 (13)	3.9 (13)	Geosync.			1			
TROPICAL EARTH RESOURCES SAT. (TERS)	3990 (0.880) 750 (1.65)	1.8 (6)	4.6 (15)	Geosync.				1		
GPS - w/o AKM - w/ AKM	1130 (2.5) 1860 (4.1)	4.6 (15)	3.0 (10)	20186 Km (10900 NM)@55		4	4	4	4	4
FORD AEROSPACE COMM. SATS										
FS - 1300	1540 (3.4)	3.1 (10)	3.1 (10)	Geosync.						
EVOLUTIONARY COMM. PLATFORM (ECP)	8490 (18.7)	-	-	Geosync.						
HAC HS - 393	1540 (3.4)	3.1 (10)	3.1 (10)	Geosync.						

* MISSION INFORMATIONS FROM CIVIL NEEDS DATA BASE (CNDB), VERSION 3, 7-16-87 EXCEPT FORD AEROSPACE COMMUNICATION SATS.

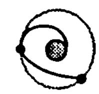
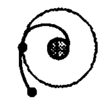
• DATA ARE NOT EXHAUSTIVE

Table 3-8. The SBTC Mission Model Was Abbreviated to GPS, GEO, and Planetary Missions Only, Contd

MISSION NAME *	P/L WT., Kg (Klbs)	L, m (ft)	D, m (ft)	MISSION ALT / INC	ORIGINAL YEAR	FLIGHT SCHEDULE '98	'99	'00	'01	'02
INTELSAT VII	1360 (3.0)	-	-	Geosync.			2	1	2	
SBS/MCI F/O	1360 (3.0)	-	-	Geosync.				1	1	
FLTSATCOM F/O	2040 (4.5)	-	-	Geosync.	1997	1				
ANIK E F/O (CANADA)	1430 (3.15)	-	-	Geosync.				1	1	
PALAPA F/O (INDONESIA)	680 (1.5)	-	-	Geosync.					1	
EUTELSAT II F/O	990 (2.2)	-	-	Geosync.			1	1	1	
TDF F/O (FRANCE)	1220 (2.7)	-	-	Geosync.				1		
CS -4A, -4B (NASDA, JAPAN)	1990 (4.4)	-	-	Geosync.		1	1			
GOES -L, -M	1060 (2.33)	-	-	Geosync.				1		
CHINA METSAT	680 (1.5)	-	-	Geosync.					1	
GMS - X	910 (2.0)	-	-	Geosync.		1				
MARS SURFACE PROBE (MSP) (MIN. C3 = 10)	1200 (2.65)	2.4 (8)	2.1 (7)	C3 = 10				1		
URANUS FLYBY/URANUS PROBE (UFUP) (MIN. C3 = 49)	4300 (9.48)	4.2 (14)	4.2 (14)	C3 = 49				1		
NEAR EARTH ASTEROID RENDEZVOUS (NEAR) (MIN. C3 = 50)	2300 (5.07)	2.4 (8)	2.1 (7)	C3 = 50	1994	1				

* MISSION INFORMATIONS FROM "OUTSIDE USERS PAYLOAD MODEL", BATTELLE, 10-86
PLANETARY MISSION DATA FROM "NASA'S PLANETARY EXPLORATION CORE PROGRAM", J. PORTER, 9-9-84

• DATA ARE NOT EXHAUSTIVE

	S/C #1 WT, Kg (Klbs)	S/C #2 WT, Kg (Klbs)	S/C #3 WT, Kg (Klbs)	COMMENTS	MISSION PROFILE
CLASS 1 SINGLE P/L	9240 (20.4)	-	-	Geosync.	
CLASS 2 TWO P/Ls	1540 (3.4)	4530 (10.0)	0	Both @ Geosync.	
CLASS 3 TWO P/Ls	1500 (3.3)	~4530 (10.0)	0	S/C#1 @ Geosync. S/C#2 to lunar orbit	
CLASS 4 MULTIPLE P/Ls	Up to six GPSs at 1130 Kg (2.5Klbs)			All to 20,186 Km (10900NM) @ 55 deg.	

SBTC PAYLOAD CAPABILITY FROM S.S.

Figure 3-35. The SBTC Performance Data Base Supported Four Classes of Mission Manifesting

The multiple (more than two) spacecraft case for this study was limited to several combinations of three spacecraft, and the GPS four spacecraft deployment (Class 4). The SBTC can deploy up to six GPSs (each weighing 1,130 kg or 2,500 lb) on the same mission. However, as shown in the mission model (Table 3-8), there are four GPS missions required a year. Therefore, without additional informations, it is assumed only four GPS payloads would be deployed in any 1 year.

Once the SBTC manifesting classes were defined, actual manifesting of payloads was then performed. The next section discusses the methodology, ground rules, and assumptions pertaining to payload manifesting.

3.3.3 PRELIMINARY MANIFESTING RECOMMENDATIONS. A constraint of our study was to preserve FACC satellites throughout our manifesting. Based on SBTC performance and multiple payload packaging constraints, six other spacecraft were chosen to form manifest combinations with the FACC satellites, as shown in Figure 3-36. These include two planetary spacecraft (the Lunar Geoscience Orbiter and the Mars Surface Probe), while the others are all geosynchronous satellites. Of all these, the planetary spacecraft are the smallest in weight. However, the Lunar Geoscience Orbiter has the largest dimension, giving it the lowest density.

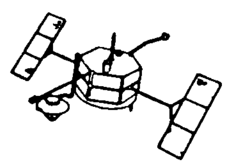
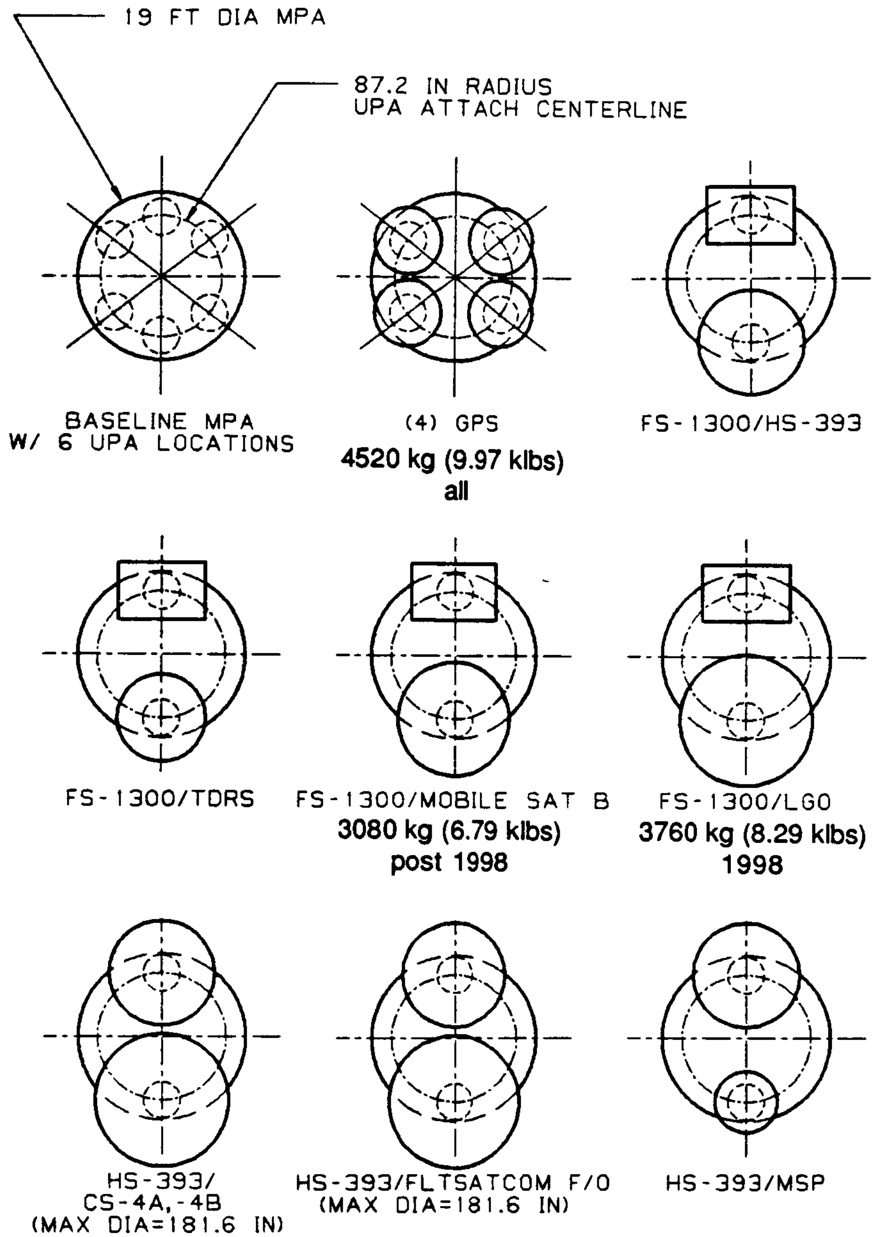
MISSION PAYLOAD		BOL WT., Kg (Klbs)	Dimension @ Launch (LxD), m (Ft)
TDRSS		2220 (4.9)	5.9x3.0 (19x10)
MOBILE SAT B	N/A	3990 (8.8)	6.1x3.9 (20x13)
LUNAR GEOSCIENCE ORBITER (LGO)	N/A	1150 (2.53)	9.1x4.6 (30x15)
CS -4A, -4B	N/A	1990 (4.4)	N/A
FLTSATCOM F/O		2040 (4.5)	N/A
MARS SURFACE PROBE (MSP)	N/A	1200 (2.65)	2.4x2.1 (8x7)

Figure 3-36. Six Communication and Planetary Payloads Were Chosen in Addition to FACC Satellites for Manifest Recommendations

Preliminary manifesting recommendations are shown in Figure 3-37. Although payload compatibility must be studied, these manifestings are representative of the SBTC's capability and of the types of payloads it can capture. It is pointed out that the manifests in Figure 3-37 are based on the SBTC performance only, and do not reflect launch cost considerations as yet.

For each combination of the manifested group, the year they are to be flown and the combined spacecraft weights are given. The single ECP satellite requires a dedicated SBTC and therefore remains separate. Each of the other FACC COMSATs, FS-1300 and HAC HS-393, is either flown combined with one another, or with other spacecraft. The four GPSs required per year can be deployed by the SBTC on a single mission, so four are combined here. Apogee kick motors (AKMs) are included when the GPSs are deployed from the ground based Medium Launch Vehicles (MLV).

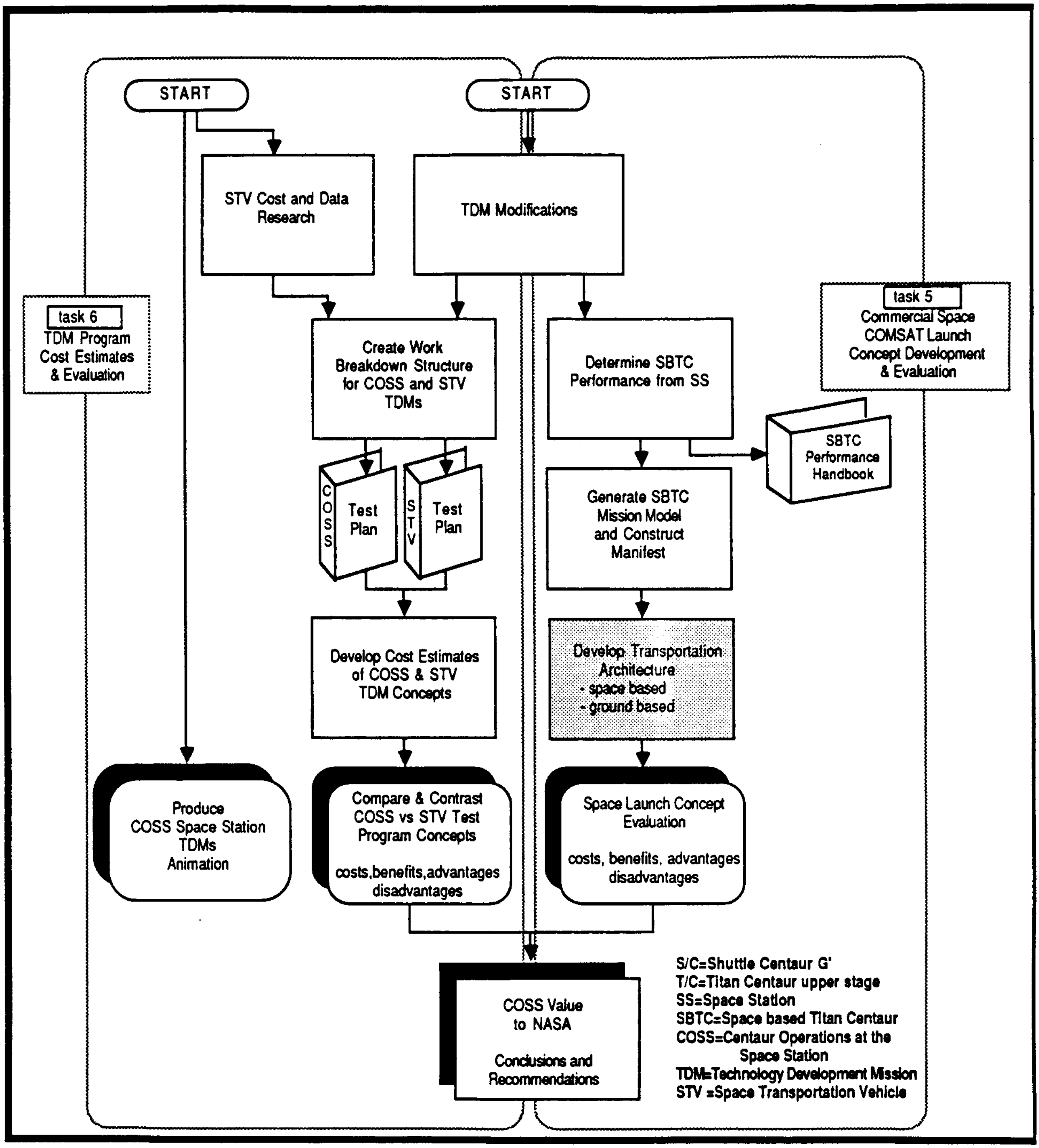


MPC allows integration of either:
- (4) payloads up to 2.19 m (86.2 in) dia each
- (3) payloads up to 2.95 m (116.1 in) dia each
- (2) payloads up to 4.12 m (162.4 in) dia each

KEY: Total payload weight kg (klbs)
Flight Years

Figure 3-37. Preliminary SBTC Manifesting Recommendations Did Not Co-Manifest More than Two Spacecraft Except for GPS

3.4 TRANSPORTATION ARCHITECTURE



3.4 TRANSPORTATION ARCHITECTURE

An important cost component of a commercial space launch operations program would be the logistics flights to deliver propellant and payloads to the Space Station. Three logistics options for SBTC operations were developed. Option I employs currently operational launch vehicles. Option II utilizes the Advanced Launch System (ALS), and Option III baselines Shuttle-C.

For each of the three options, the launch vehicles are also analyzed independently of space logistics as candidate boosters for a COMSAT ground launch program; in competition with SBTC. For example, in Option II, the cost of using ALS in a conventional ground-launched satellite placement mission is compared to space launch costs using SBTC with ALS logistics support.

3.4.1 LOGISTICS SUPPORT GROUND RULES. The following is a list of logistics support ground rules:

- Shuttle-C and ALS can launch multiple payloads on a single flight.
- Unmanned logistics boosters (TIV, STS-C, ALS) deploy logistics payloads from a 100 n.mi. noncircular staging orbit. Then attached solid rocket motors transfer the logistics payloads to a circular orbit close to Space Station. The OMV deploys from Space Station, rendezvous with the payloads, and ferries them to Space Station or COP.
- When the Space Shuttle delivers the spacecraft and the SBTC, it performs proximity and docking operations without the help of the OMV.
- When propellant is delivered to the COP, the OMV performs docking of new tanks to the COP. OMV then disposes of the empty tanks by deorbiting them
- After payload spacecraft are integrated to the SBTC, the OMV takes the SBTC-CISS Spacecraft assembly to the COP for tanking.
- The SBTC is tanked at the COP. Pre-launch activities and launch operations lead up to COP ejection and coast to a safe distance, then ignition of SBTC engines.
- The SBTC deploys its payloads and is expended.

3.4.2 GROUND-BASED BOOSTER PERFORMANCE AND OPTION GROUPS. Ground-based boosters, including the Space Shuttle, will be used to supply payloads, propellant, and spare parts to the Space Station as logistics support to the STV. The specific vehicles employed will depend on the match between their performance and cost effectiveness to the weight and volume of support packages.

The COP is the tanking and launch facility for COSS space transportation concept. The delivery of the propellant to supply the COP could occur by several different means. Our original COSS concept called for the Shuttle to deliver oxygen tanks on two different flights directly to the COP. The hydrogen tank would be delivered by a Titan IV expendable rocket.

For continuous space operations, the ALS or STS-C rocket systems could be used to deliver the entire cryogenic storage assembly and propellant mass in one flight. Since both ALS and STS-C systems, at the time of analysis, planned for suborbital, expendable flight profiles, the delivery would require an upper stage to move the propellant tank to Space Station altitude. Once circularized, the OMV would maneuver the tank to the COP and provide disposal of the empty tank set.

Similarly, delivery of the SBTC/CCA to the Space Station may be carried out by one of several approaches. The original COSS program approach was to have the unfueled CCA delivered by the Shuttle. The Shuttle would dock at the Space Station and its RMS and the Space Station MRMS would remove the SBTC from the cargo bay and stow it in the hangar.

The empty CCA could be carried up using the Shuttle-C cargo vehicle or the ALS launch vehicle. The vehicles would have sufficient performance to not only bring up the CCA, but several COMSATs. The SPF would store the COMSATs until they are integrated to the SBTC in the Centaur hangar in preparation for launch. The CCA would be delivered dry because we believe payload mating would be done at the Space Station where EVA service will be available if needed. We also believe no cryogenic propellants will be allowed on the station. The use of the ALS or STS-C launch vehicles would require an upper stage which would deliver the payload to station altitude at a safe separation distance. The OMV would fly to the vehicle and pick up the CCA and spacecraft and bring them to the station.

For an operational program, as many logistics payloads as possible are launched on a single flight to efficiently utilize the LV capacity. These may include components for a planned SBTC launch and components for subsequent SBTC launches. For example, maximum amount of propellant should be delivered to the COP so that a single propellant flight can support up to three SBTC missions. Note, however, this study assumes new logistics delivery flights to support each specific candidate SBTC mission. In our analysis, propellant costs were apportioned, but remaining capability on a logistics LV was not carried over to benefit the next mission. This one-at-a-time approach was done to simplify manifesting, but may have inflated space launch costs.

The payload performance to LEO, and to Space Station altitude of ground-based boosters is tabulated in Table 3-9. Also shown in the table are the types of logistics payloads for which each vehicle is best suited. Three launch vehicle options were analyzed for SBTC. These are as follows.

3.4.2.1 Option I. Current vehicles (see Figure 3-48). This category includes the Space Shuttle, MLV, Atlas/Centaur, and the Titan IV. An anomaly for this option is the introduction of Shuttle C for propellant delivery. This was done because the Titan IV is so much less cost efficient for resupplying the 45.5 kg (100 klb) COP propellant tank set baselined for an operational program. Current vehicles were also evaluated as COMSAT ground launch systems, in competition with SBTC. The Space Shuttle was not allowed in the ground launch competition because of present uncertainties in launch policy and manifest availability.

Table 3-9. The Same Ground Launch Vehicles Employed for SBTC Logistics Support Were Used as COMSAT Launch Competition for SBTC

	VEHICLE	PAYLOAD TO ORBIT, DUE EAST, Kg (Klbs.) 186.8 KM	(100 N.M.)	500 KM	(270 N.M.)	CANDIDATE PAYLOAD CARGO
CURRENT VEHICLES OPTION	TITAN IV	17700	(39.0)	16800	(37.0)	S/C
	MLV	5030	(11.1)	1820	(4.01) GTO*	GPS
	ATLAS-CENTAUR (A/C)**	6530	(14.4)	6120	(13.5)	S/C
	SHUTTLE ORBITER	27400	(60.5)***	18400	(45.0)****	S/C, dry SBTC
ALS OPTION	ALS-3 SRM	11700	(25.9)	7240	(16.0)	S/C, dry SBTC
	ALS-6 SRM	25100	(55.3)	20600	(45.4)	S/C, dry SBTC, propellant
	ALS-9 SRM	37400	(82.5)	32900	(72.6)	S/C, dry SBTC
	ALS-12 SRM	49900	(110)	45400	(100.1)	Propellant
	ALS-FBB	68000	(150)	63500	(140.1)	Propellant
SHUTTLE C OPTION	SHUTTLE C	~49900	(110)	~45300	(100.0)	S/C, dry SBTC, propellant

* GTO = Geosynchronous Transfer Orbit
** BLOCK II, 14' FAIRING
*** PERFORMANCE TO 110 NM, 28.5 DEG.
**** PERFORMANCE TO 220 NM, 28.5 DEG.

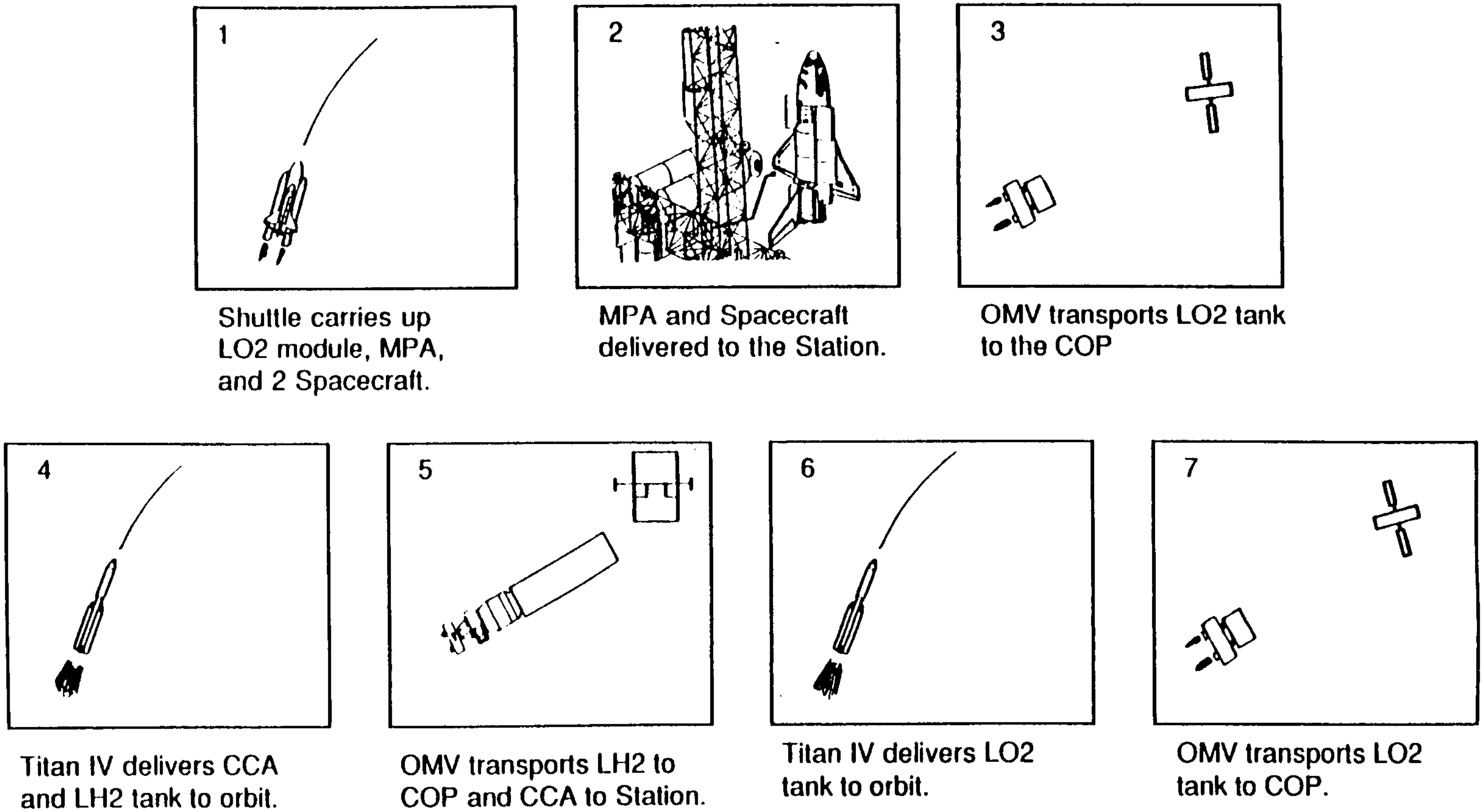


Figure 3-38. SBTC Logistics Option I (Current Vehicles) Employs the Shuttle and a Titan IV ELV

3.4.2.2 Option II. The ALS option is shown in Figure 3-49. ALS supports both spacecraft and propellant deliveries, and is used as a COMSAT ground launch system. As Table 3-9 shows, up to 63,500 kg (140,000 lb) can be boosted to deployment orbit by the largest ALS version.

3.4.2.3 Option III. Figure 3-40 shows this option is identical to Option II, but here the STS-C cargo vehicle replaces the ALS. Similarly, the STS-C can support both Spacecraft and propellant deliveries. Referring to Table 3-9, up to 49,900 kg (110 klb) can be placed in deployment orbit by STS-C.

Recall that when employed as COMSAT ground launch systems, ALS, and STS-C deploy their payloads at 100 n.mi. where attached upper stages transfer them to higher energy orbits (GEO, escape, etc.).

3.4.3 GROUND-BASED COMSAT LAUNCH PROGRAM SATELLITE MANIFESTING. Ground-based launch program manifesting used for this study is shown in Table 3-10. The mission payloads are listed in column "Mission," including the four GPS payloads. The launch vehicle capturing the mission in each of the three options is tabulated in the next three columns. It is assumed that each spacecraft is launched singly, except for the four GPS spacecraft on STS-C and ALS boosters. These are all to be flown on the same flight.

Option I shows high usage of the T/C launch , because it is the only available vehicle with adequate performance. (Shuttle is ground ruled out as previously explained). The GPSs are deployed from the MLVs. Option II shows that the ALS-12 SRM version of ALS is the most economical, when taking into account allocation of launch costs. This is expected because this vehicle has very large payload capability. Finally, Option III consists of only the Shuttle C, therefore it is the only booster flown. Note that upper stage performance requirements forced the selection of the Inertial Upper Stage (IUS) and the Centaur G-Prime except for GPS payloads where the Star 48 kick stage was inherent to MLV.

3.4.4 SPACE-BASED COMSAT LAUNCH PROGRAM LOGISTICS MANIFESTING. The space based logistics support vehicles are shown in Table 3-11 and Figures 3-41 and 3-42 for all three Booster Category options. For Option I, both the Space Shuttle and the Titan IV are utilized as Titan IV is the only sensible selection to deliver (LH_2) propellant to the COP. The Shuttle can transfer both SBTC payload spacecraft to the Space Station and (LO_2) propellant logistics to the COP.

For Option II, the 12-SRM ALS version delivers the dry SBTC, its CISS, and spacecraft to the Space Station, and provides all propellant deliveries. For Option III, Shuttle C is the only logistics vehicle, therefore it performs all logistics missions to the Space Station and the COP.

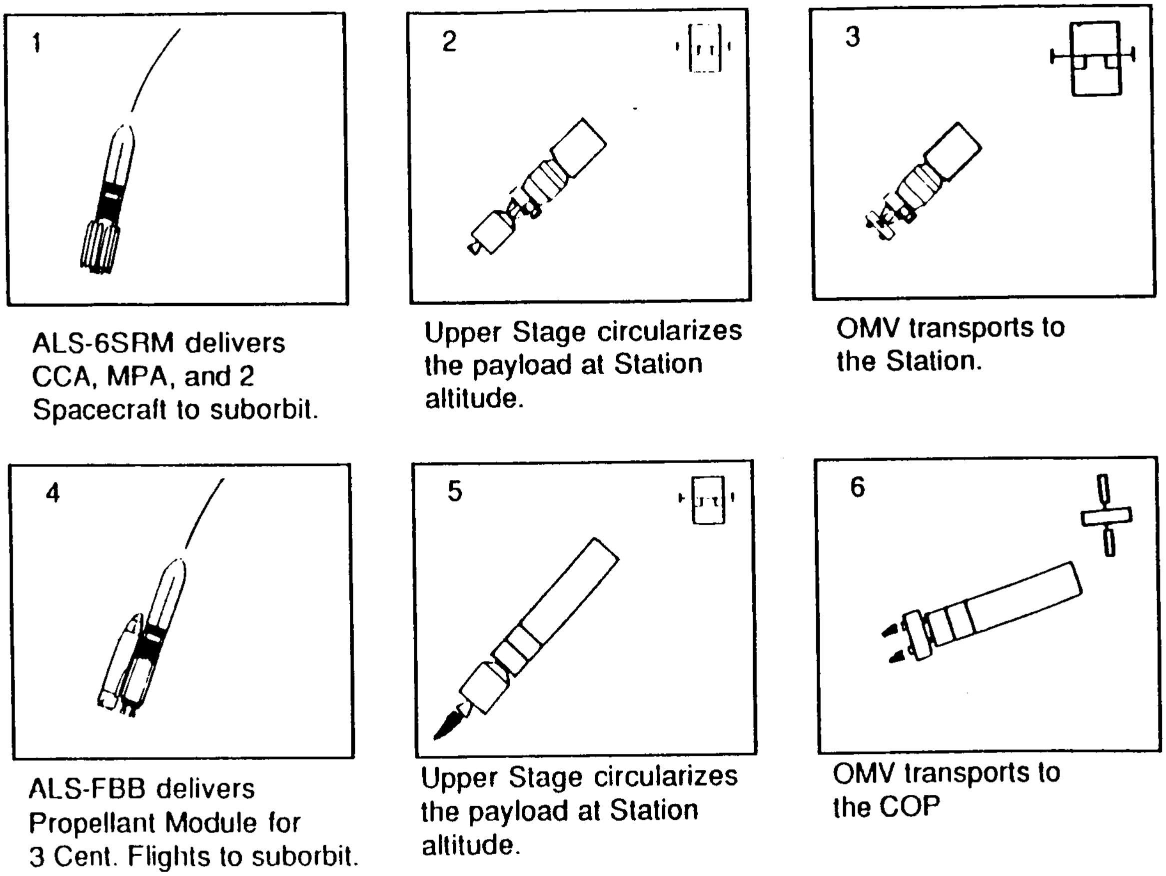


Figure 3-39. SBTC Logistics Option II (ALS) Does Not Employ the Shuttle

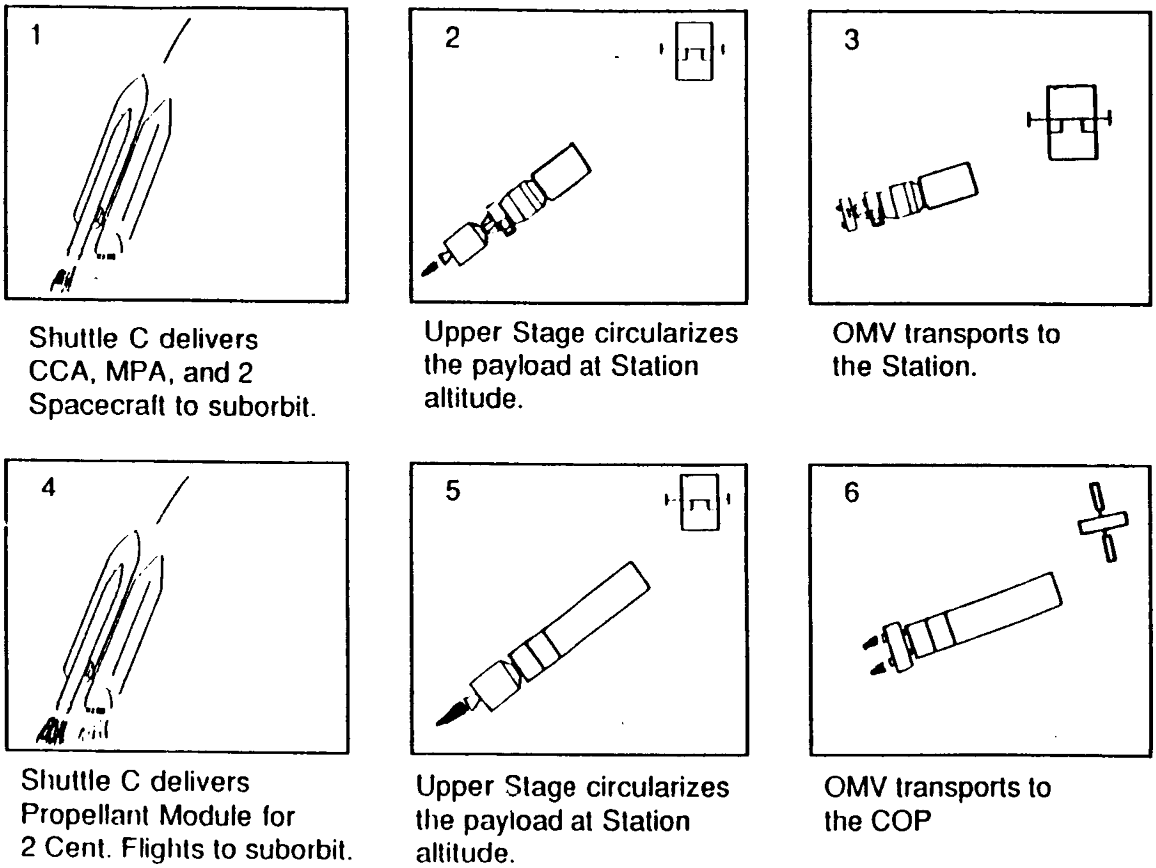


Figure 3-40. SBTC Logistics Option III (STS-C) Does Not Employ the Shuttle

Table 3-10. Ground-Based COMSAT Missions Were Manifested As Single Launches Only

MISSION	LAUNCH VEHICLES OPTION I*	OPTION II	OPTION III
ECP	-	ALS-9SRM/G'	STS-C/G'
FS - 1300	T/C	ALS-6SRM/IUS	STS-C/IUS
HAC HS - 393	T/C	ALS-6SRM/IUS	STS-C/IUS
TDRSS	T/C	ALS-6SRM/G'	STS-C/G'
MOBILE SAT B	T/C	ALS-6SRM/G'	STS-C/G'
LGO	T/C	ALS-6SRM/IUS	STS-C/IUS
CS -4A, -4B	T/C	ALS-6SRM/IUS	STS-C/G'
FLTSATCOM F/O	T/C	ALS-6SRM/IUS	STS-C/G'
MSP	A/C	ALS-6SRM/IUS	STS-C/IUS
4 GPSs	MLVs	ALS-9SRM/ 4(STAR 48)	STS-C/ 4(STAR 48)

* STS ORBITER EXCLUDED

Table 3-11. Space-Based COMSAT Missions Were Manifested As Single and Dual Launches, Except GPS (Four-Launch)

MISSION	LOGISTICS VEHICLES OPTION I	OPTION II*	OPTION III**
ECP	STS, TIV	ALS-12SRM	STS-C
FS - 1300, HAC HS - 393	STS, TIV	ALS-12SRM	STS-C
FS - 1300, TDRSS	STS, TIV	ALS-12SRM	STS-C
FS - 1300, MOBILE SAT B	STS, TIV	ALS-12SRM	STS-C
FS - 1300, LGO	STS, TIV	ALS-12SRM	STS-C
HAC HS, 393 / CS -4A -4B	STS, TIV	ALS-12SRM	STS-C
HAC HS, 393 / FLTSATCOM F/O	STS, TIV	ALS-12SRM	STS-C
HAC HS, 393 / MSP	STS, TIV	ALS-12SRM	STS-C
4 GPSs	STS, TIV	ALS-12SRM	STS-C

* ALS-12SRM DELIVERS PROPELLANT
** STS-C DELIVERS PROPELLANT

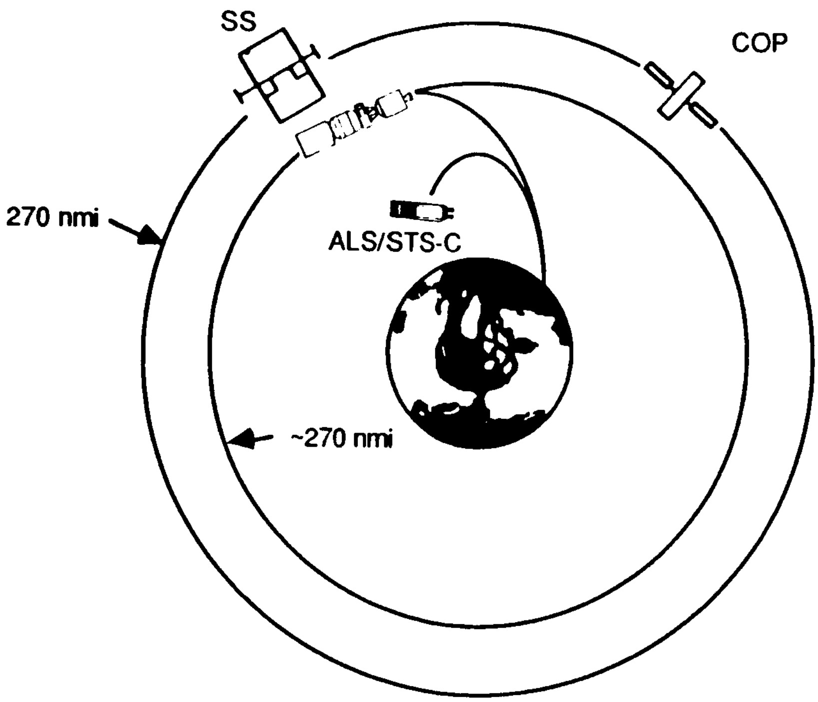


Figure 3-41. Logistics Modules Kicked From Sub-Orbit Booster Deployment to Circular Orbit Near Station

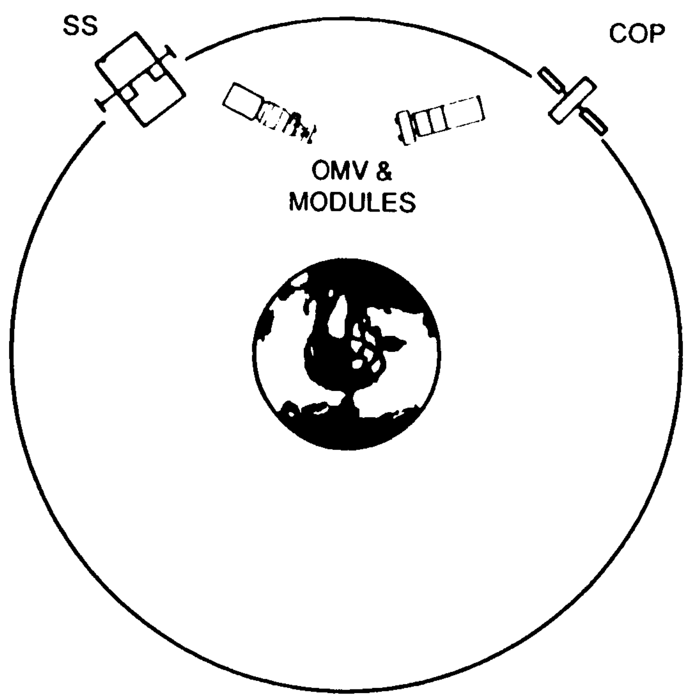
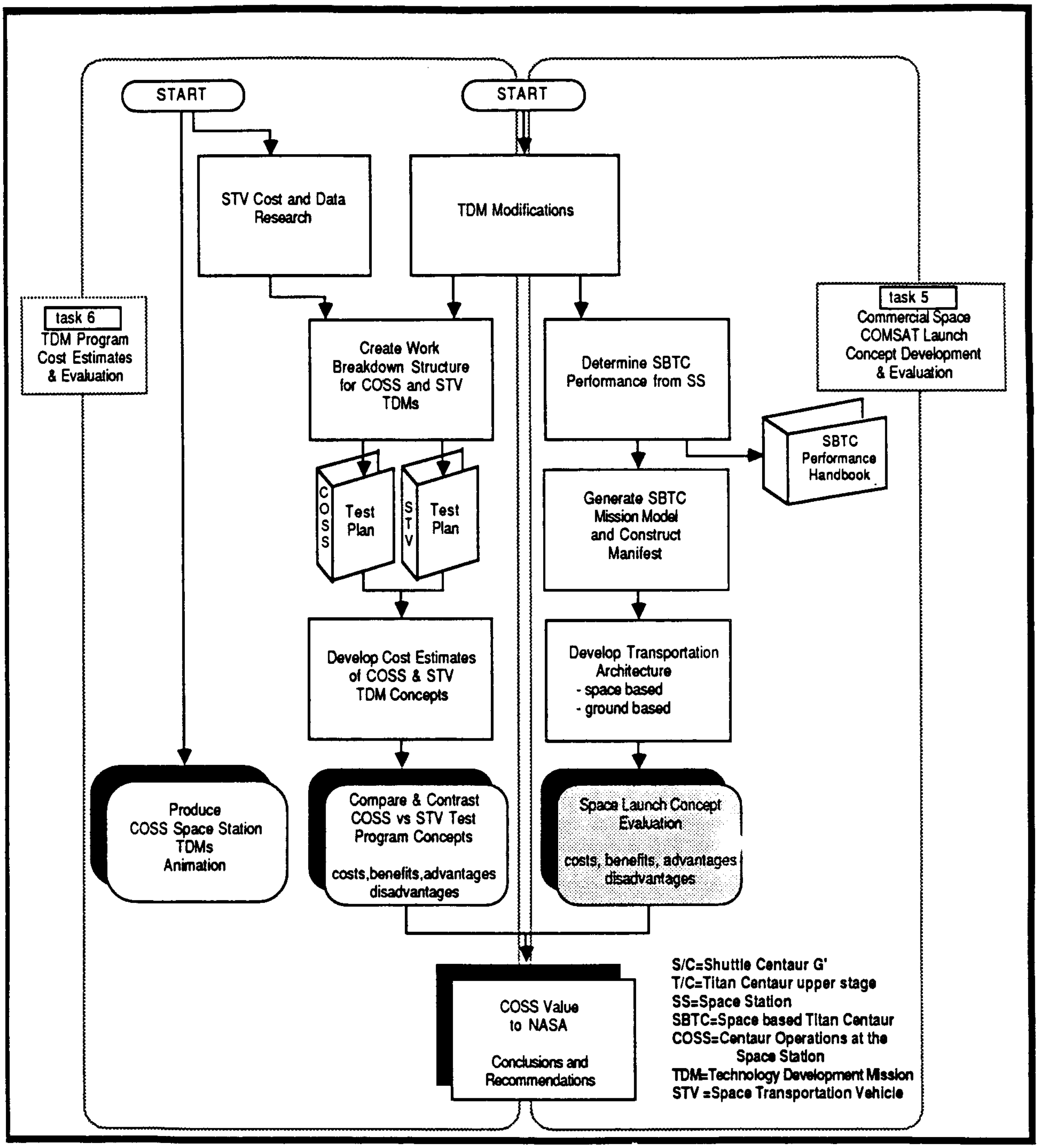


Figure 3-42. OMV Rendezvouses and Ferries Logistics Modules to the Station or COP

3.5 SPACE LAUNCH CONCEPT EVALUATION



3.5 SPACE LAUNCH CONCEPT EVALUATION

If COSS is implemented, the launch experiment part of the operations TDM could be the harbinger of commercial space operations. One of the factors to be considered in determining the feasibility of continuous operations will be the comparison between space launch costs and business-as-usual ground launch costs. This section develops a cost comparison based on the manifesting recommendations of Section 3.3.4.

3.5.1 COST GROUND RULES AND COMPONENT COST ESTIMATES. Because of the nature of this study, all costs presented are rough-order-of-magnitude (ROM) estimates for preliminary planning and trade study comparison purpose only. The average launch costs used are assumed to reflect maximum capabilities at Eastern Test Range (ETR); these launch rate capabilities are recommended in several studies such as the STAS and the LTCSF. The ALS-E (expendable GDSS ALS version) and ALS-FBB (partially recoverable GDSS ALS version) missions are based on current configuration design to deliver the payload to sub-orbit destinations. Pricing options for the Space Shuttle issued by the Congressional Budget Office have been used to estimate the launch costs for partial STS cargo bay usage. The operations costs utilized here may vary due to the future configuration changes of ALS, Shuttle-C, and the resupply tanker.

Table 3-12 states the specific ground rules and assumptions used to develop operations costs for this study. Table 3-13 shows the component cost estimates, including current vehicle operations costs, and Table 3-14 shows labor estimates for IVA, EVA and ground support.

Table 3-12. These Operations Cost Ground Rules and Assumptions Were Used

- All costs are in constant year 1987 dollars
- All estimates include 10% fee and exclude management reserve and government support
- Preliminary planning estimates for trade study and comparison purpose only
- Propellant storage capability available in COP is from 100 Klbs to 150 Klbs
- Average launch costs are based on the following flight rates:

  ALS - E at 35 flts/year
  ALS / FBB at 27 flts/year
  Shuttle Orbiter and Shuttle Derived Vehicle at 16 flts/year
  Titan IV and Titan Centaur at 12 flts/year
  100 Klb Propellant Resupply Tanker at 8 flts/year
  140 Klb Propellant Resupply Tanker at 6 flts/year
  IUS at 12 flts/year
  MLV at 4 flts/year
- Avionics for solid rocket motor cost is 7.2 \$M, payload adapter cost is .8 \$M
- All costs for the TDMs and Scars are excluded
- SBOMV costs are based on NASA ground rules for the Space Transportation Architecture Study (STAS)
- IVA costs and EVA costs are based on GD OTV study
- No costs for NASA mission manifest to Shuttle Orbiter
- Standard multiple payload adapters are available
- Cost difference between multiple and single P/L manifest to the SS is negligible
- Mass guaging device (0-g) and Liquid Acquisition device are government furnished equipments (GFE)
- The percent weight or percent fairing volume, whichever is larger, of the booster that is allocated to a payload is used to calculate the individual payload's launch cost

Table 3-13. These Vehicle Component and Launch Service Cost Estimates Were Used

VEHICLE, COMPONENT AND LAUNCH SERVICE	RATE CAPABILITY	AVE. FLIGHT COST (87 $M)	RATIONALE/ REFERENCE
Atlas/Centaur	N/A	N/A	Commercial A/C
Titan IV	12 /year	84.2	STAS Adj.
MLV	4 /year	35.3	STAS Adj.
Titan Centaur	12 /year	36.2	STAS
SBTC	12 /year	37.7	STAS-Mod. to fit TDM CISS
STS Orbiter	16 /year	113.3	Pricing Options for STS
SDV	16 /year	88.4	STAS-Max. Capability
ALS-E w/3 SRM	N/A	N/A	ALS-E LCC*
ALS-E w/6 SRM	N/A	N/A	ALS-E LCC*
ALS-E w/9 SRM	N/A	N/A	ALS-E LCC*
ALS-E w/12 SRM	N/A	N/A	ALS-E LCC*
ALS / FBB	N/A	N/A	ALS-FBB LCC*
IUS	12 /year	32.7	STAS Adj.
100 Klb Propellant Resupply Tanker	8 /year	7.0	LTCSF Study
140 Klb Propellant Resupply Tanker	6 /year	8.4	LTCSF Study
P/L Adapter-Single	SUBSTANTIAL	0.1	ROM Estimate
P/L Adapter-Multiple	SUBSTANTIAL	0.5	ROM Estimate
Star 37XFP	1	1.4	Morton Thiokol ROM Quote
Star 48B	1	1.7	Morton Thiokol ROM Quote
Star 75	1	3.3	Morton Thiokol ROM Quote
OMV	SUBSTANTIAL	3.9	STAS Adj.
CISS	-	3.1	S/C CISS Refurbishment
T/C Space Launch Service (Addition to SS)	1	2.5	ROM Estimate

* CORE-TO-ORBIT ALS

Table 3-14. The Use of Ground Support for SBTC Operations Was Maximized to Reduce On-Orbit EVA and IVA Charges

GROUND PROCESSING COMMON TO ALS AND SBTC	MAN-HOURS	PROCESSING UNIQUE TO SBTC	MAN-HOURS IVA	MAN-HOURS GND	TOTAL COST
AVIONICS SYSTEM CHECKS	2000	STATION MONITORING			
FLUID SYSTEM VERIFICATION	3000	PAYLOAD INTEG. (2 SC)	30	800	
PNEUMATICS CHECKS	1000	OMV TRANSFER	20	1000	
PROPELLANT MONITORING	500	TANKING	10	4000	
SYSTEM TEMPERATURE MONITORING	500	DEPLOYMENT OPS	5	800	
PAYLOAD INTEGRATION	N/A	MISSION OPS	5	2000	
PAYLOAD HEALTH MONITORING	N/A				$2.5 M
		ONE TIME UNIQUE PREPARATION:			
		JOINT SIMULATIONS		20,000	
		GROUND TRAINING		10,000	
		CREW TRAINING		2,000	
		SYSTEM SIMULATOR		30,000	
					$4.5 M

3.5.2 SPACE LAUNCH VERSUS GROUND LAUNCH COST RESULTS. The cost result for the three options are found on the next three tables (Tables 3-15 to 3-17).

Table 3-15. Space Launch Proved More Expensive Than Current Vehicle Ground Launches (Option I) Under Study Constraints

	OPTION I	LAUNCH COST ('87$M)	
MISSION	GROUND BASED	SPACE BASED + LOGISTICS SUPPORT*	PERCENT CHANGE
ECP	-	308.0	-
FS - 1300 , HAC HS - 393	213.4	304.5	44 %
2(FS - 1300) , HAC HS - 393	320.1	304.5	- 5 %
FS - 1300 , TDRSS	224.1	304.5	36 %
2(FS - 1300) , TDRSS	330.8	304.5	- 8 %
FS - 1300 , MOBILE SAT B	227.9	308.4	35 %
2(FS - 1300) , MOBILE SAT B	334.6	308.4	- 8 %
FS - 1300 , LGO	206.3	273.4	33 %
2(FS - 1300) , LGO	313.0	285.4	- 9 %
HAC HS - 393 , CS -4A -4B	221.6	308.4	39 %
HAC HS - 393 , FLTSATCOM F/O	222.4	308.4	39 %
HAC HS - 393 , MSP	166.5	274.4	65 %
HAC HS - 393 , MSP , FS - 1300	273.2	286.4	5 %
4 GPSs	144.4**	308.4	114 %
6 GPSs	215.0***	308.4	43 %

* INCLUDES GROUND LOGISTICS SUPPORT AND SPACE LAUNCH OPERATIONS
** COST OF LAUNCHING FOUR GPSs ON FOUR MLVs AT $35.3M / LAUNCH
*** COST OF LAUNCHING SIX GPSs ON SIX MLVs AT 35.3 M / LAUNCH

As shown in Table 3-15, both ground-based and space-based costs are recorded, together with the percentage change in cost. For Option I, about 40% difference in launch cost can be seen when comparing ground based LV scenario to SBTC deployment scenario. For the four GPSs to be deployed annually, the increase in cost is up by 114%. The SBTC operations cost is consistently at about $300-310M per SBTC mission of two payloads. The cost driver is propellant resupply transportation. Current vehicle cost per launch is high, and a space operations delta cost is added to this. Since COMSAT would pay nearly the same for a ground launch as for a logistics booster, there can be no contest for single, or even dual launches. Note, however, that the ECP can only be launched by SBTC in Option I.

For Option II, Table 3-16, the space based scenario cost effectiveness becomes evident, with many of the missions costing 10% to 20% less than ground based. It was assumed that ALS-12SRM vehicles carried up all propellants and equipment for the space based scenario. The ground based scenario assumed the ALS configuration most closely sized to the mission was used. Additionally, it was assumed that pricing policy for ALS will parallel that of the Shuttle; i.e., any flight where over 75% of the vehicle capability was required was considered a dedicated flight. The availability of the ALS will make the space basing option a viable, cost-effective solution to COMSAT delivery.

Table 3-16. Space Launch Proved More Cost Effective Than ALS Ground Launch (Option II) Under Study Constraints

MISSION	OPTION II LAUNCH COST ('87$M) GROUND BASED	SPACE BASED + LOGISTICS SUPPORT*	PERCENT CHANGE
ECP	89.8	112.4	+25 %
FS - 1300 / HAC HS - 393	134.4	112.8	-16 %
FS - 1300 / TDRSS	138.9	118.6	-15 %
FS - 1300 / MOBILE SAT B	155.2	119.3	-23 %
FS - 1300 / LGO	121.5	125.7	+3 %
HAC HS - 393 / CS -4A, -4B	137.5	112.8	-18 %
HAC HS - 393 / FLTSATCOM F/O	137.8	112.8	-18 %
HAC HS - 393 / MSP	125.8	112.8	-10 %
GPS	68.9**	118.0**	+71%

* INCLUDES GROUND LOGISTICS SUPPORT AND SPACE LAUNCH OPERATIONS ALS-12SRM PROVIDES PROPELLANT DELIVERY
** COST OF LAUNCHING FOUR GPSs ON ONE ALS

Table 3-17. Space Launch Proved More Expensive Than STS-C Ground Launches Under Study Constraints

MISSION	OPTION III LAUNCH COST ('87$M) GROUND BASED	SPACE BASED + LOGISTICS SUPPORT*	PERCENT INCREASE
ECP	101.1	147.8	46 %
FS - 1300 / HAC HS - 393	125.6	147.4	17 %
FS - 1300 / TDRSS	130.0	156.0	20 %
FS - 1300 / MOBILE SAT B	139.6	156.9	12 %
FS - 1300 / LGO	114.4	165.9	45 %
HAC HS - 393 / CS -4A, -4B	128.7	147.8	15 %
HAC HS - 393 / FLTSATCOM F/O	129.0	147.8	15 %
HAC HS - 393 / MSP	118.1	147.8	25 %
GPS	71.2**	168.7**	137 %

* INCLUDES GROUND LOGISTICS SUPPORT AND SPACE LAUNCH OPERATIONS
** COST OF LAUNCHING FOUR GPSs ON ONE STS-C

For Option III (Table 3-17), cost difference ranges from 12% to 46% for the space based scenario. For the four GPSs, the increase in cost is now 137%. The space based cost is about $150M per SBTC mission of two payloads. In general, the use of the Shuttle C is 50% more cost effective than with the current launch systems, but is 25% more costly than the ALS. As before, a ground launch program is cheaper for the STC-C case.

3.5.3 SUGGESTIONS FOR CONCEPT OPTIMIZATION. Although the space based scenarios show higher initial launch costs, there are opportunities for reducing them:

a. Scavenging propellant could lower space based launch costs. Logistics manifesting in the study supported each launch as a standalone. Our algorhithms charged full flight costs for logistics vehicles loaded beyond 75% of capacity, similar to Shuttle policy. No allowance was made for scavenging propellant by filling unused ELV or Shuttle volume (when weight limits allow) with propellant to benefit a future flight.

b. Reusing empty COP tanks could reduce operating costs. There will be a need for Space Station refuse and waste disposal. Most schemes envision the use of an empty shuttle, or deorbiting refuse tanks using the OMV. Spent COP tanks are deorbited by OMV in our present scenario. If they were first filled with Space Station refuse, OMV use charges could be paid, or at least shared, with Space Station. This would reduce space launch overhead by $2M to $4M per propellant resupply trip, thereby contributing to lower launch fees.

c. The more COMSATs launched on a single SBTC, the lower the individual COMSAT cost. There is certainly a point where the number of multiply launched SBTC COMSATs cannot be matched by a single ground launched vehicle. The requirement for an additional ground launch would drive competing launch costs closer, if not in favor of the space launch. This trade, however, is more complex than it would at first seem. SBTC size/performance growth effects should be studied. Our generalized multiple deployment performance data base would have to be extended. Co-manifesting policies and insurance effects would also have to be considered.

d. Reusing COP tanks could increase SBTC performance. Instead of deorbiting empty COP tanks, they could be adapted to serve as SBTC auxiliary propellant modules. In the limit, this could increase performance to a weight beyond what any anticipated heavy launch vehicle design could boost to parking orbit, either individually, or clustered as in the Augmented ground launch mode described below. However, such giant capacity would be of limited use (probably for Lunar or Planetary base resupply or construction).

3.5.4 AUGMENTED GROUND LAUNCH. A promising area briefly investigated was mission staging, or the "Augmented Ground Launch" mode. In this concept, payloads normally too heavy for a particular ground system could be sent to Space Station undertanked so as not to exceed the booster weight limit. The upper stage would then be ferried to the COP, "topped off," and launched. For this service there would be some delta charge added to the ground launch cost. Also, this mission staging scheme could extend the capability of some smaller ground launch systems if heavy lift systems were not available, or were over manifested.

Table 3-18 shows the results of augmentation for the Titan IV launch vehicle. The total ground based cost reflects the cost of launching the satellites individually on the ELV most closely matching its performance requirements. The space augmented cases assume one Titan IV carries up a multiply manifested Centaur which is partially tanked to allow delivery to the Station COP. Propellant is carried to the COP using the STS-C. The net result is a substantial reduction in cost using a combination of the Titan IV and COP to launch COMSATS. This scenario is illustrated in Figure 3-43.

The benefits of using the ALS In conjunction with the COP are shown in Table 3-19. The main advantage comes from being able to launch two partially filled Centaurs with multiple payloads on a single ALS-12SRM vehicle and top them off at the depot. This is less expensive than launching two Centaurs on separate ALSs. The scenario is shown in Figure 3-44. Table 3-20 illustrates the regions where augmentation is less expensive than dual launch. The 75% rule was again employed, that is a payload requiring more than 75% of the vehicle capability must pay for the whole launch. This rule causes the gaps at 74 klb (74.1%) because the whole vehicle cost is not allocated to such a payload.

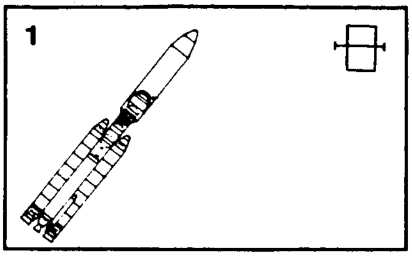
The STS-C effect from use of the COP for on-orbit top-off is shown in Table 3-21. The scenario is the same as that of the ALS and is illustrated in Figure 3-44. The benefit is not as good because the cost of fuel at the depot is greater. The region over which the COP augmented scenario is beneficial is shown in Table 3-22.

The introduction of larger upper stages or an increase in Centaur size could force even single Centaur mission options above the 100 klb capability of ALS-12SRM or STS-C. Again, the COP tanking would make the mission possible and cost effective.

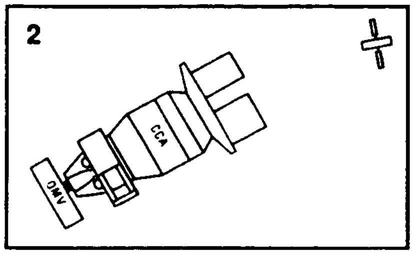
Table 3-18. Use of the COP to Augment Titan IV Capability Saves Money

MISSION	PAYLOAD COMBINATIONS	TOTAL GROUND BASED COST (Requires 2 or 3 Separate ELV Flights)	TOTAL AUGMENTED COST (1 T-IV Flight and some COP Fueling)**	PERCENT CHANGE
CASE A	EVO COMM PLATFORM	- -	$ 135.4 M	- -
CASE B	FS 1300, HS 393, FLTSATCOM*	$ 177.0 M	$ 126.8 M	-28.4%
CASE C	2 TDRS, HS 393*	$ 227.4 M	$ 131.1 M	-42.3%
CASE D	FS 1300, TDRS*	$ 143.2 M	$ 108.5 M	-24.2%

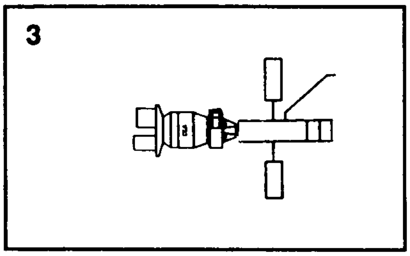
* REQUIRES A NEW, LARGER MPA
** INCLUDES COP USE FEE, OMV FLIGHTS, AND PROPELLANT COSTS



TITAN IV CARRIES UP
PARTIALLY TANKED
CCA WITH PAYLOADS



OMV GRAPPLES CCA AND
MANEUVERS TO COP.



CENTAUR TANKED FULL
AT COP AND IS LAUNCHED.

Figure 3-43. The Use of the COP With the Titan IV Will Improve Titan's Capabilities

Table 3-19. Augmenting ALS With COP Fueling Can Save Money

		Centaur 2 Weight (klbs)							
		58.0	62.0	66.0	70.0	74.0	78.0	82.0	86.0
	50.0	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
	54.0	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
	58.0	XXXX	XXXX	XXXX	XXXX		XXXX	XXXX	XXXX
	62.0	XXXX	XXXX	XXXX	XXXX		XXXX	XXXX	XXXX
	66.0	XXXX	XXXX	XXXX	XXXX		XXXX	XXXX	
Centaur 1	70.0	XXXX	XXXX	XXXX	XXXX		XXXX		
Weight	74.0								
(klbs)	78.0	XXXX	XXXX	XXXX	XXXX				
	82.0	XXXX	XXXX	XXXX					
	86.0	XXXX	XXXX						
	90.0	XXXX							
	94.0								
	98.0								

XXX = Region where CSOD is financially advantageous

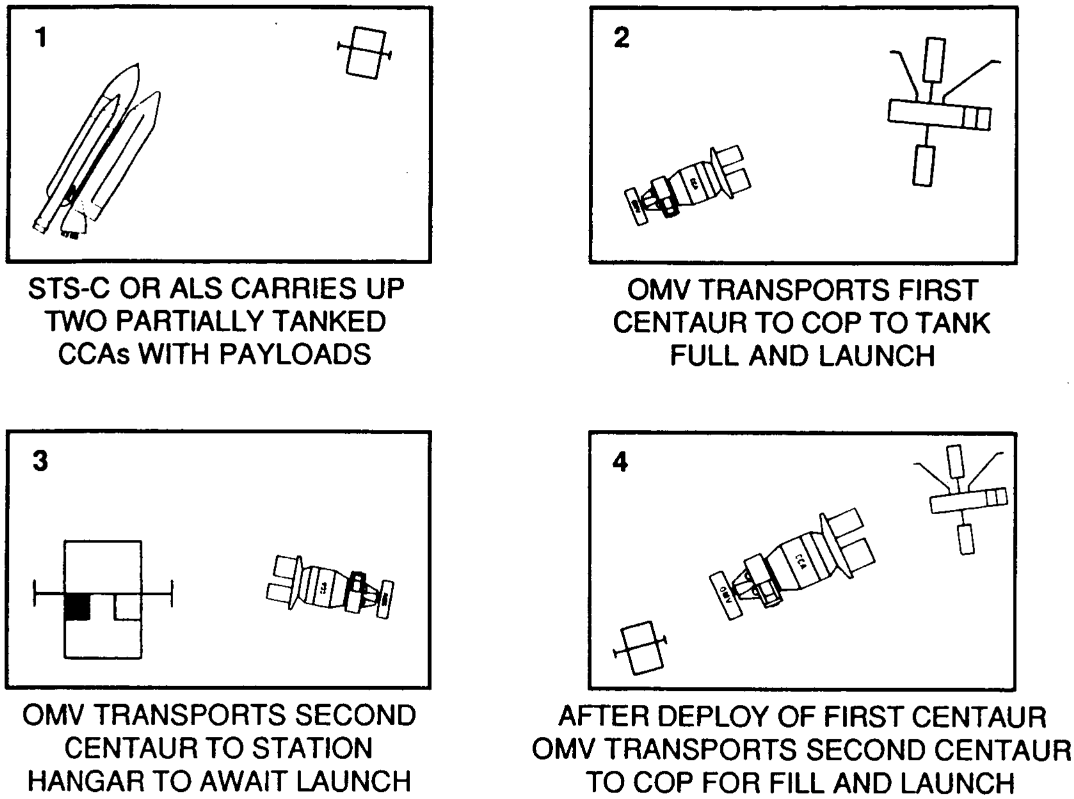


Figure 3-44. Use of the COP Can Reduce Launch Costs and Increase STS-C and ALS Capabilities

Table 3-20. The COP Allows the ALS to Increase Its Capability Per Flight at a Lower Cost

MISSION	PAYLOAD COMBINATIONS	TOTAL GROUND BASED COST (Parts of 2 Separate ALS Flights)	TOTAL AUGMENTED COST (1 ALS-12SRM Flight and some COP Fueling)**	PERCENT CHANGE
CASE A		$ 112.6 M	$ 114.0 M	1.2%
CENTAUR 1	EVO COMM PLATFORM			
CENTAUR 2	FS 1300, HS393, FLTSATCOM*			
CASE B		$ 112.6 M	$ 111.0 M	-1.4%
CENTAUR 1	FS 1300, HS393, FLTSATCOM*			
CENTAUR 2	2 TDRSS, HS393*			
CASE C		$ 116.0 M	$ 114.0 M	-1.7%
CENTAUR 1	5 GPS*			
CENTAUR 2	5 GPS*			
CASE D		$ 116.0 M	$ 117.0 M	0.9%
CENTAUR 1	3 TDRSS*			
CENTAUR 2	3 TDRSS*			
CASE E		$ 109.2 M	$ 98.0 M	-10.3%
CENTAUR 1	FS 1300, HS393			
CENTAUR 2	EVO COMM PLATFORM			
CASE F		$ 102.4 M	$ 82.0 M	-19.9%
CENTAUR 1	FS 1300, TDRSS			
CENTAUR 2	FS 1300, CS -4A			

* REQUIRES A NEW, LARGER MPA
** INCLUDES COP USE FEE, OMV FLIGHTS, AND PROPELLANT COSTS

Table 3-21. The STS-C Could Benefit From COP Augmentation by Maximizing Payload to Orbit

MISSION	PAYLOAD COMBINATIONS	TOTAL GROUND BASED COST (Parts of 2 Separate STS-C Flights)	TOTAL AUGMENTED COST (1 STS-C Flight and some COP Fueling)**	PERCENT CHANGE
CASE A		$ 151.4 M	$ 159.4 M	5.3%
CENTAUR 1	EVO COMM PLATFORM			
CENTAUR 2	FS 1300, HS393, FLTSATCOM*			
CASE B		$ 151.4 M	$ 155.4 M	2.6%
CENTAUR 1	FS 1300, HS393, FLTSATCOM*			
CENTAUR 2	2 TDRSS, HS393*			
CASE C		$ 132.2 M	$ 158.4 M	19.8%
CENTAUR 1	5 GPS*			
CENTAUR 2	5 GPS*			
CASE D		$ 176.8 M	$ 163.4 M	-7.6%
CENTAUR 1	3 TDRSS*			
CENTAUR 2	3 TDRSS*			
CASE E		$ 131.9 M	$ 137.4 M	4.2%
CENTAUR 1	FS 1300, HS393			
CENTAUR 2	EVO COMM PLATFORM			
CASE F		$ 94.7 M	$ 114.4 M	20.8%
CENTAUR 1	FS 1300, TDRSS			
CENTAUR 2	FS 1300, CS -4A			

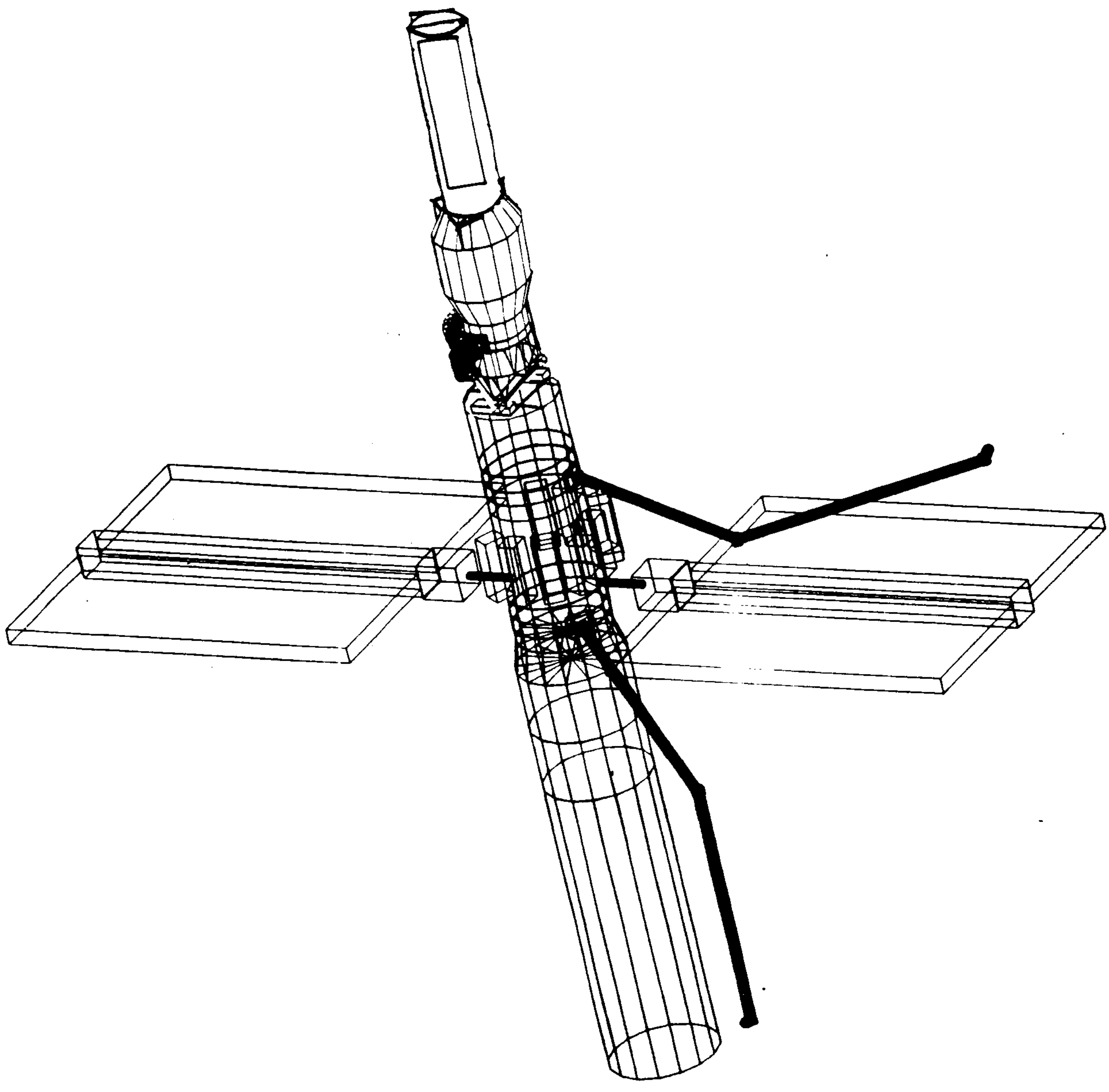
* REQUIRES A NEW, LARGER MPA
** INCLUDES COP USE FEE, OMV FLIGHTS, AND PROPELLANT COSTS

Table 3-22. For Large STS-C/Centaur Flight Weight, It Is Less Expensive to Use the COP for Fuel Top-Off

Centaur Weight (klbs) \ Secondary Payload Weight (klbs)	36.0	38.0	40.0	42.0	44.0	46.0	48.0	50.0
50.0								
54.0								
58.0								
62.0								
66.0								
70.0								
74.0	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
78.0	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
82.0	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
86.0	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX	XXXX
90.0								
94.0								
98.0								

XXX = Region where CSOD is financially advantageous

SECTION 4
TASK 6 — PROGRAM COST ANALYSIS



This section determines the value to NASA of the planned COSS TDM program.

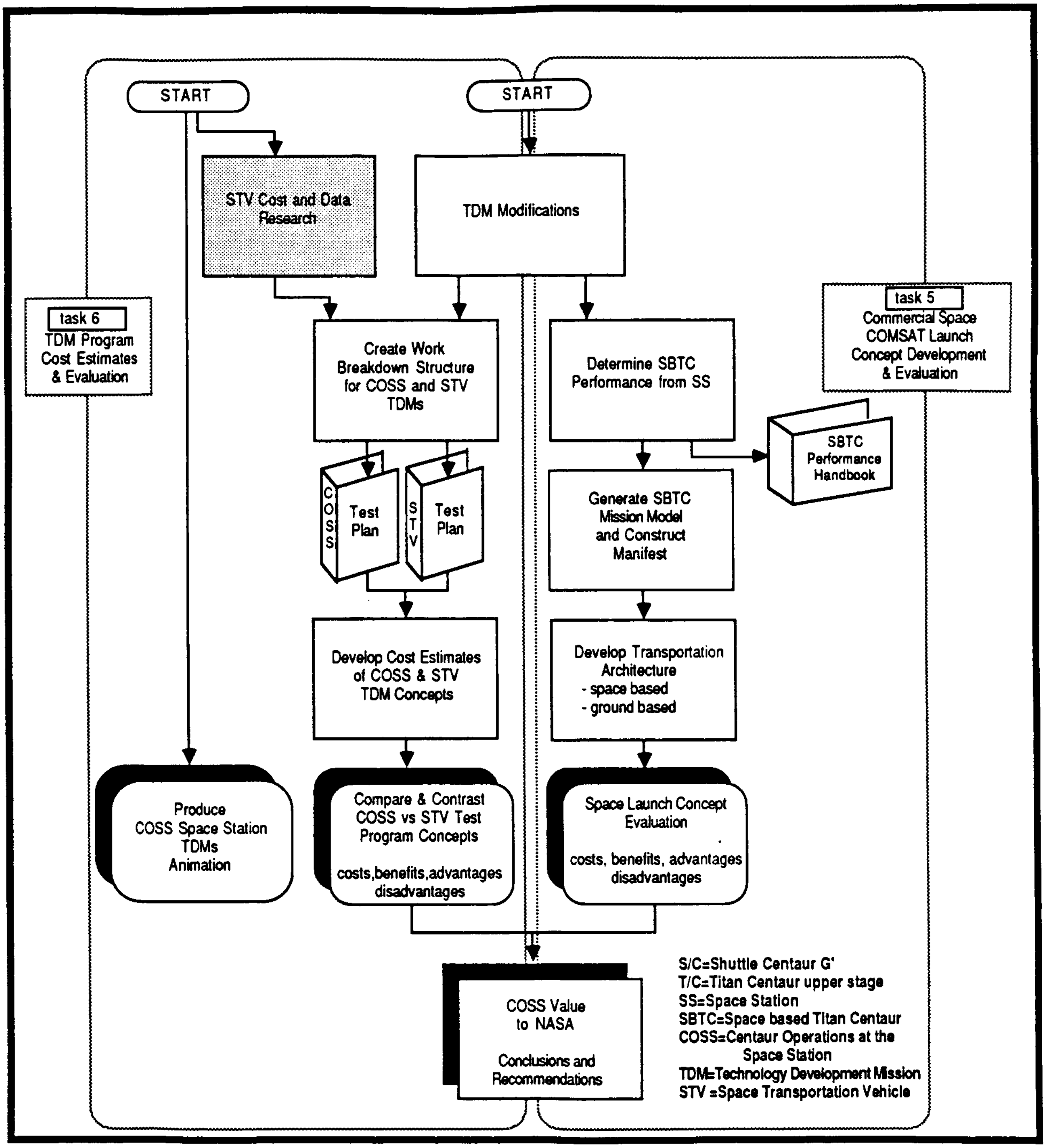
The COSS study generated a TDM program concept that uses an SBTC. TDMs are experiments and exercises to provide experience/develop STV accommodations and operations at the Space Station before STV operational deployment.

Here, the gross and net cost to NASA of the COSS program is estimated. This is done by comparing the costs and the functions of COSS TDMs to similar TDMs, not using SBTC, which are currently part of STV development planning.

COST DISCLAIMER

The cost estimates herein are for planning and comparison purposes only and do not constitute a commitment on the part of General Dynamics.

4.1 STV COST AND DATA RESEARCH



4.1 STV COST AND DATA RESEARCH

The cost analysis task described here is based on the Phase II requirements of NASA contract NAS3-24900, Centaur Operations at the Space Station. The objectives of the cost analysis task were to:

a. Develop the Work Breakdown Structure (WBS) for the COSS II program concept.

b. Develop ROM program cost estimates for the COSS II TDM program.

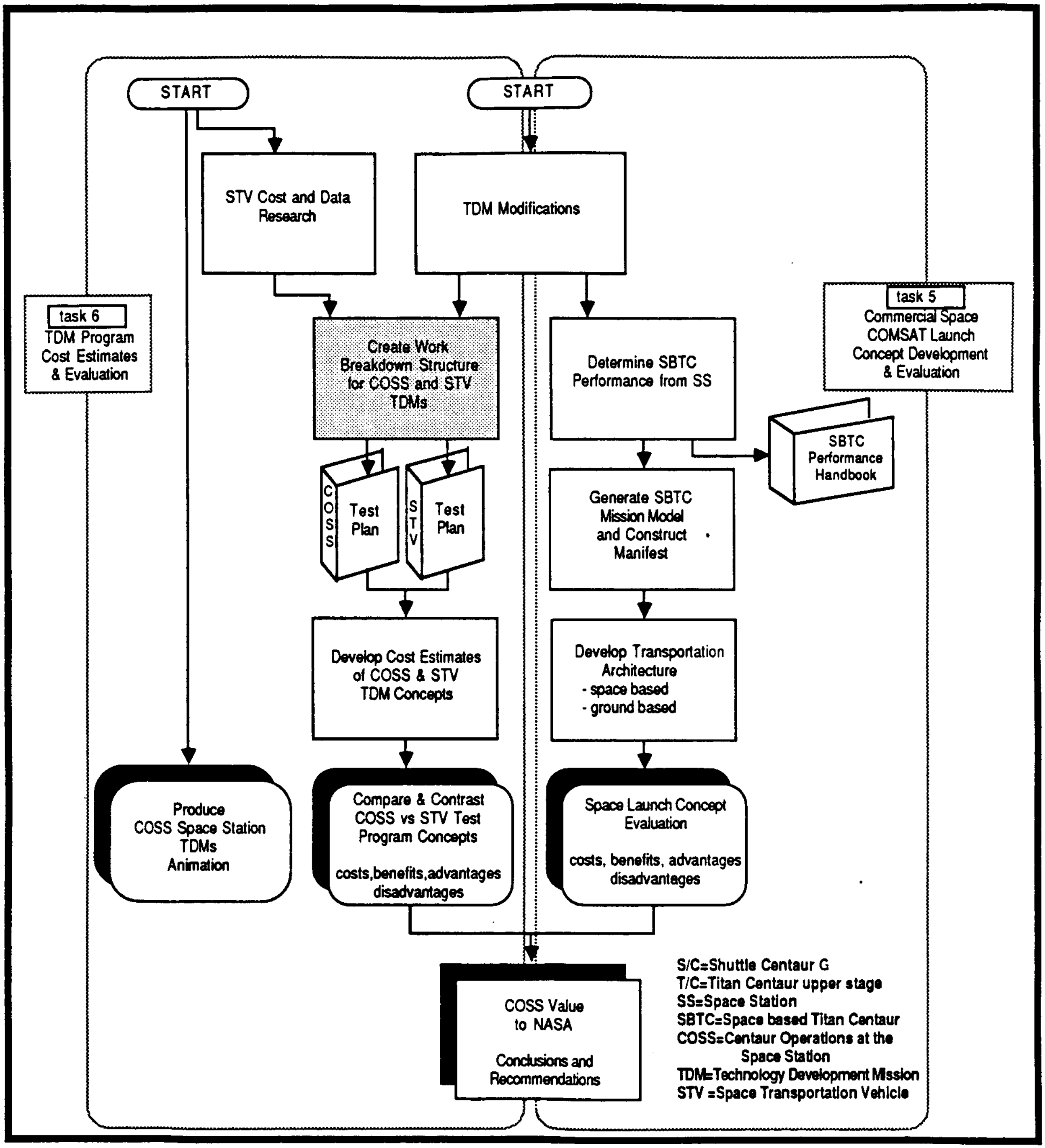
c. Assess the value of the COSS TDM concept to NASA.

To prepare for this task, cost and data research on COSS TDMs and STV development plans for accommodations and operations (A&O) at the Space Station was accomplished. Analysis then began by constructing a WBS for COSS II and STV A&O development. The corresponding WBSs were the framework for defining Test Plans organizing the TDM procedure for COSS and for tantamount test functions in STV development planning. The WBS was also the framework for the principal cost estimating tool for COSS, the parametric cost model. It generated costs at the subsystem level from the engineering technical and software requirements of the TDMs. The principal tool for cost estimates of STV A&O development was the 1987 "Turnaround Operations Analysis for OTV" study (NAS8-36924 DR-3). Cost estimates for TDMs or test functions similar to those in principal to COSS II were extracted and adjusted to include fee.

The following are the ground rules and assumptions used in this analysis:

- All costs are ROM for preliminary planning purposes only.
- Costs are in constant FY 1987 $M.
- No government support or STS costs are included.
- All estimates include 10% fee and excluded management reserve.
- IVA costs and EVA costs are based on GD STV study.
- The Propellant Transfer Storage and Reliquefaction technologies are available.
- No cost of space-based maintainability of SBTC and CISS is included.
- Flight test and GSE are excluded.
- SBOMV costs are based on NASA ground rules for Space Transportation Study (STAS).
- ELV vehicle costs are included with appropriate launch rate and technologies.

4.2 WORK BREAKDOWN STRUCTURE (WBS) FOR COSS AND STV TDMS



4.2 WORK BREAKDOWN STRUCTURE (WBS) FOR COSS AND STV TDMs

To assess the value of the COSS TDM concept to NASA (objective 3), it was necessary to compare the cost for the COSS test program concept to similar development currently planned for STV. This required the creation of a WBS, WBS dictionary, and cost estimates for both programs. The WBS for the COSS TDMs is a hierarchical organization of the proposed programs elements that must be considered in performing programmatic and cost analyses. The COSS WBS was integrated into the Space Station WBS, since it affects Space Station operations at Level 4. Our data research indicated this was the level reserved for A&O development. From Level 4, COSS WBS element definition extends down to Level 8.

The COSS WBS contains seven major systems: Accommodations TDM, Operations TDM, SBTC Vehicle, Space Station Modifications, COP, and COSS II Delivery Transportation. An abbreviated WBS illustrating the relationships of these elements is shown in Figure 4-1. The complete COSS WBS, and a dictionary describing major program elements identified by the WBS, are given in Appendix F.

Our research found that no STV TDM WBS existed. From an examination of Turnaround Operations Analysis for STV contract NAS8-36924, December 1987, we created an STV TDM program WBS broken into five major systems: Simulated STV, Accommodations TDM, Shuttle Airborne Support Equipment (ASE), Cryogenic Transfer, and Space Station Modifications (Scars). The relation of these elements is shown in Figure 4-2. Because STV A&O development cost estimates were made by extending existing data, and did not depend on the WBS, no further detailing was done. However, the functions implied in the WBS of Figure 4-2 were compared to those in the COSS WBS in Figure 4-1 to aid in value analysis. No STV TDM WBS dictionary is available.

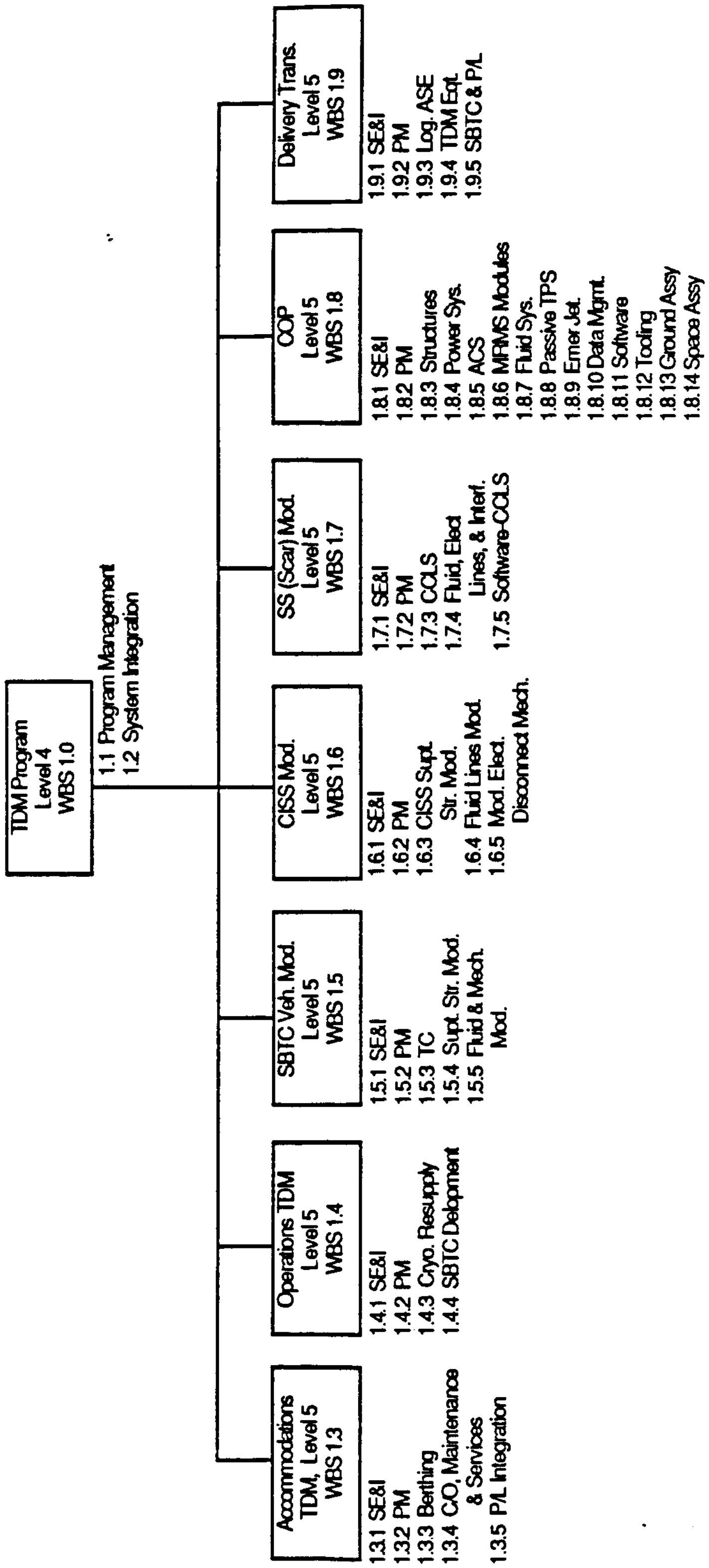


Figure 4-1. COSS Has Seven Major Work Components Which Are Related by Our WBS

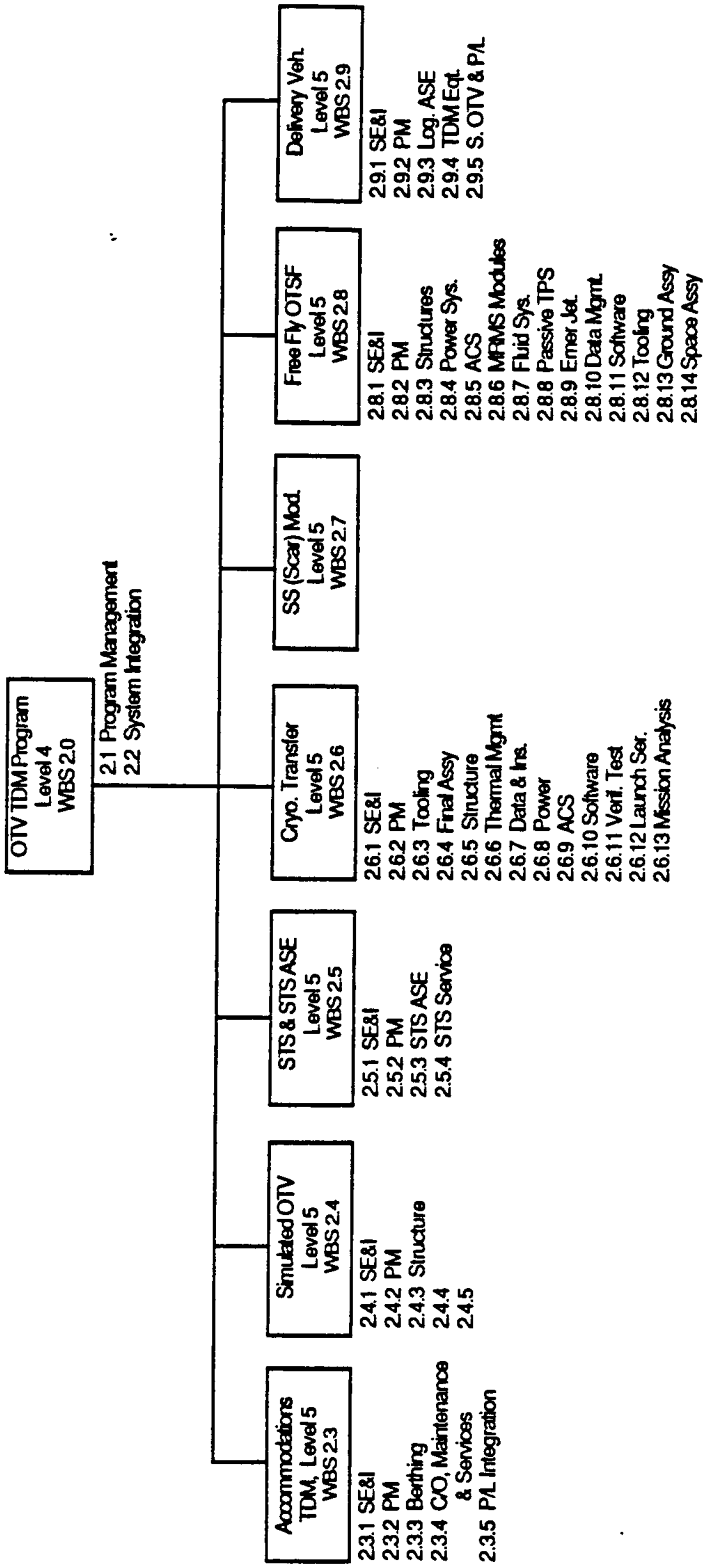
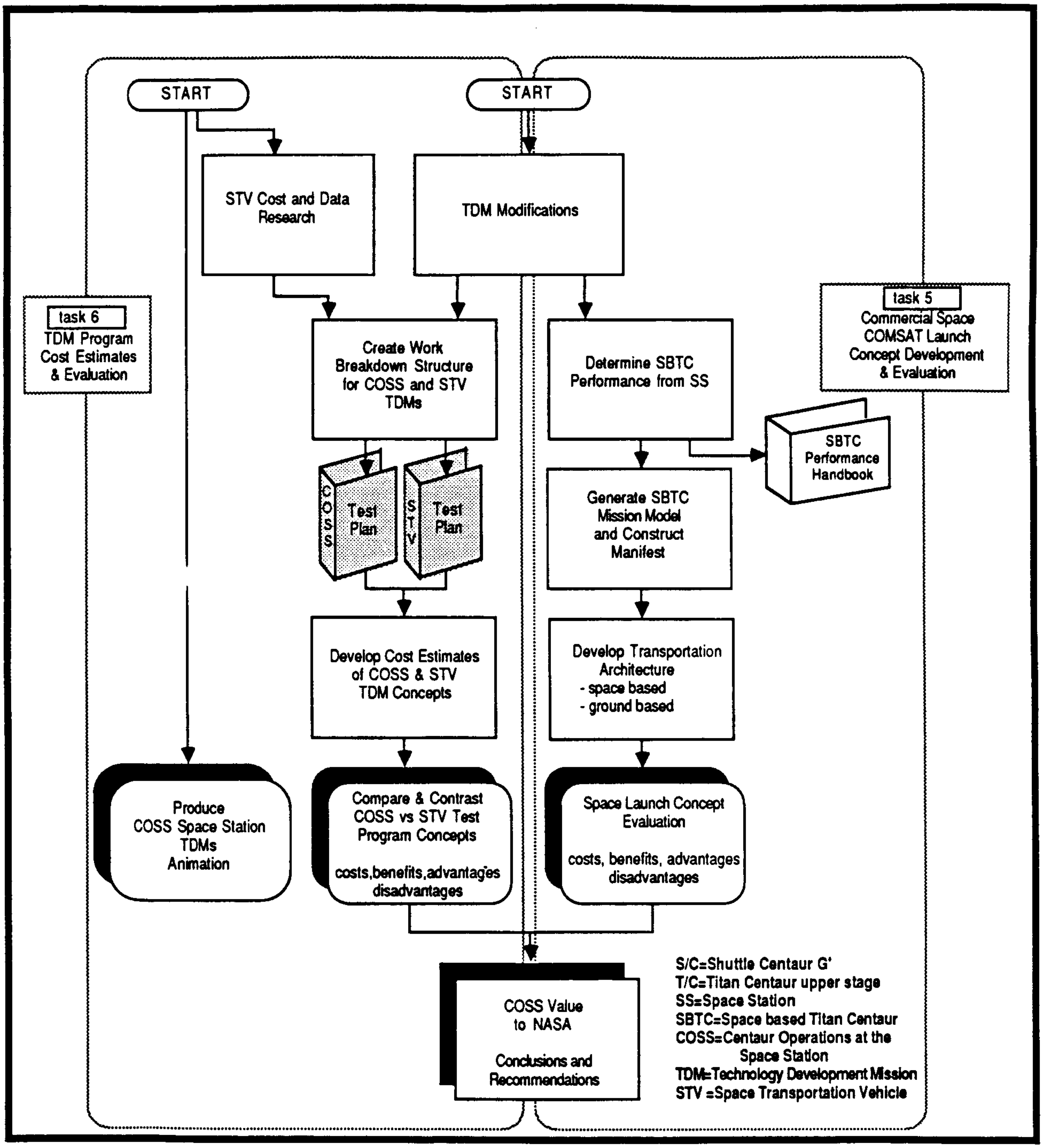


Figure 4-2. A WBS Was Created For General Dynamics' Currently Proposed STV TDM Program

4.3 GENERATE TEST PLANS FOR COSS AND STV TDMs



4.3 GENERATE TEST PLANS FOR COSS AND STV TDMs

The STV development program will conduct training and technology verification/development missions at the Space Station called TDMs. They will be low-risk demonstrations of the required technologies for STV turnaround operations at the Space Station. Data and experience gained from these TDMs will be useful in verifying the Space Station A&O concepts and procedures for the STV.

Two Test Plan outlies were developed to highlight the similarities and differences between the currently planned STV TDM program, and a proposed COSS TDM program. The test plans facilitated comparison and evaluation of the two test programs cost and value to NASA in Section 4.3. The proposed STV TDM program would use a dummy vehicle, or parts of vehicles in tests to simulate an STV. The COSS program uses an operational Centaur G-Prime vehicle, which is very near to early STV dimensions and capabilities, for realism in testing.

The COSS test program conducts five experiments within two TDMs:

- Accommodations TDM
  - Berthing
  - Checkout, Maintenance, and Servicing
  - Payload Integration
- Operations TDM
  - Cryogenic Propellant Resupply
  - Centaur Launch Operations and Deployment

Four TDMs are conducted by the STV test program:

- Docking and Berthing
- Maintenance and Servicing
- Payload Mating
- Cryogenic Propellant Transfer and Storage

4.3.1 COSS TEST PLAN OUTLINE. COSS TDMs begin with the delivery of a hangar to the Space Station for the berthing experiment. After hangar assembly, a specially modified SBTC attached to a (CISS) assembly arrives at the Space Station. Together they are known as the CCA. The Orbiter RMS arm grapples the CCA and hands it to the Space Station MRMS arm. The MRMS translates from the shuttle dock to the SBRC hangar, where it penetrates the CCA into the hangar for a second hand-off to the two hangar TRAs. The TRAs perform the final installation into hangar as shown in Figure 4-3.

The checkout, maintenance, and servicing experiment is conducted in the Space Station hangar. To simulate STV, some Centaur avionics boxes and small hardware will be built for in-space removal. These Orbital Replaceable Units (ORUs) will be removed and replaced to demonstrate maintenance and servicing operations in the low-gravity environment at the Space Station. Actions will be performed as IVA by the hangar TRAs controlled from a Space Station control room. Station mission specialist will be available to perform EVA assistance, if required.

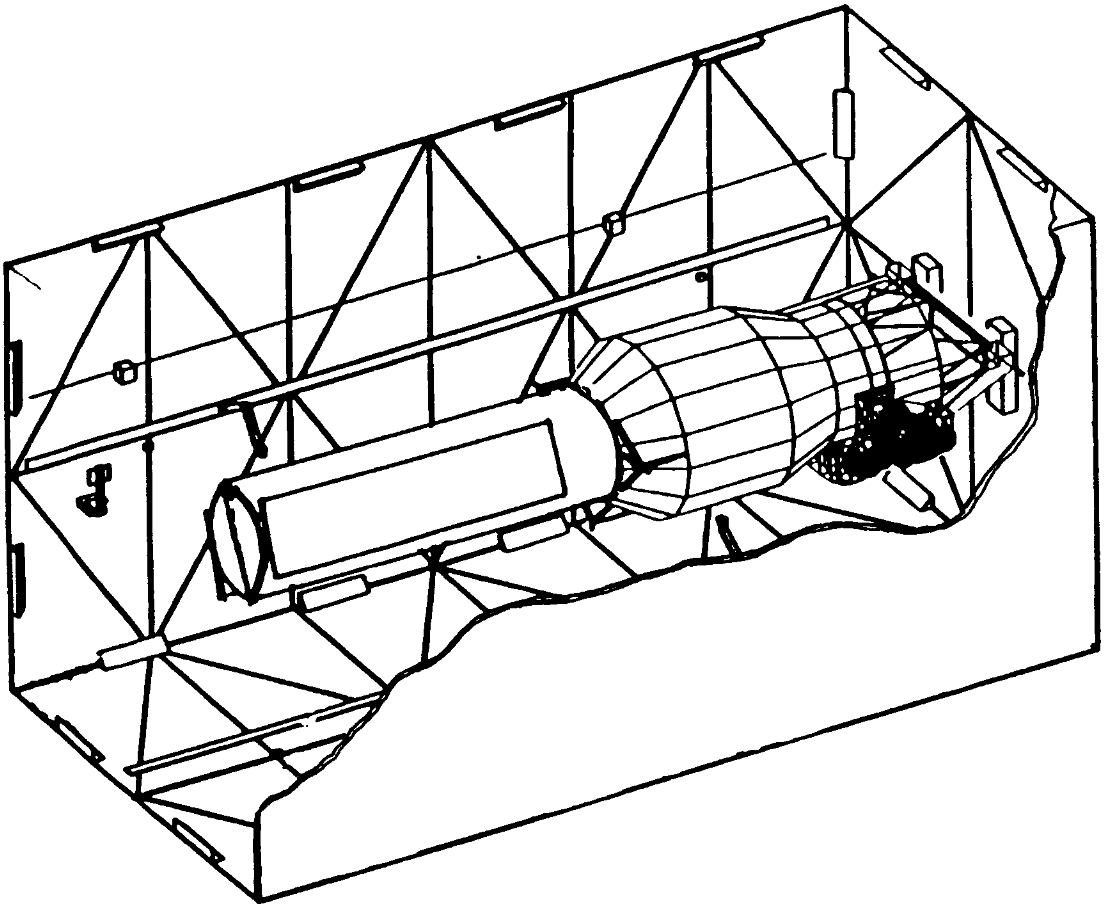


Figure 4-3. The COSS Accommodations TDMs Are Conducted in a Space Station Hangar

The COSS payload integration experiment will mate single and multiple dummy payload configurations to the STV UPA and MPA. These dummies will have circuitry to emulate payload status signals when interrogated. This feature will help test the success of payload mating operations. Aside from data base value, the experience from the payload integration experiment will be applied in the actual payload mating and checkout in the SBTC launch deployment experiment.

Three cryogenic propellant resupply tanking exercises are conducted to demonstrate the technologies required to transfer cryogens in the low-gravity environment of space. To minimize risk to the Space Station, the COSS cryogenic experiment is performed at a COP.

The Centaur launch operations and deployment experiment is preceded by mating and checkout of an actual payload to the CCA in the Centaur hangar at Space Station.

C-2

The CCA/payload stack is transferred to the COP for cryogenic tanking and final checkout. The deployment sequence is conducted utilizing the same systems, data links, and operations that will be required by STV to conduct launches from a space-based platform. Figure 4-4 typifies the systems required for launch operations and deployment, Again, the experience and data base gained will streamline future operations at the Space Station.

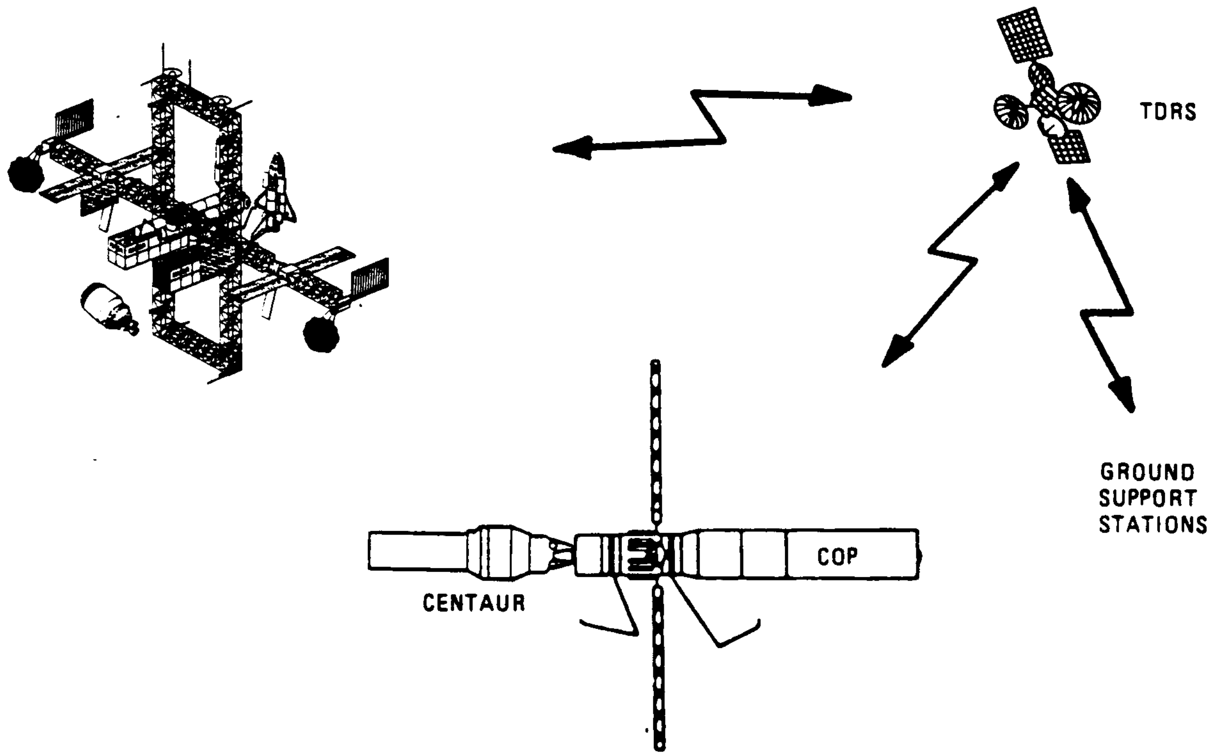


Figure 4-4. The Centaur Deployment Experiment Will Demonstrate the Systems, Data Links, and Operations

4.3.2 STV TEST PLAN OUTLINE. The simulated STV TDM begins with delivery of an STV simulator, berthing carriage, and TDM support equipment to the Space Station. Upon arrival, the hardware is attached to the Space Station truss system.

The Docking and Berthing TDM begins after attaching the OMV to the STV simulator. The mated assembly performs free flight maneuvers under control of the OMV propulsion system. The simulated STV is captured by the Space Station MRMS and secured to the berthing carriage to complete the docking and berthing experiment. Figure 4-5 shows simulated STV docking to the station MRMS, and as it would appear on the berthing carriage where the maintenance and servicing experiment will be performed.

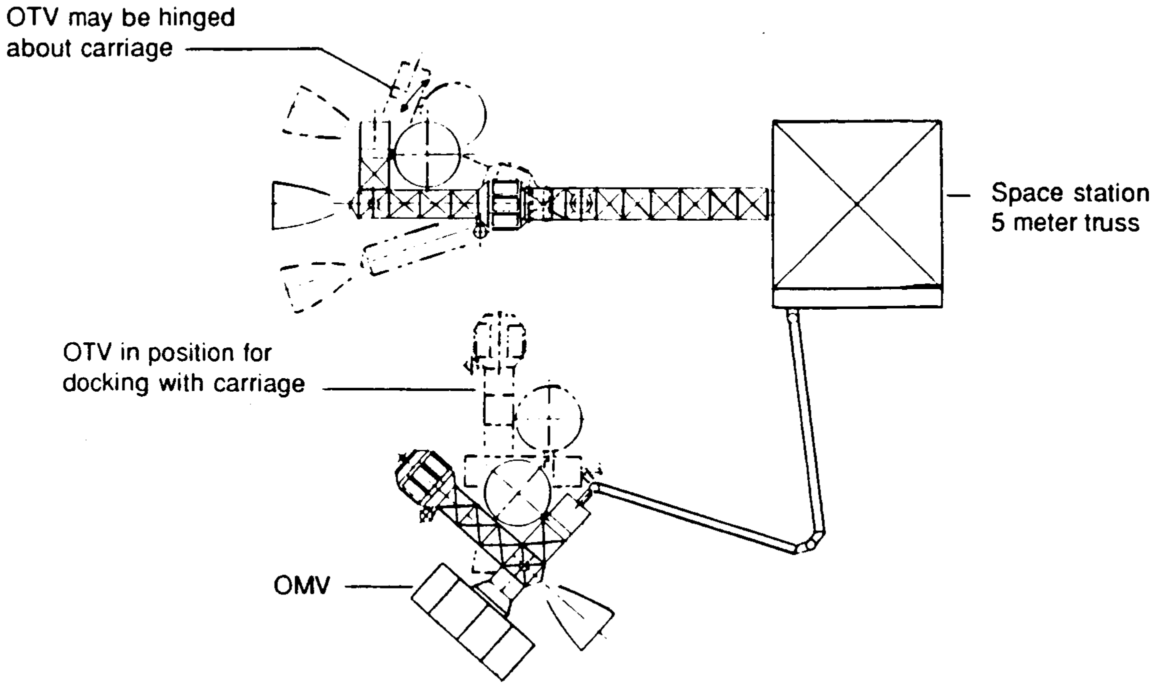


Figure 4-5. The OTV Docking and Berthing TDM Conducted With the Space Station MRMS and a Truss

The maintenance and servicing experiment involves Remove and Replace (R&R) operations on ORUs, both by EVA and IVA. Five ORUs are subjected to remove and replace operations to complete this experiment.

The payload mating experiment, shown in Figure 4-6, is conducted with the simulated STV residing in a 90 degree position on the berthing carriage. The payload mating operations will utilize both EVA and IVA experiments using a dummy payloads similar to those envisioned for the COSS payload mating TDM experiment.

Cryogenic propellant transfer and storage experiment conducts a full-scale LH_2 receiver and supply tank on the Space Station. No LO_2 is involved. A full LH_2 supply tank, and an empty receiver tank are delivered by an expendable booster and OMV (Figure 4-7) to the station and installed. LH_2 will be transferred between tanks to demonstrate the technologies required by OTV turnaround operations in the low-gravity environment at the Space Station.

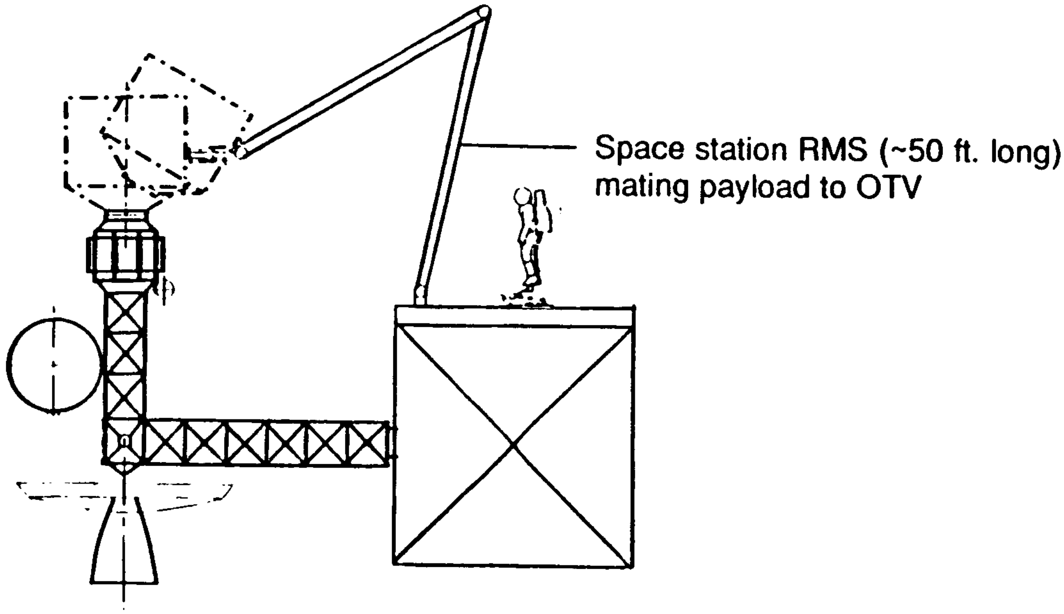


Figure 4-6. The OTV Payload Mating TDM Will Utilize Both EVA and IVA

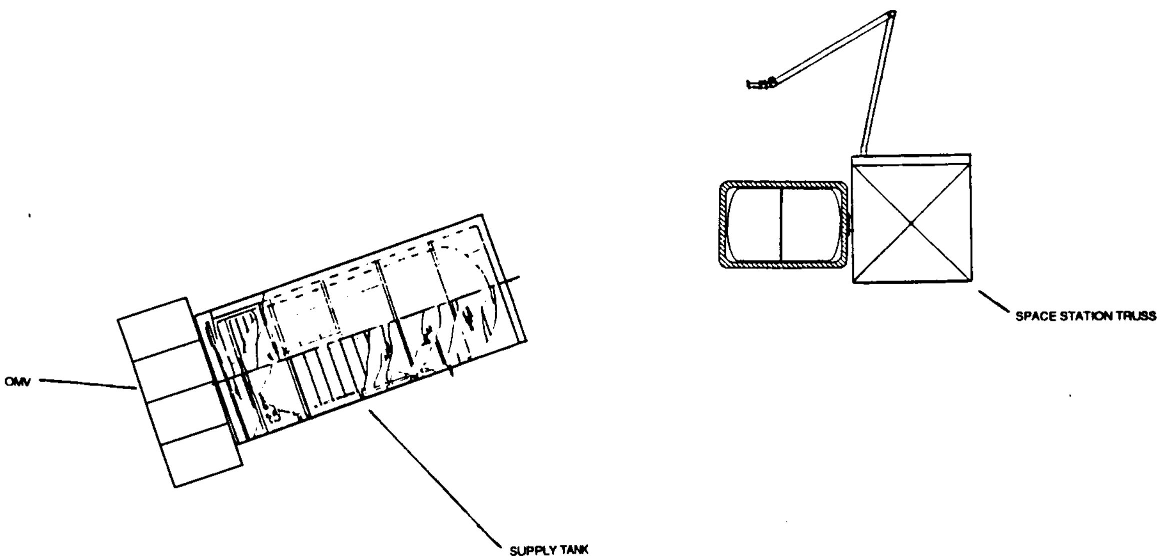
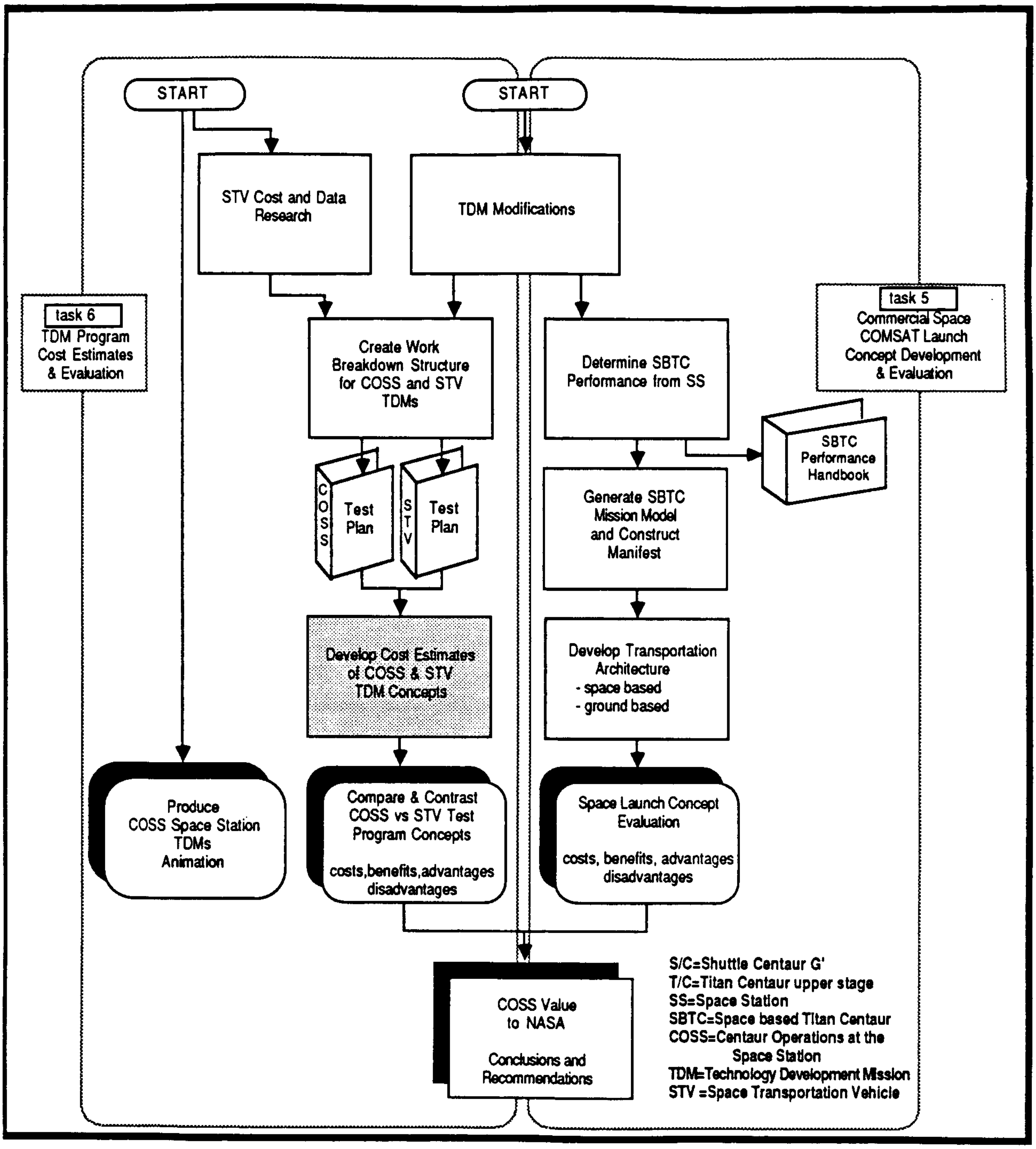


Figure 4-7. The OMV Ferries a Full LH_2 Supply Tank to the Space Station for an OTV Propellant Transfer TDM

4.4 DEVELOP COST ESTIMATES OF COSS AND STV TDM CONCEPTS



4.4 DEVELOP COST ESTIMATES OF COSS AND STV TDM CONCEPTS

The overall estimating procedure is illustrated in Figure 4-8. The principal tool for generating COSS estimates was a parametric cost model. It generates costs at the subsystem level from engineering technical and software input sheets, and program definition data. The cost model and input sheets are exhibited in Appendix E. The model was computerized using Macintosh Microsoft Excel software and contains a series of cost estimating relationships (CERs) and factors designed to represent each hardware element. The CERs are derived based on an analysis of historical cost data and on an analysis of cost-driving parameters for the range of technical approaches and performance parameters encountered in the program. The model generates costs by program phase, specifically: DDT&E and manufactured flight hardware. The DDT&E phase is subdivided into: Design & Development, Ground Test, and Initial Spares.

One of the earliest COSS tasks was to switch test vehicle baselines from a STS/Centaur to a Titan/Centaur (T/C), altered for space station basing (SBTC). Table 4-1 estimates the total cost of a one-of-a-kind SBTC as $71.8M. It also shows that production units, applicable to an ongoing commercial space transportation operation, would cost only $41.5M. No cost allowance was included for space-based maintainability beyond the planned 9-month TDM lifetime. Grossly speaking, modifications were relatively minor. It will be noticed that the most expensive item is Vent & Feed System modifications. This is basically rerouting and requalifying T/C plumbing to work with an existing CISS. The CISS was ASE, built to cradle and monitor the STS/Centaur while riding in Shuttle. These functions are provided by ground umbilicals for normal T/C, but the CISS will be needed for SBTC. None of the modifications are new technology except for the incorporation of zero-gravity mass gauging and propellant acquisition devices. It is assumed these items will be preproven in the precursor LTCSF development program. Refer back to Section 3.1.1 for further details of T/C to SBTC conversion.

A summary of the COSS TDM test program cost estimates, using the Shuttle and Titan IV for logistics transportation, is shown in Table 4-2. Note that the COP accounts for about 47% of the proposed TDM program cost. Without it, the cryogenic propellant resupply experiment, and the COMSAT launch could not be conducted. This is because Space Station policy is tending toward not allowing cryogenic propellants on the station for safety reasons, and the desire to insulate delicate Space Station experiments from launch vibrations and effluents. In the year 1998, the Shuttle Derived Vehicle (SDV) and/or the ALS should be operational. If these vehicles are used for TDM logistics, the proposed program costs could be reduced by $75.8M to $136.6M.

Historical funding data from similar space programs, viz., Apollo, Shuttle, etc., was used to spread the total costs in Table 4-2 into the funding profile of major WBS elements presented in Figure 4-9. As reflected in the profile shape, typically 60% of the DDT&E costs are spent by the midpoint of the DDT&E phase, and about 50% of the flight hardware costs are spent by the halfway point of the flight hardware manufacturing phase. Supporting detail is given in Table 4-3.

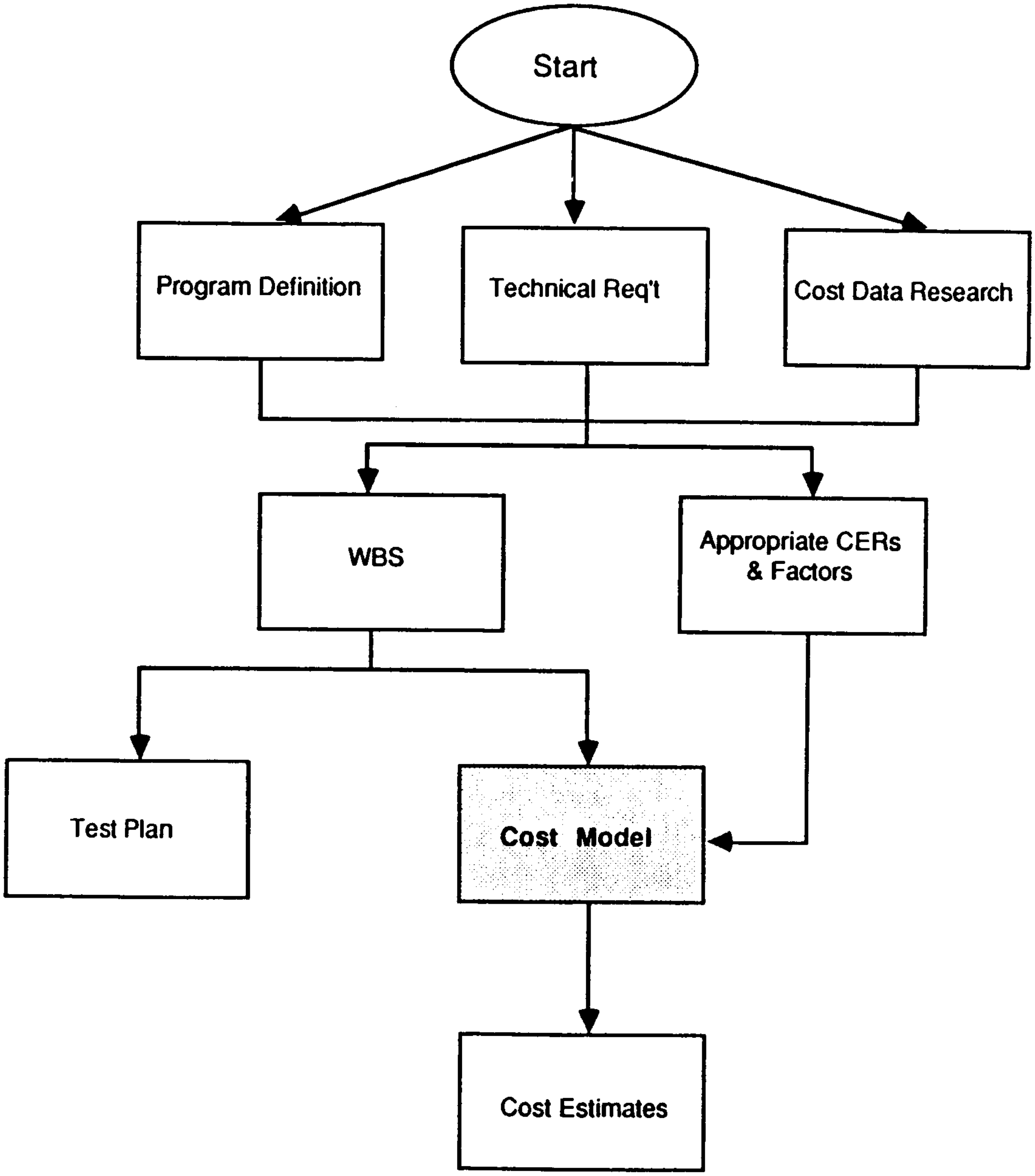


Figure 4-8. The Principal Tool for Generating Cost Information for COSS TDMs Was Our Parametric Cost Model

Table 4-1. Rearranging Vent and Feed System to Fit STS/CENT CISS is Costliest Item in Converting T/C to SBTC

DESCRIPTION OF T/C MODIFICATION	Δ WGT. (#)	DDT&E 87$M	RECUR 87$M	TOTALS
Structural				
Add S/C G' Forward Trunion/Keel support	711	3.0	0.1	3.1
Replace Aft Adapter w S/C 13.95 " Dia	(58)	1.1	0.7	1.8
Vent & Feed Sys. (Mod to fit CISS)	836	14.7	4.7	19.4
Data Management				
Liquid Acquisition Device (Similar to LTCSF)	205	1.1	0.6	1.7
0-g Mass Guaging Device (Similar to LTCSF)	5	1.7	0.2	1.9
Diffuser/Dissipator (Similar to LTCSF)	10	0.8	0.1	0.9
Program Management		3.5	1.0	4.5
SE & I		4.4	1.0	5.4
Titan/Centaur Vehicle (Dry Wt.)	6720		33.1	33.1
TOTALS	8429	30.3	41.5	71.8

Note : No cost allowance for space based maintainability

Table 4-2. COP Accounts for 47% of Proposed COSS TDM Program Costs, But Without It, Cryogenic Propellant Transfer and COMSAT Launch Would Not Be Included

WBS NO	MAJOR SYSTEMS	DDT&E PRICE 87M$	Flt. Hardware PRICE 87M$	TOTALS
1.1	Program Management (System Level)	11.1	4.8	15.9
1.2	System Integration (System Level)	19.2	8.3	27.5
1.3	Accommodations TDM	148.0	42.1	190.1
1.4	Operations TDM	11.1	11.1	22.2
1.5	SBTC Vehicle and Modifications	30.3	41.5	71.8
1.6	CISS Modification	19.1	2.4	21.5
1.7	Space Station Modifications (Scars)	34.9	5.7	40.6
1.8	Co-Orbiting Platform	517.9	125.0	642.9
1.9	Delivery Transportation	67.4	259.4	326.8
1.0	TOTAL CSOD PROGRAM	859.0	500.4	1359.3

Note:

1. P/L Simulators assumed 4 subscale mockup for GPS and 1 subscale mockup for TDRSS
2. No cost for STS service due to NASA mission (2 STS flights will be required)
3. Costs for Titan service assumed 3 flights
4. Manifest on STS - SBTC (8429 lb), P/L (18700 lb), P/L Simulators (3000 lb), CCLS & Scar (310 lb), UPA & MPA (1175 lb), Hangar (5968 lb), C/O Tool & ORU (700 lb), COP Core Module (16957 lb), CISS (7669 lb), & ASE
5. Manifest on Titan - COP LO2 Module & Fuel (2-28602 lb), COP LH Module & Fuel (17109 lb), & ASE (2-3000 lb)

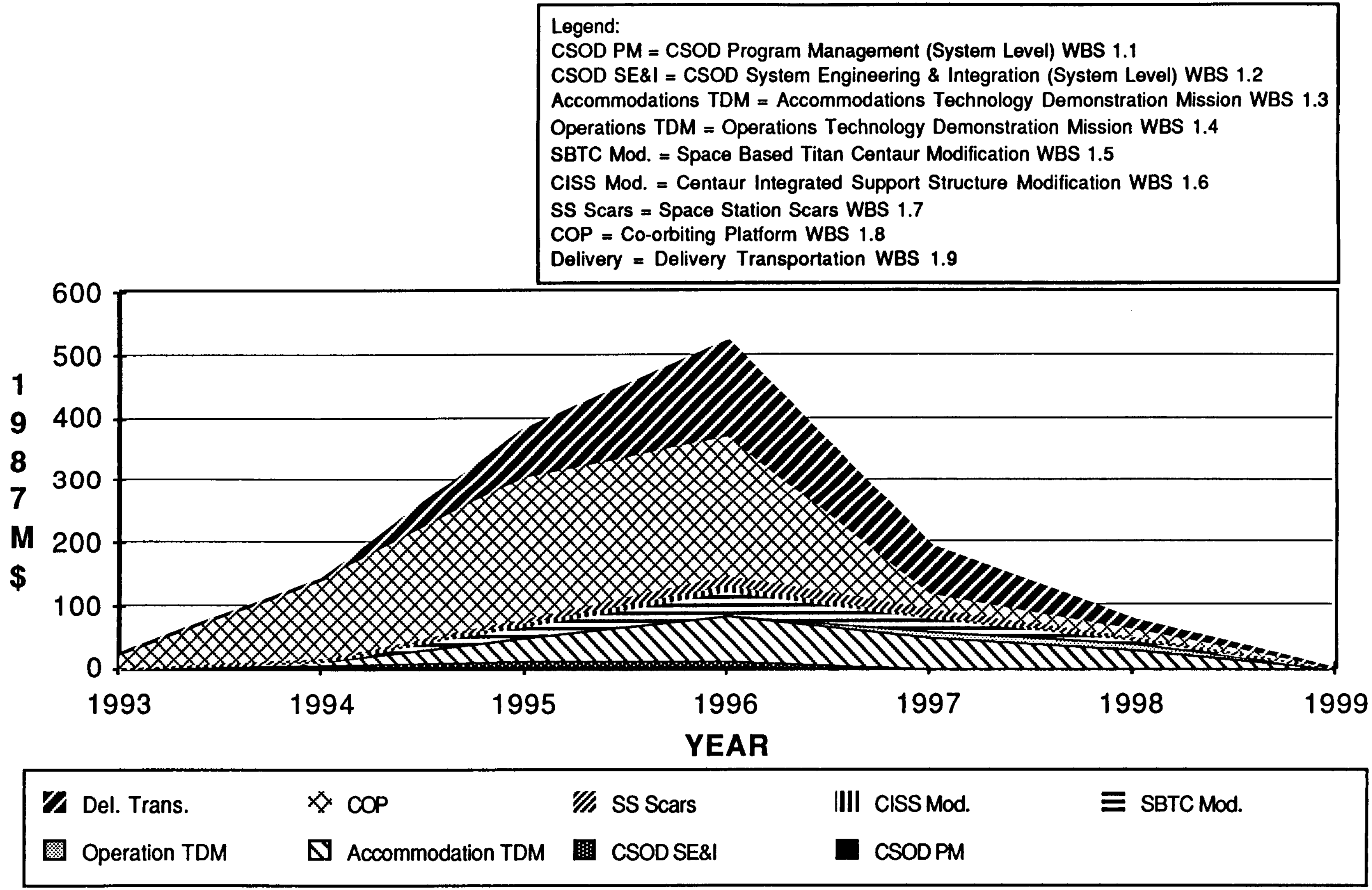


Figure 4-9. A Funding Profile Was Developed From Historical Analysis of Similar Programs

Table 4-3. COSS Program Funding Profile Was Developed at the Major Systems Level

WBS NO	MAJOR SYSTEMS	FUNDING PROFILE 87 M$						
		1993	1994	1995	1996	1997	1998	TOTAL
1.1	Program Management (Sys. Level)	0.6	3.2	5.6	5.6	0.6	0.3	15.9
1.2	System Integration (Sys. Level)	1.1	5.5	9.7	9.6	1.1	0.5	27.5
1.3	Accommodations TDM	0.0	6.0	35.2	71.7	48.6	28.6	190.1
1.4	Operations TDM	0.0	0.0	0.0	0.0	11.1	11.1	22.2
1.5	SBTC Vehicle and Modifications	0.0	0.0	17.2	33.6	17.2	3.8	71.8
1.6	CISS Modification	0.0	0.0	5.2	10.1	5.2	1.0	21.5
1.7	Space Station Modifications (Scars)	0.0	1.3	7.5	15.3	10.4	6.1	40.6
1.8	Co-Orbiting Platform	24.6	129.7	227.4	225.0	24.6	11.6	642.9
1.9	Delivery Transportation	0.0	0.0	78.4	152.9	78.4	17.0	326.8
	TOTAL FUNDING REQUIREMENTS	26.4	145.7	386.2	523.8	197.2	80.0	1359.3

Note : 2 STS service costs are excluded

Complementary to the COSS estimate, ROM cost estimates were completed for major elements of the STV TDM program. These estimates did not use a WBS or milestone chart, but instead were based on cost data from the final review presentation of NASA contract NAS8-36924-D-R-3 ("Turnaround Operations Analysis for OTV," Final Review Meeting at NASA/MSFC, December 9, 1987), as displayed in Appendix H. The Turnaround Operations document developed cost estimates for STV A&O TDMs at the Space Station using a dummy vehicle constructed from trusses, an empty tank, and dummy engine bells. These costs did not include fee, or space transportation and ASE costs for LH_2 used in a planned cryogenic propellant transfer experiment. We extended the STV TDM costs with these items, escalated all costs to 1987 dollars, and constructed the cost comparison chart of the COSS versus STV proposed test programs shown in Table 4-4. The reader should not be alarmed that the initial cost of the proposed COSS TDM program would be about twice as much as the STV A&O development program. This is because COSS provides more hardware and functions than its STV development counterpart. These topics will be covered in the next section (4.1.4). A soft area in the STV test program estimate could be the cryogenic propellant transfer experiment. If future Space Station policy continues its trend toward disallowing cryogenic testing/refueling on the station, the STV test program would have to develop a structure similar to the COP, at extra cost, or wait for full-scale STV prototype experiments in concert with the COLDSAT project (see reference cited above).

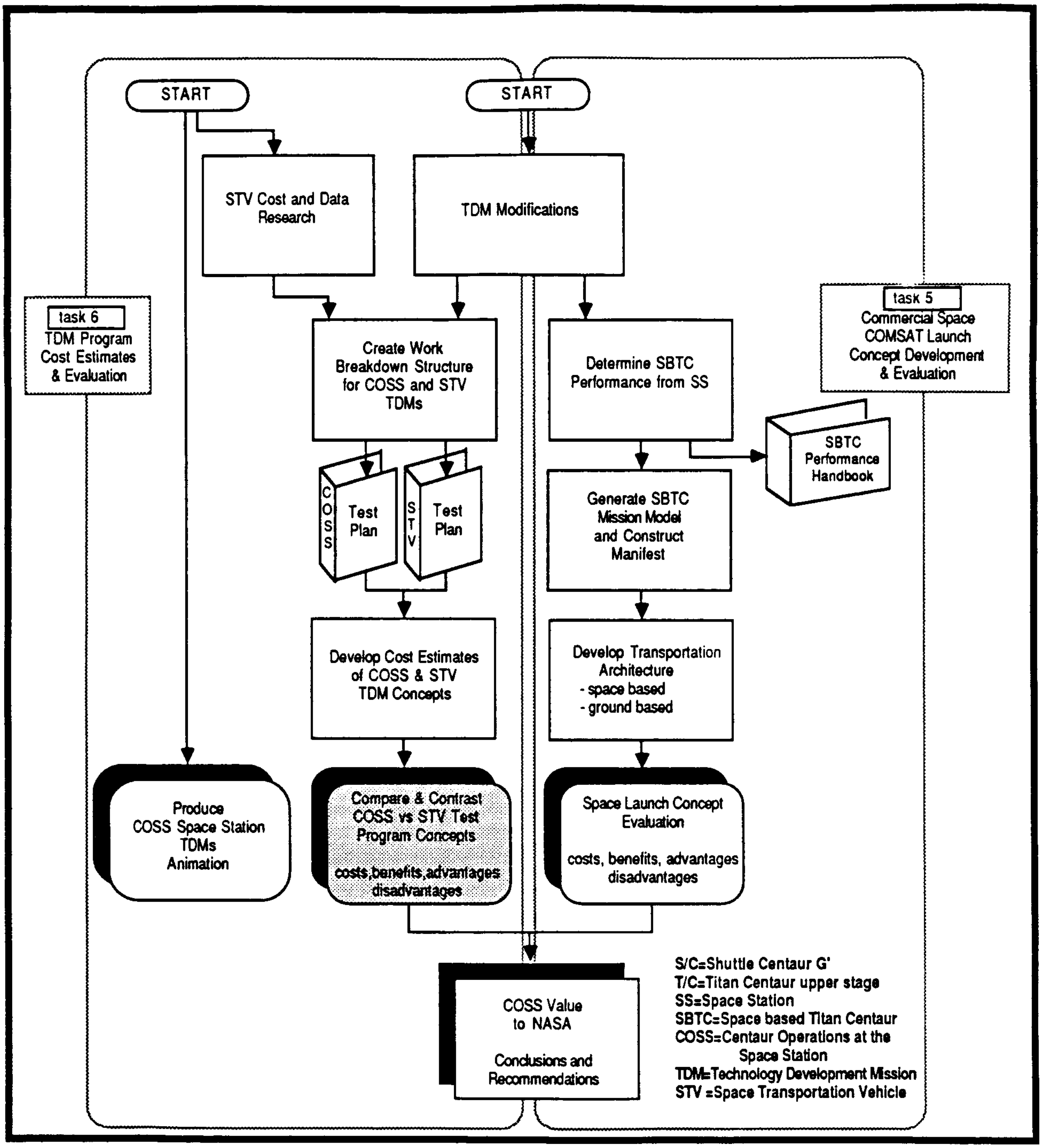
Table 4-4. The COSS Test Program Would Initially Cost More Than the Planned STV Test Program Because It Has More Hardware and Functions

Technology Demonstration Mission Description	COSS Test Program 87M$	Planned STV Test Program 87 M$
Berthing	197.2	48.2
Maintenance & Servicing	8.1	53.4
Payload Mating/Integration	32.4	8
Cryogenic Resupply	691.5	391.6
Launch Deployment	10.0	-
SBTC + CISS	93.3	-
Delivery & ASE	326.8	102.2
Total	1359.3	603.4

Note :

1. No cost assumed for STS service since there are NASA missions
2. No CCLS or CCLS operations in the OTV TDM program
3. Cryogenic resupply TDM may require a platform in the OTV TDM program
4. CSOD includes a payload deployment
5. No test of mutiple payload integration on OTV TDM program
6. No hangar in the OTV TDM program

4.5 COMPARE AND CONTRAST COSS AND STV TEST PROGRAM CONCEPTS



4.5 COMPARE AND CONTRAST COSS AND STV TEST PROGRAM CONCEPTS

Section 4.1.3 pointed out that the initial cost of the proposed COSS TDM program would be about twice as much as the STV A&O development program. The reader must be cautioned that this is raw data and does not consider: 1) test depth and fidelity, or 2) resource reuse credit.

The STV TDM program was not intended to test to the depth or fidelity of the COSS program. It has a dummy vehicle, no Computer Control Launch Set (CCLS), no hangar, no mutiple payload integration test. It includes a cryogenic resupply TDM conducted on the Space Station with LH_2 only, and has no actual launch. It therefore does not incur the expense of a COP. This is not necessarily bad. It merely represents a trade of fidelity for cost. But with less fidelity, more follow-up demonstration missions would probably be found necessary. As Table 4-5 annotates, the advantage of adopting the COSS approach is in the execution of its broader, more realistic TDMs. They should provide problems and solutions more faithful to, and therefore more applicable to, STV experience. This should reduce the development risk and cost of the STV program.

An additional advantage of COSS TDMs is their reuse philosophy. They were designed to become useful components of the Space Station, and of a future maintenance and launch facility for operational STVs. For example, the still evolving Space Station data base document, JSC 30000 (initial release), states in Section 3.2.5, "Payload Checkout, Integration, and Deployment," on page 3-2 (see Appendix G) that "Expendable stages will be stored and serviced. The growth station will also provide the capability for payload deployment to high-energy orbits." Therefore, the SBTC hangar, maintenance and servicing equipment, etc., designed for the COSS TDMs would not have to be duplicated if it were ever decided to store and service a Centaur expendable stage at the Space Station. Table 4-6 details that providing these items in COSS could avoid $330.47M in the Space Station budget. The same section of the Space Station data base document states that: "Reusuable transfer stages will be based, serviced, and maintained and refueled at the station." This provides for an STV servicing facility, generically represented by Figure 4-10 as incorporating COP components after conclusion of COSS TDMs. Candidate reusable COP items and their attendant cost avoidances are listed in Table 4-6. As shown, total avoidance is estimated at $300.19. An alternative way of representing this cost avoidance is to credit the initial cost of the COSS program shown in Table 4-2. With this approach, the net cost of COSS becomes $728.74M. This does not take into account possible revenue from the COMSAT launch, which would further reduce the net cost.

Table 4-5. COSS TDMs Cost More to Implement But Are Higher Fidelity to OTV Than Currently Planned TDMs

EXPERIMENT	COSS ADVANTAGES	COST M$	STV ADVANTAGES	COST M$
Berthing	Hangar remains with Space Station as expendable vehicle service facility after the CSOD program is completed.	197.2	Low risk, simple truss structure, dummy vechle. (Cost includes simulated vehicle, but no hangar)	48.2
Maintenance and Servicing	Demonstration performed on actual flight hardware with CCLS checkout capability. High fidelity TDM has the benifit of the hangar for radiation shielding and micrometeriod protection.	8.1	No risk to vehicle since dummy OTV and inert ORU's.	53.4
Payload Mating/ Integration	Demonstration includes multiple dummy payload excercise with [illegible] and EVA activity for contingency. Actual payload installed for deployment.	32.4	EVA experience gained from the simulated OTV TDM.	8.0
Cryogenic Resupply	Experiment performed at COP using full scale vehicle, transfer lines, and depot tank. COP hardware transferable to OTSF after CSOD is complete. (i.e. solar panels MRMS, CCLS. CCA/OMV docking maneuver performed at COP and Space Station during this experiment. Operational CCLS demonstrates the ability to control tanking and de-tanking manuevers.	691.5	No COP required since the experiment is performed at the Space Station. (Current policy tends toward "No cryogenic propellant experiments being performed at the Space Station")	391.6
Launch Deployment	All experience gained from previous TDM's will be demonstrated. Total launch system, including ground network will be utilized to gain experience for OTV. Cost of TDM partially recovered by deployment of actual payload.	10.0	Not applicable for simulated OTV. (No real vehicle)	
SBTC AND CISS		93.3		
DELIVERY AND ASE	Shuttle and expendable vehicle logistics alternatives for flexibility.	326.8		102.2
TOTAL TDM COST		1359.3		603.4

Table 4-6. Net $728.74M Cost for Proposed COSS TDM Program If Assets Are Reused

Note : *Delivery transportation based on Titan IV 2276 $/lb
** T&E= Tools and equipment
*** All costs include DT&E and flight hardware

REUSABLE COP COMPONENT	$ AVOIDANCE
Power System	64.27
Attitude Control System	10.41
MRM Modules (2)	110.00
Fluid System	48.68
Data Management	69.19
Software (Launch & System)	17.85
Delivery Transportation (10068 lb)	22.91
Modification to fit OTV OTSF (Assumed 10%)	(34.33)
OMV Service (2)	(7.80)
IVA Space Disassembly	(0.99)
Benefit To STV FACILITY	*** 300.19

ALL COSTS ARE 87$,

REUSABLE SS TDM COMPONENT	$ AVOIDANCE
Berthing Hangar	155.26
Checkout, Maintenance, and Service	5.35
Payload Integration	23.02
Space Station ScarS (INCL. CCLS)	41.77
ECP Ground Launch (ALS-E)	88.10
Delivery Transportation (7453 lb)	16.97
Benefit To Space Station & ECP Launch	*** 330.47

ALL COSTS ARE 87$,

COSS NET COST: $1359.4- $300.19 - 330.47 = $728.74

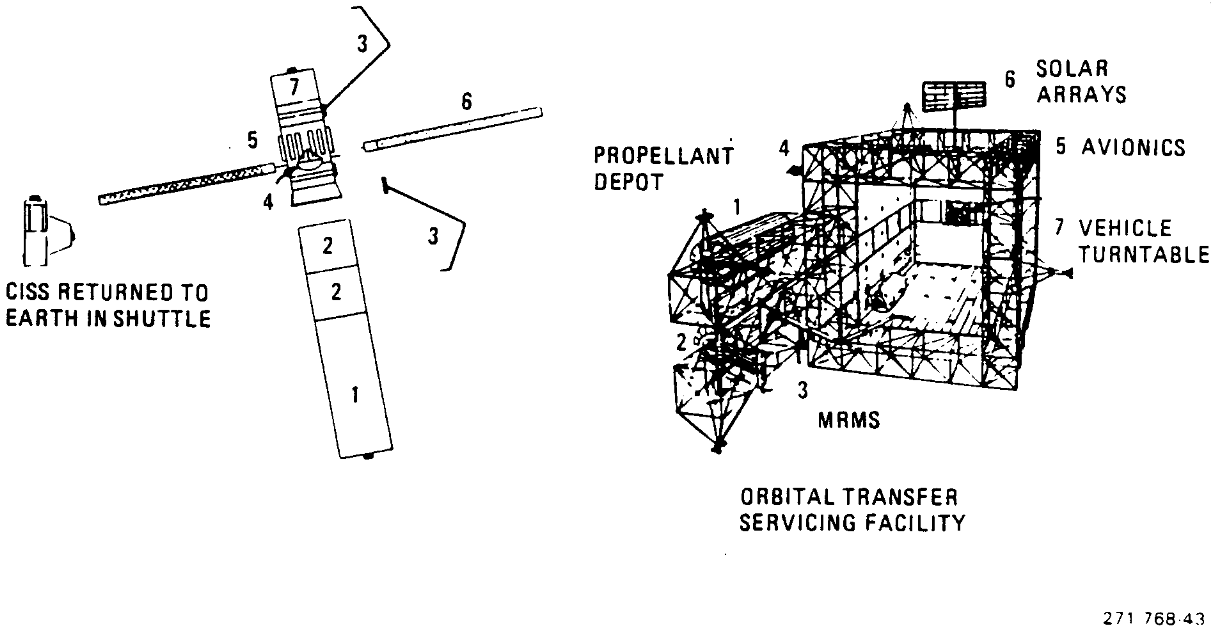


Figure 4-10. COP Components Will Be Incorporated Into the STV Maintenance and Servicing Facility

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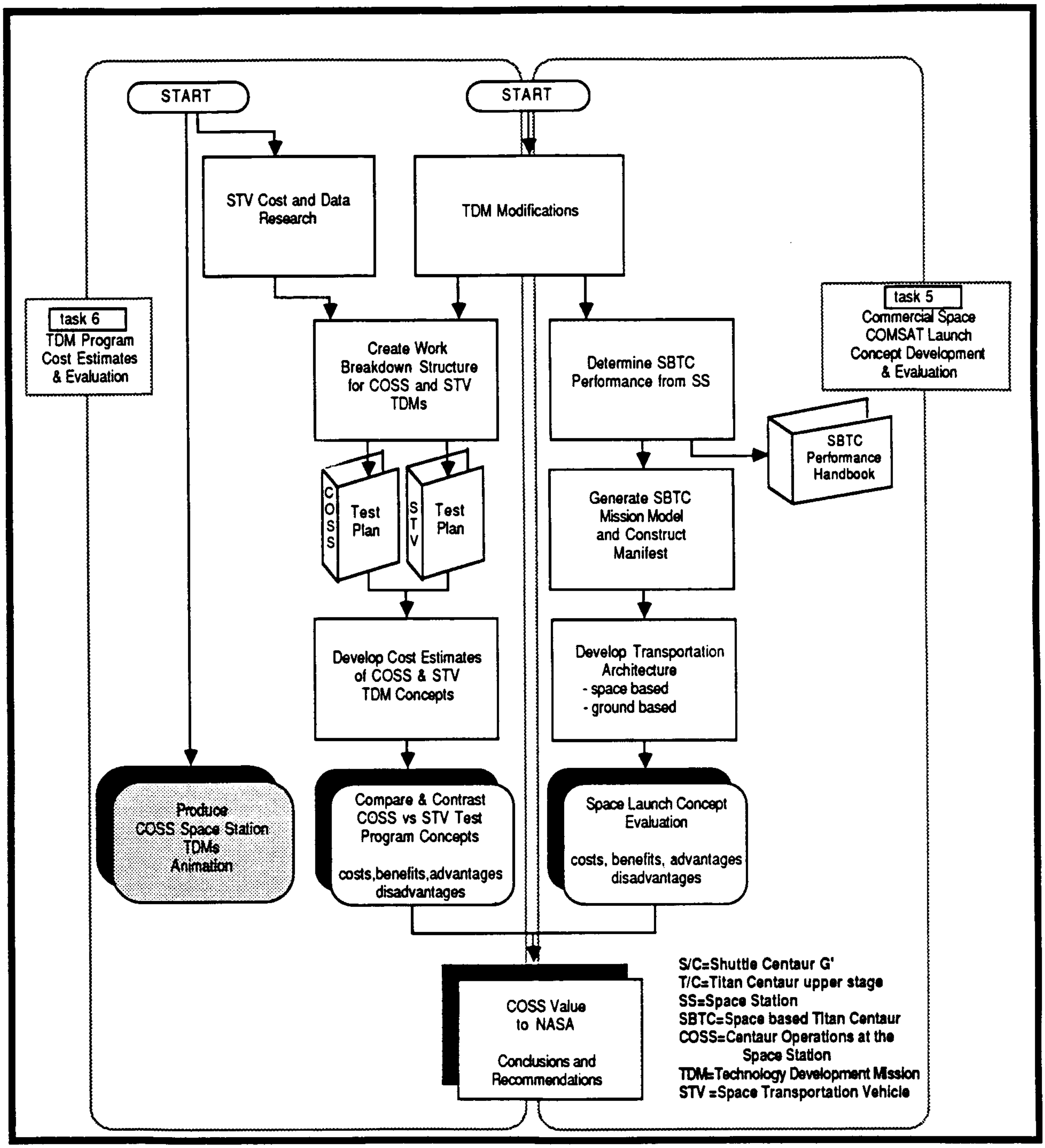
4.6 COSS SPACE STATION TDMS ANIMATION: PRE/POST OPERATIONS



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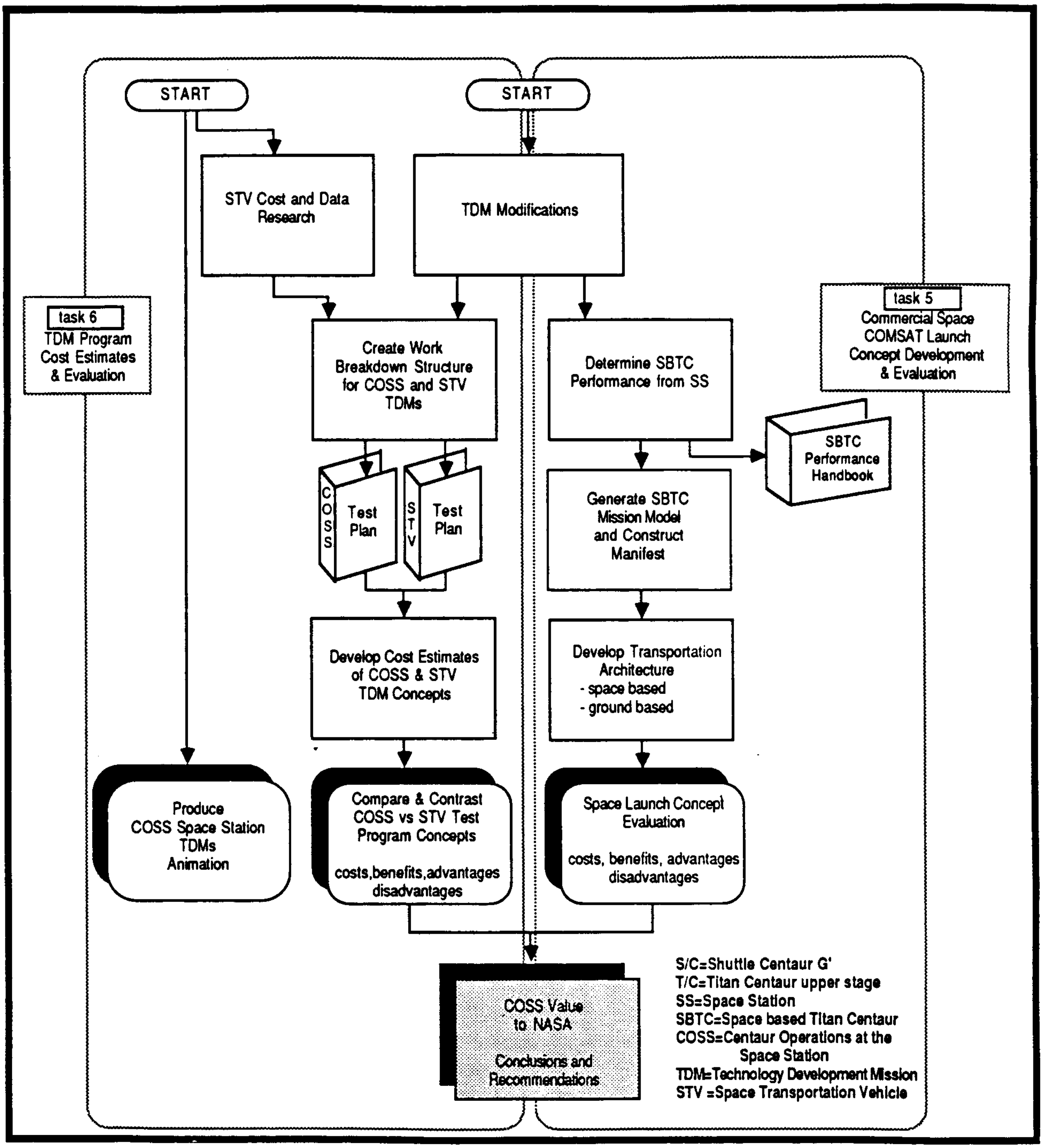
4.6 COSS SPACE STATION TDM ANIMATION: PRE/POST OPERATIONS. A three-dimensional animation was created on an Interactive Machines Incorporated 500 computer by GDSS's Space Simulation Laboratory. Its purpose was to illustrate the operations required to perform the COSS program; from SBTC delivery in the Shuttle, to COMSAT launch from the COP. The preliminary COSS Operations Animation was completed and released by late November 1987 on VHS tape format and on 3/4-in. video tape. As an extra benefit, the animation confirmed that a shorter Centaur Hangar eases MRMS hand-off manipulations in placing equipment into the hangar. For details of sequence planning, see Appendix C.

SECTION 5
VALUE OF PROPOSED COSS PROGRAM TO NASA



This report has detailed the two concepts COSS would present:

- TDMs to demonstrate/develop STV accommodations and operations at the Space Station using a Titan/Centaur modified for space basing (SBTC) as a test vehicle.
- A commercial COMSAT launch program using expendable SBTC transportation vehicles.

This section distills major conclusions, value to NASA of COSS, and makes next step recommendations.

5.1 CONCLUSIONS

A COSS launch program would be valuable to NASA. As developed in Section 3, a purely space-based transportation program does not seem to be economically feasible using presently applied technology. That is to say, it would seem to cost as much, or more, to supply payloads, propellant, and space launch vehicles to the Space Station, as to use the same logistics vehicles as ground launched transportation. The main driving variable here is propellant. If future propulsion systems are developed that are less dependent on Earth-supplied propellant, space launch costs could be drastically reduced. However, a SBTC-based COMSAT launch program used in the augmented (topping off) mode can be economically feasible! For a spectrum of multiple payload weights, it can be cheaper than ground launches. Additionally, this space-based staging mode reduces mission risk since there are no weather windows, and unlike ground launches, only one stage is required for deployment.

Further enhancement of program effectiveness is possible if SBTC propellant tanks can be made to accept modular extensions. The program could have a symbiotic benefit if depleted COP tanks were used to dispose of Space Station refuse, and/or as auxiliary SBTC tanks.

A COSS TDM program would be valuable to NASA. Section 4 implies that for nearly the same cost as current STV development planning, COSS TDMs could provide much more realistic demonstrations and the first space-based COMSAT launch. Lessons learned should reduce STV development risk. The fidelity of COSS TDMs versus STV TDMs should offer expanded opportunities to find, duplicate, and pre-solve STV accommodations and operations hardware and operations problems. The early first space launch may also provide public relations benefits by capturing public attention. By providing early expendable launch vehicle accommodations to the Space Station, COSS may reduce or defray the apparent Space Station budget requirements.

5.2 RECOMMENDATIONS

The recommendations of this report are that NASA initiate:

- An Expendable STV Operations Study
- An Early Space Launch Feasibility Study
- A Space Station Accommodations Technology Demonstration Feasibility Study

The implementation of a COMSAT launch program using SBTC would essentially be an early or expendable STV operation. An Expendable STV Operations Study analyzing the incremental benefits of additional concept modification steps, e.g., larger tanks, larger MPA, low thrust capability, aerobraking, etc., would optimize the value of the concept to NASA.

Using an optimized operations concept baseline, an Early Space Station Feasibility Study would be the first step in creating a legitimate new program.

We believe this study has shown, to a first approximation, that there would be significant benefits to NASA for initiating the COSS TDM program. We believe the concept is ready for a definitive feasibility study.

SECTION 6
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APPENDIX A
SBTC PERFORMANCE ANALYSIS COMPUTER PROGRAM FLOWCHARTS AND SOURCE CODE LISTINGS

A.1 ORBXPL (ORBIT TRANSFER PAYLOAD PROGRAM)

This program determines the payload capability of a restartable liquid-propellant stage to perform a series of transfers between given orbits. The transfers are of the Hohmann variety, with the plane change distribution selected for minimum total transfer velocity increment. Start and stop losses for main impulse and auxiliary propellants are input along with the start and stop impulse values. Provision is made to offload propellants to maintain a given gross weight limit, and auxiliary payloads can be jettisoned at the end of any transfer burn.

A.2 DRIFT (MULTIPLE SPACECRAFT ORBIT SEPARATION PROGRAM)

This program calculates the velocity increments and firing angles necessary to provide a specified angular separation between two spacecraft in a circular orbit. The following two options are available.

A.2.1 SLOW TRANSFER. For this option, a tangential burn is assumed, and the velocity increments and transfer time for a given angular separation is output for one through seven passages in the transfer orbit. Negative or positive separation angles may be chosen, and the transfer orbit perigees and apogees are also output to verify feasibility of each case. A second burn of equal magnitude and opposite direction is required at the end of the coast period to acquire the original orbit.

A.2.2 FAST TRANSFER. Non-tangential burns are assumed for this option. Positive pitch angle of attack during the first transfer burn causes the secondary spacecraft to climb to an apogee above the primary spacecraft orbit and drop behind (negative separation); conversely, negative pitch angle of attack causes the secondary spacecraft to drop to a perigee below the primary spacecraft orbit and move ahead (positive separation). The second transfer burn is of equal magnitude and pitch angle of attack to the first burn. Apogee and perigee altitudes of the transfer orbit and the true anomalies of the burn locations in the transfer orbit are output, in addition to the burn vector magnitude and pitch angle of attack.

A Newton-Raphson iteration subroutine is incorporated for use with both transfer modes. With the slow transfer, it is used to adjust the burn velocity increment until the desired separation angle is achieved. With the fast transfer, the transfer orbit apside radius is adjusted until the desired transfer time is achieved.

A.3 TIP (TRAJECTORY INTEGRATION PROGRAM)

This program determines the end conditions obtained with a given launch vehicle/payload combination departing from a given position in a departure orbit. The departure orbit is defined by perigee and apogee altitudes; end conditions are ideal velocity, velocity loss, orbital energy (C3), and equivalent circular velocity excess at the departure altitude. Time-referenced output of orbit parameters (altitude, velocity, flight path angle, and central range angle) and vehicle weight and angle of attack are available at any desired increments. Provision is made for addition of one or two upper stage vehicles with a specified coast time between burns. A single burn of the primary vehicle is assumed, with no provision for out-of-plane orientation of any stage. Pitch steering is referenced to

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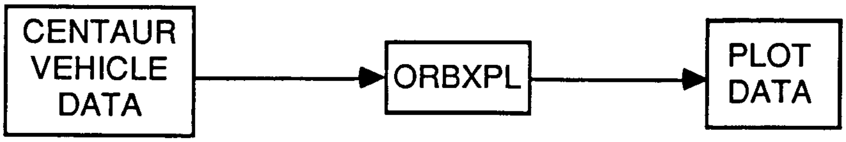
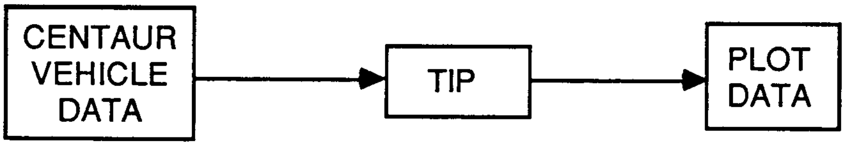
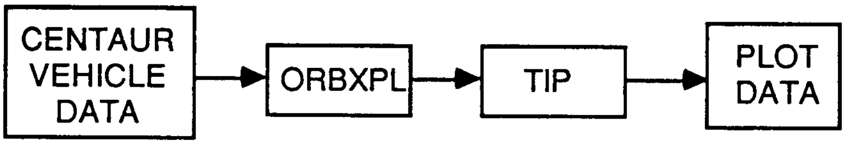


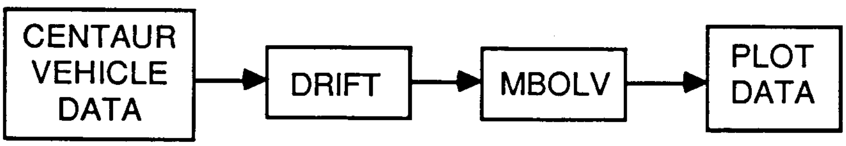
ORBXPL - Provides single payload plane change performance, altitude performance, circular velocity excess capability and multiple payloads to different orbits capability.



TIP - Calculates maximum C3 capability given a payload weight, used for interplanetary predictions.



ORBXPL/TIP - Calculates the maximum interplanetary satellite deployment capability available after delivering an earth satellite to a specified orbit.



DRIFT/MBOLV - Computes required Delta-V and derives satellite weights for placing multiple satellites in the same orbit, separated by a given phase angle.

Figure A-1. Four Computer Programs Were Used for Our SBTC Performance Evaluation the inertial velocity vector, with provision for a given initial angle-of-attack and a time-referenced angle-of-attack rate. Integration of the basic differential flight equations is accomplished with a Runge-Kutta subroutine. Vehicle-related input is made from a storage file which may be edited as necessary. Payload weight, print interval, and pitch control parameters are made available for keyboard input.

A.4 MBOLV (MULTI-BURN ORBIT-LAUNCHED VEHICLE PROGRAM)

This program determines either:

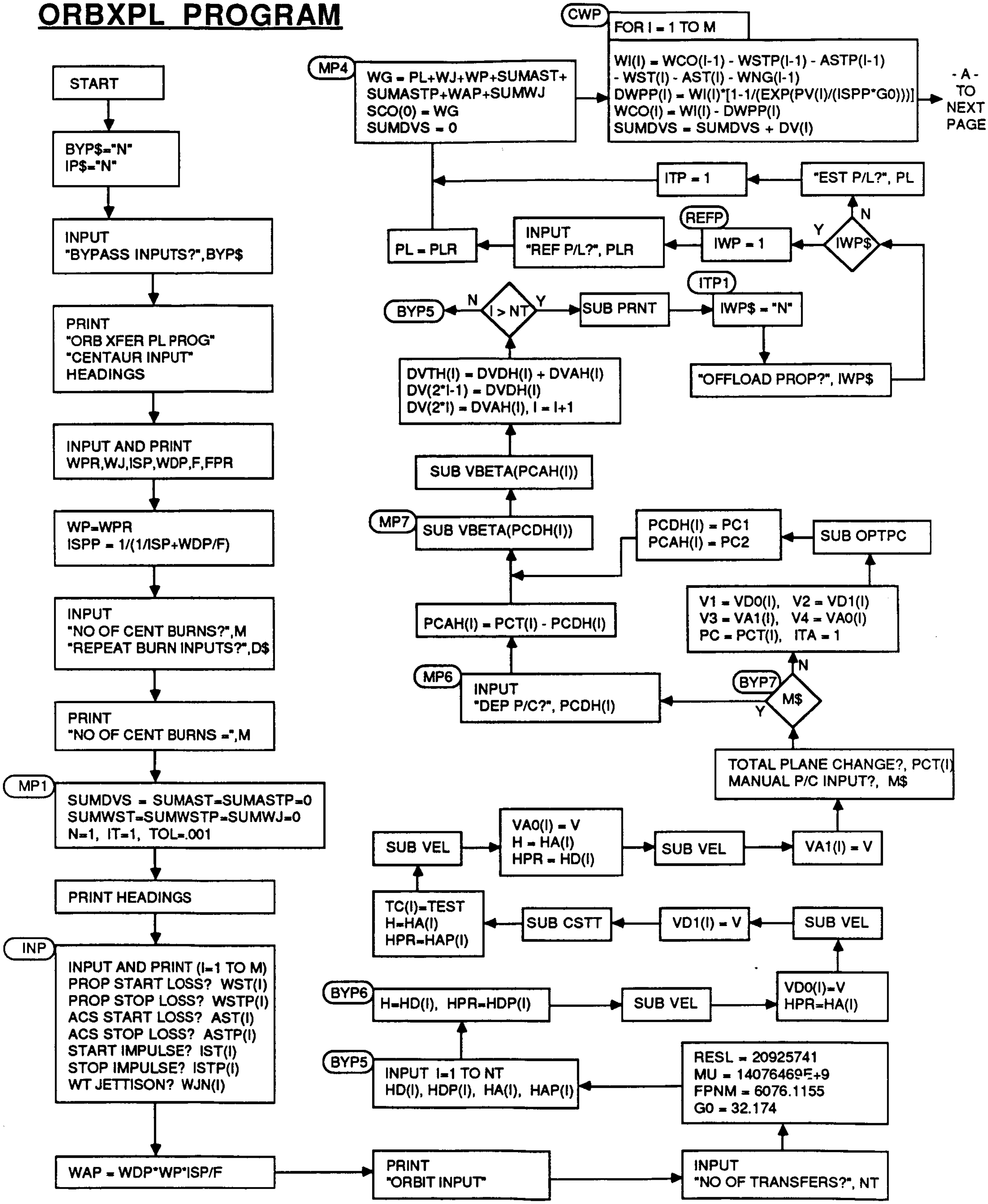
- Payload capability with fixed payload, or for fixed gross weight
- Jettison weight capability after any burn
- Payload capability and propellant offload for fixed gross weight

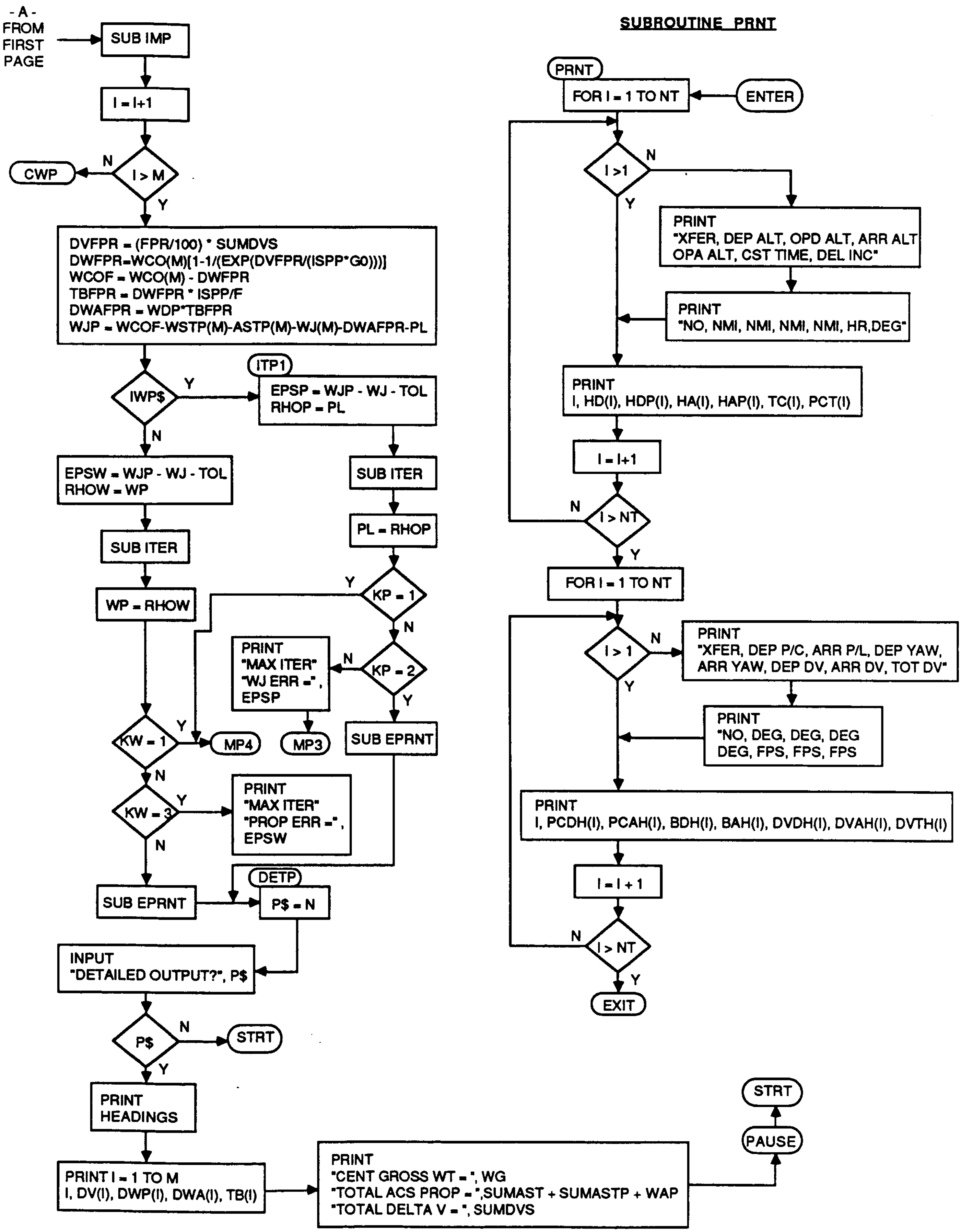
for an orbit-launched vehicle with multi-burn capability. Start and stop losses of main impulse propellants and auxiliary propellants and start and stop impulses of the motors are incorporated. The program is also capable of providing for venting of a given percentage of the remaining propellants prior to each burn. Burn times, propellant usages, vent quantities and velocity increments from start and stop impulses are supplied as output. Flight performance reserve propellants are retained to supply a velocity reserve which is a given percentage of the total ideal velocity increment. Since this program only computes the mass ratios necessary to supply the given velocity increments, it is independent of the initial orbit characteristics and velocity losses incurred during the burns. If velocity losses are known, they may be added to the input velocity increment to improve accuracy of the final solution.

The significant new capabilities resulting from these analyses will be presented in the following sections. The format will consist of an introductory figure with an example of how the results may be used from that area of the performance analysis. The complete set of actual performance plots are included in Appendix B.

The performance analyses on the SBTC concept required the use of four computer programs. These programs are described in this Appendix. Each of the programs has an introductory description and has been flowcharted along with a complete list of the variables defined.

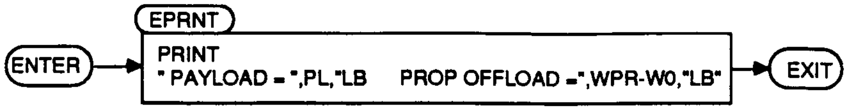
ORBXPL PROGRAM



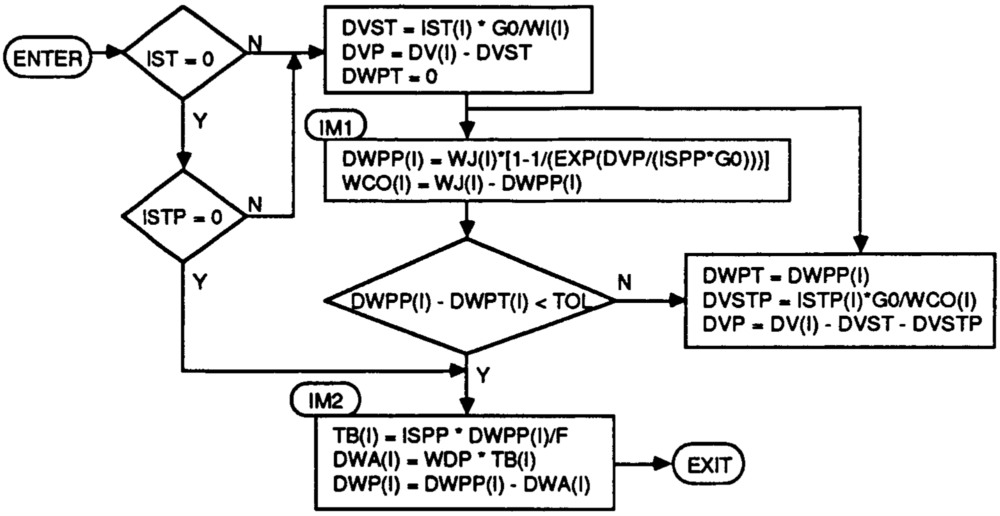


- A -
FROM FIRST PAGE
SUB IMP
I = I+1
I > M
N
CWP
Y
DVFPR = (FPR/100) * SUMDVS
DWFPR=WCO(M)[1-1/(EXP(DVFPR/(ISPP*G0)))]
WCOF = WCO(M) - DWFPR
TBFPR = DWFPR * ISPP/F
DWAFPR = WDP*TBFPR
WJP = WCOF-WSTP(M)-ASTP(M)-WJ(M)-DWAFPR-PL
IWP$
Y
ITP1
EPSP = WJP - WJ - TOL
RHOP = PL
N
EPSW = WJP - WJ - TOL
RHOW = WP
SUB ITER
SUB ITER
PL = RHOP
WP = RHOW
KP = 1
Y
N
KP = 2
N
PRINT
"MAX ITER"
"WJ ERR =",
EPSP
Y
SUB EPRNT
KW = 1
Y
MP4
MP3
N
KW = 3
Y
PRINT
"MAX ITER"
"PROP ERR =",
EPSW
N
DETP
SUB EPRNT
P$ = N
INPUT
"DETAILED OUTPUT?", P$
P$
N
STRT
Y
PRINT HEADINGS
PRINT I = 1 TO M
I, DV(I), DWP(I), DWA(I), TB(I)
PRINT
"CENT GROSS WT = ", WG
"TOTAL ACS PROP = ",SUMAST + SUMASTP + WAP
"TOTAL DELTA V = ", SUMDVS
PAUSE
STRT
SUBROUTINE PRNT
PRNT
FOR I = 1 TO NT
ENTER
I >1
N
Y
PRINT
"XFER, DEP ALT, OPD ALT, ARR ALT
OPA ALT, CST TIME, DEL INC"
PRINT
"NO, NMI, NMI, NMI, NMI, HR,DEG"
PRINT
I, HD(I), HDP(I), HA(I), HAP(I), TC(I), PCT(I)
I = I+1
I > NT
N
Y
FOR I = 1 TO NT
I > 1
N
Y
PRINT
"XFER, DEP P/C, ARR P/L, DEP YAW,
ARR YAW, DEP DV, ARR DV, TOT DV"
PRINT
"NO, DEG, DEG, DEG
DEG, FPS, FPS, FPS
PRINT
I, PCDH(I), PCAH(I), BDH(I), BAH(I), DVDH(I), DVAH(I), DVTH(I)
I = I + 1
I > NT
N
Y
EXIT

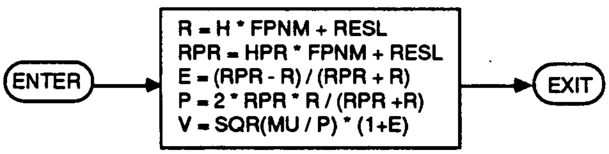
SUBROUTINE EPRNT



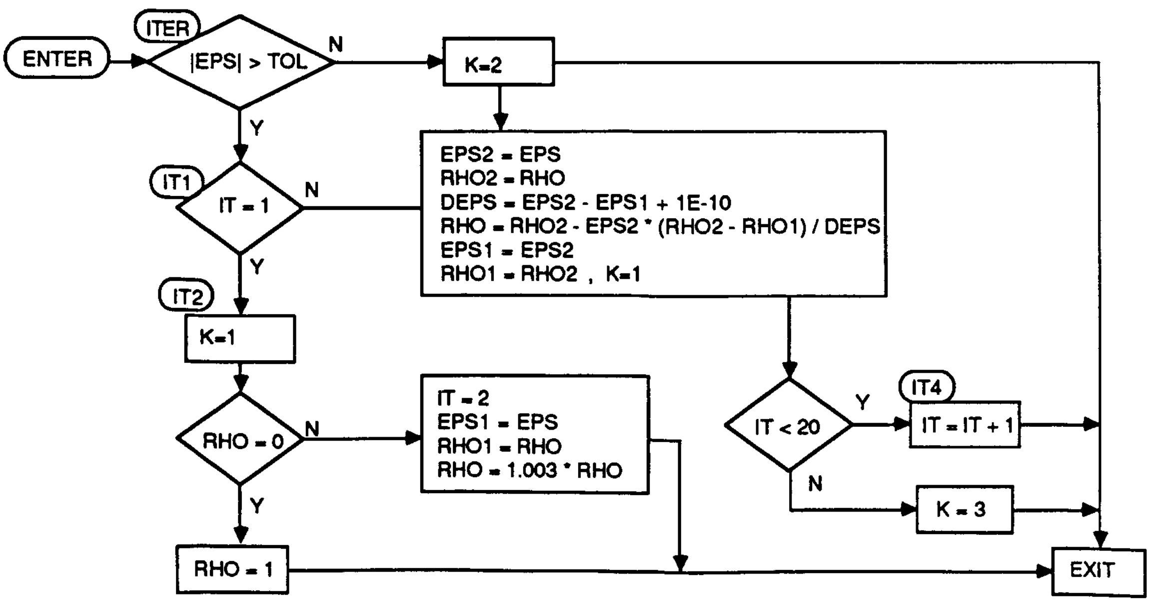
SUBROUTINE IMPULSE



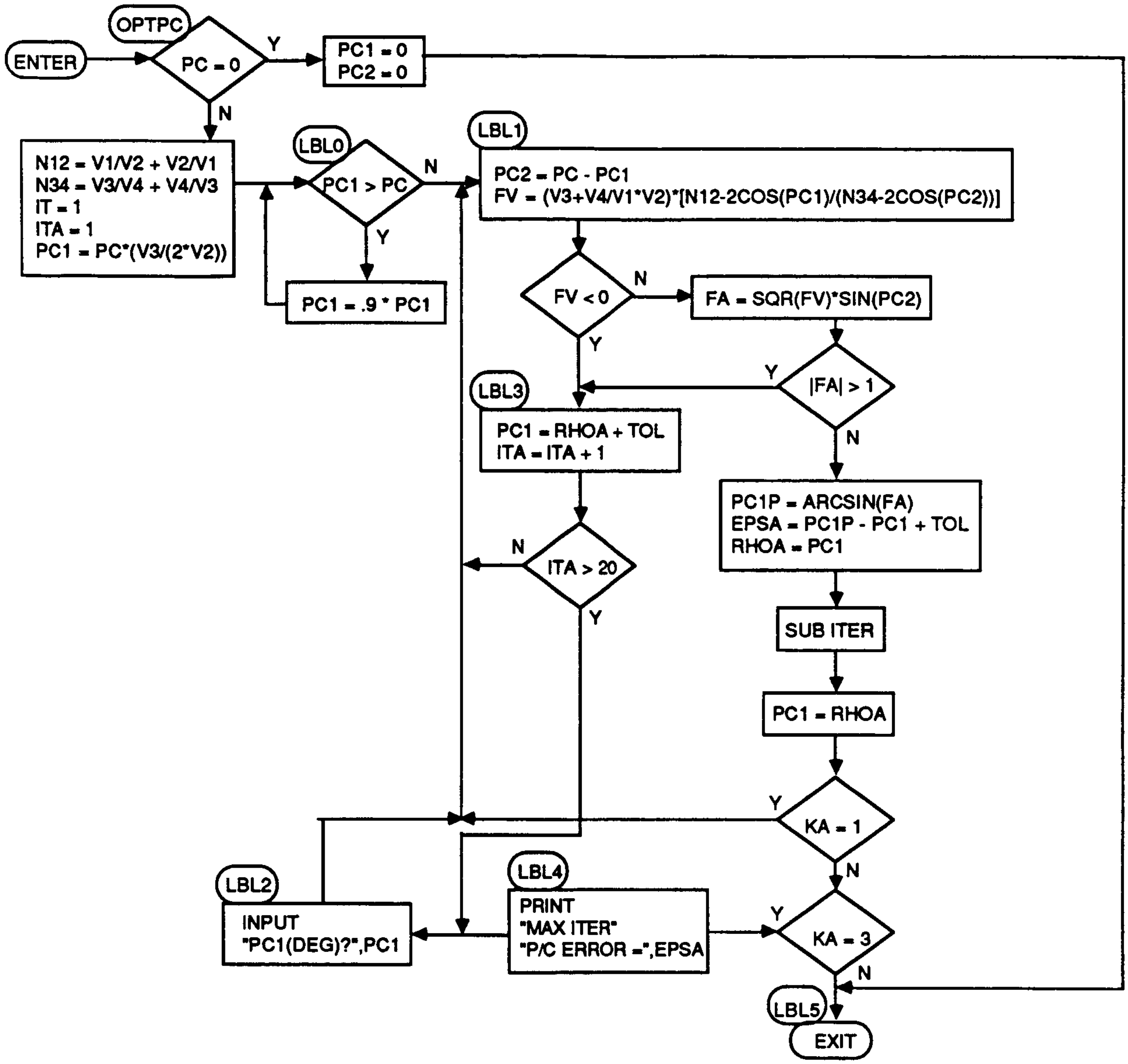
SUBROUTINE VEL



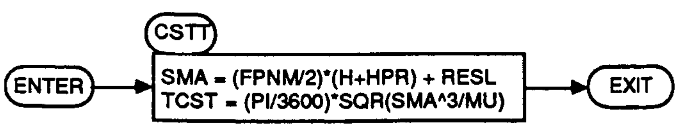
SUBROUTINE ITER



SUBROUTINE OPTPC



SUBROUTINE CSTT

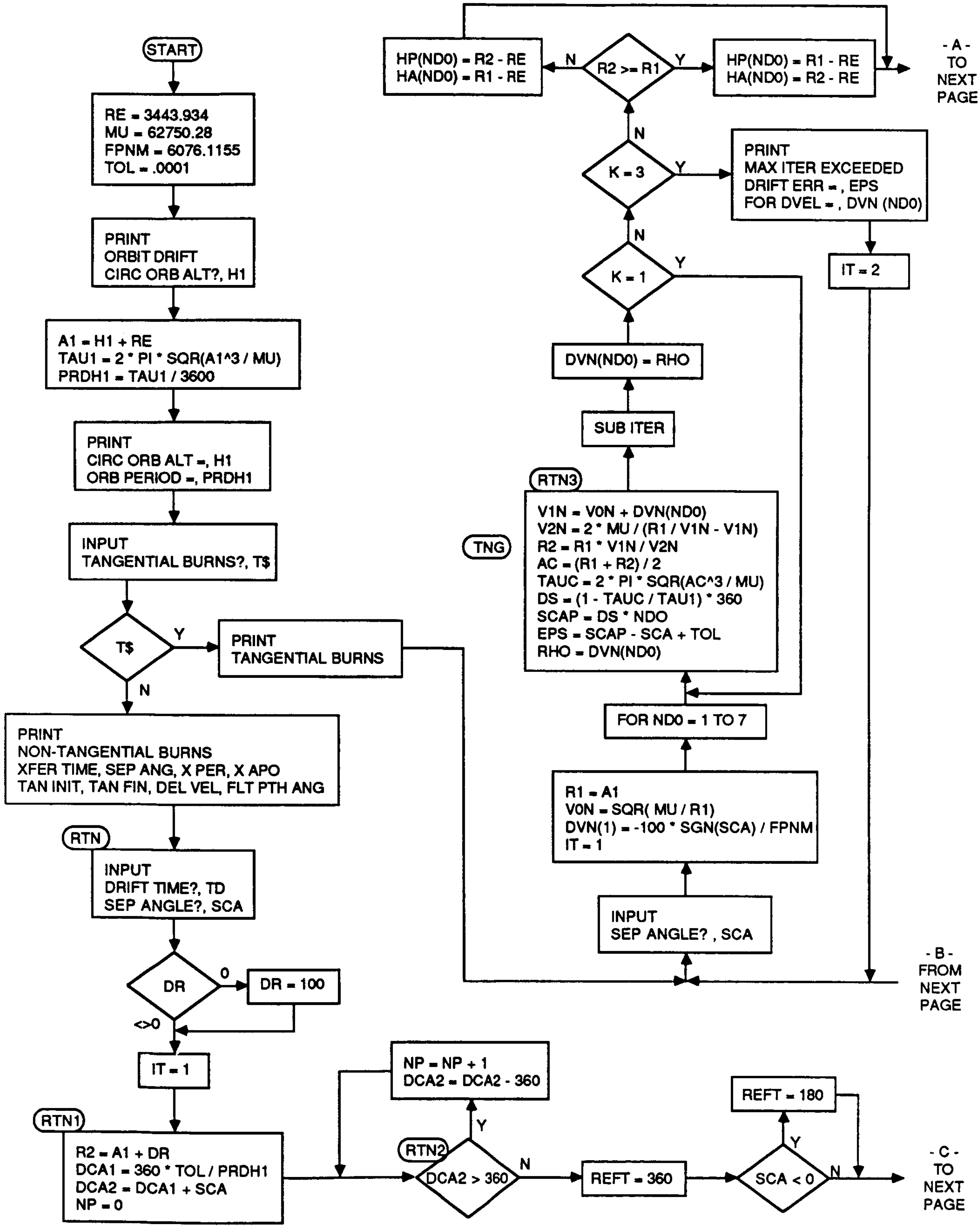


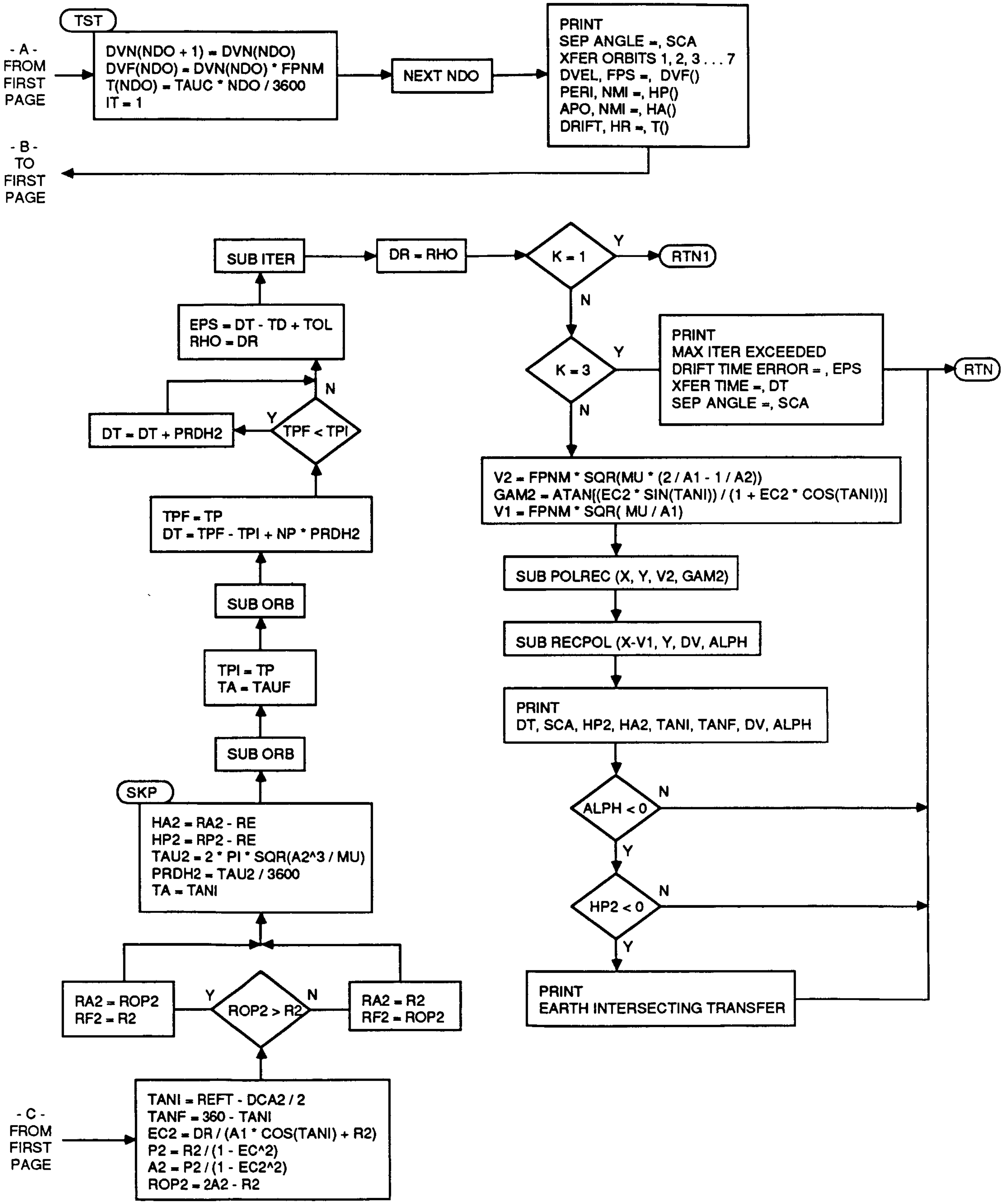
Variable	Description
AST()	RCS Start Loss
ASTP()	RCS Stop Loss
BYP$	Flag to bypass statements
D$	Flag to repeat inputs for same burn
DEPS	Delta Epsilon
DGRD	Deg to Rad Conversion 57.2957795 Deg/Rad
DV()	Delta Velocity
DVFPR	Delta Velocity Reserved for Dispersions (FPR)
DVP	Delta Velocity Supplied for Steady State Burn
DVST	Delta Velocity During Engine Start Sequence
DWA()	Boost Pump Propellant used During Steady State Burn
DWAFPR	Boost Pump Propellant used During FPR Burn
DWFPR	Propellant Reserved for Dispersions
DWP()	Propellant Comsumed During Steady State Burn
DWPP()	Propellant (+RCS) Consumed During Steady State Burn
DWPT	Iterated Value of DWPP
DWSTP	Delta Velocity During Engine Stop Sequence
E	Orbit Eccentricity
EPS	Error in Desired Value of Dependent Variable
EPS1	Error in Desired Value of Dependent Variable
EPS2	Error in Desired Value of Dependent Variable
EPSA	Iteration Error in Plane Change Angle
EPSP	Iteration Error in Jettison Weight Versus Payload
EPSW	Iteration Error in Jettison Weight Versus Propellant
F	Force or Thrust of Engines
FA	Function of Velocities and Plane Change Angles
FPNM	Feet per Nautical Mile Conversion 6076.1155
FPR	Flight Performance Reserve Propellant Ratio (% Δ Vt)
FV	Function of Velocities and Plane Change Angles
GO	Gravity at Sea Level 32.174 Ft/Sec^2 9.81 M/Sec^2
H	Initial Altitude for Subroutine VEL
HA()	Arrival Altitude for I th Transfer
HAP()	Altitude Opposite Arrival Apside I th Transfer
HD()	Departure Altitude for I th Transfer
HDP()	Altitude Opposite Departure Apside I th Transfer
HPR	Final Velocity for Subroutine VEL
I	Iteration Counter
ISP	Specific Impulse
ISPP	Specific Impulse Corrected for Boost Pump Flow
IST()	Engine Start Impulse
ISTP()	Engine Stop Impulse
IT	Iteration Counter
ITA	Iteration Counter
IWP$	Set to Y to Calculate Propellant Offload
K	Flag for Iteration 1 = Continue 2 = Tol met 3 = Too Many Iter
KA	Flag for Iteration 1 = Continue 2 = Tol met 3 = Too Many Iter
KP	Flag for Iteration 1 = Continue 2 = Tol met 3 = Too Many Iter
KW	Flag for Iteration 1 = Continue 2 = Tol met 3 = Too Many Iter

Variable	Description
M	Number of Engine Burns
MU	Gravitational Parameter For Earth 1.4076469E+16 ft^3/sec^:
N12	Function of V1 and V2
N34	Function of V3 and V4
NT	Number of Orbit Transfers
P	Semi-Latus Rectum of Orbit
P$	Print Flag for Detailed Output
PC1	Plange Change During First Transfer Burn
PC1P	Plane Change Angle as Function of Velocities and Angles (FA)
PC2	Plane Change During Circularization Burn
PCAH()	Arrival Plane Change Angle - I th Transfer
PCDH()	Total Plane Change Angle - I th Transfer
PCT()	Departure Plane Change Angle - I th Transfer
PI	Constant 3.141592654
PL	Payload Weight Input
PLR	Reference Payload
R	Radius from Earth Center
RESL	Radius of Earth 20,925,741 ft 6378.145 km
RHO1	Value of Dependent Variable
RHO2	Value of Dependent Variable
RHOP	Payload Estimate for Jettison Weight Iteration
RHOW	Propellant Estimate for Jettison Weight Vs Propellant
RPR	Radius of Opposing Apside
SMA	Semi-Major Axis of Orbit
SUMAST	Summation of ACS Start Losses
SUMASTP	Summation of ACS Stop Losses
SUMDVS	Summation of Delta Velocities
SUMWJ	Summation of Jettioned Weights
SUMWST	Summation of Prop Stop Losses
SUMWSTP	Summation of Prop Start Losses
TBFPR	Time Required to Consume FPR Propellants
TC()	Coast Time for I th Transfer
TCST	Coast Time - Sub CSTT
TOL	Tolerance in Dependent Variable
V	Velocity Output from Sub VEL
V1	Velocity Before Departure Burn in Sub OPTPC
V2	Velocity After Departure Burn in Sub OPTPC
V3	Velocity Before Arrival Burn in Sub OPTPC
V4	Velocity After Arrival Burn in Sub OPTPC
VA0()	Velocity Before Arrival Burn - I th Transfer
VA1()	Velocity After Arrival Burn - I th Transfer
VD0()	Velocity Before Departure Burn - I th Transfer
VD1()	Velocity After Departure Burn - I th Transfer
W0	Initial Weight
WAP	Total Boost Pump Propellant
WCO()	Weight at Cutoff at I th Burn
WCOF	Final Cutoff Weight (After FPR Prop Consumed)
WDP	Boost Pump Flow Rate

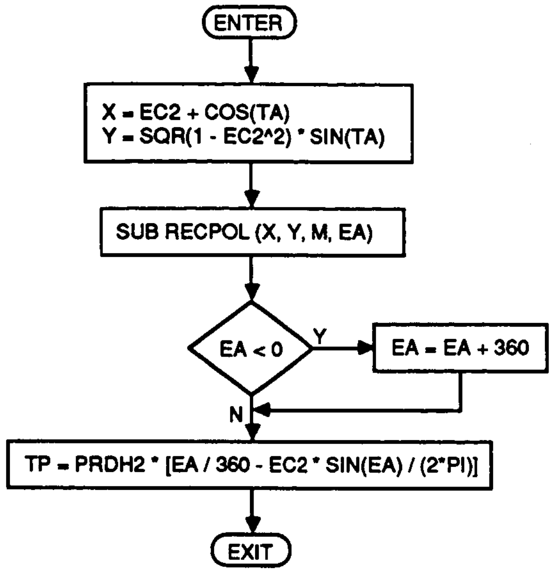
Variable	Description
WG	Gross Weight of Stage at Separation (Including Payload)
WI()	Weight at Start of I th Burn
WJ	Jettison Weight - Minimum Remaining Propellant
WJN()	Jettisoned Weight at end of I th Burn
WJP	Jettison Weight - Current Value
WP	Expendable Propellant Weight - Current Value
WPR	Expendable Propellant Weight - Reference Value
WST()	Propellant Start Loss - I th Burn
WSTP()	Propellant Stop Loss - I th Burn

DRIFT PROGRAM

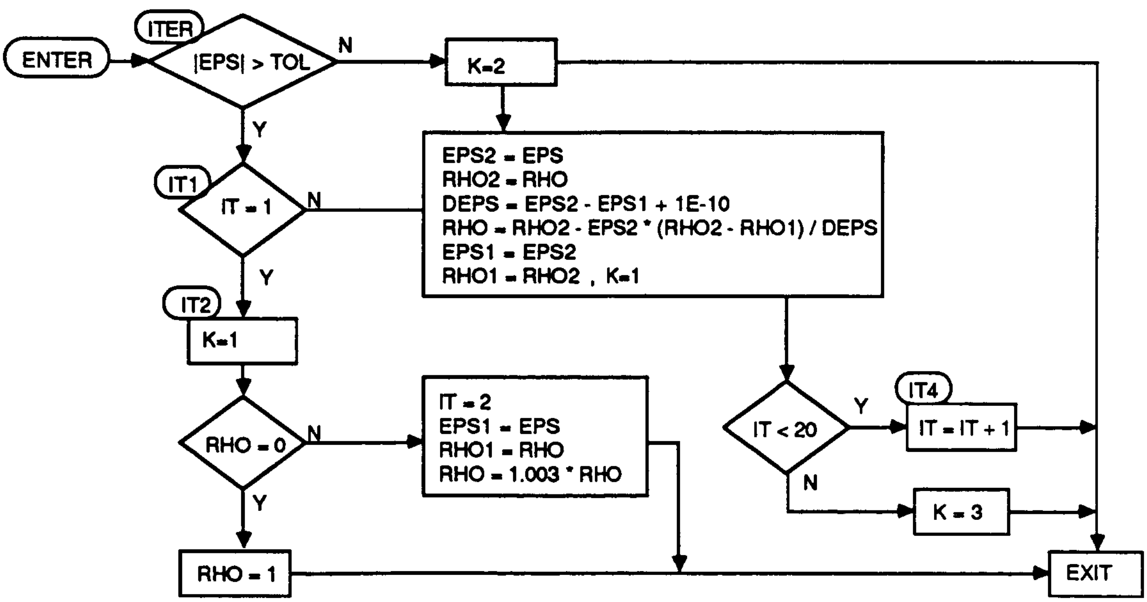



TST
- A - FROM FIRST PAGE
DVN(NDO + 1) = DVN(NDO)
DVF(NDO) = DVN(NDO) * FPNM
T(NDO) = TAUC * NDO / 3600
IT = 1
NEXT NDO
PRINT
SEP ANGLE =, SCA
XFER ORBITS 1, 2, 3 . . . 7
DVEL, FPS =, DVF()
PERI, NMI =, HP()
APO, NMI =, HA()
DRIFT, HR =, T()
- B - TO FIRST PAGE
SUB ITER
DR = RHO
K = 1
Y
RTN1
N
EPS = DT - TD + TOL
RHO = DR
K = 3
Y
N
PRINT
MAX ITER EXCEEDED
DRIFT TIME ERROR = , EPS
XFER TIME =, DT
SEP ANGLE =, SCA
RTN
N
DT = DT + PRDH2
Y
TPF < TPI
V2 = FPNM * SQR(MU * (2 / A1 - 1 / A2))
GAM2 = ATAN[(EC2 * SIN(TANI)) / (1 + EC2 * COS(TANI))]
V1 = FPNM * SQR(MU / A1)
TPF = TP
DT = TPF - TPI + NP * PRDH2
SUB POLREC (X, Y, V2, GAM2)
SUB ORB
SUB RECPOL (X-V1, Y, DV, ALPH
TPI = TP
TA = TAUF
PRINT
DT, SCA, HP2, HA2, TANI, TANF, DV, ALPH
SUB ORB
SKP
HA2 = RA2 - RE
HP2 = RP2 - RE
TAU2 = 2 * PI * SQR(A2^3 / MU)
PRDH2 = TAU2 / 3600
TA = TANI
ALPH < 0
N
Y
HP2 < 0
N
Y
RA2 = ROP2
RF2 = R2
Y
ROP2 > R2
N
RA2 = R2
RF2 = ROP2
PRINT
EARTH INTERSECTING TRANSFER
- C - FROM FIRST PAGE
TANI = REFT - DCA2 / 2
TANF = 360 - TANI
EC2 = DR / (A1 * COS(TANI) + R2)
P2 = R2 / (1 - EC^2)
A2 = P2 / (1 - EC2^2)
ROP2 = 2A2 - R2

SUBROUTINE ORB



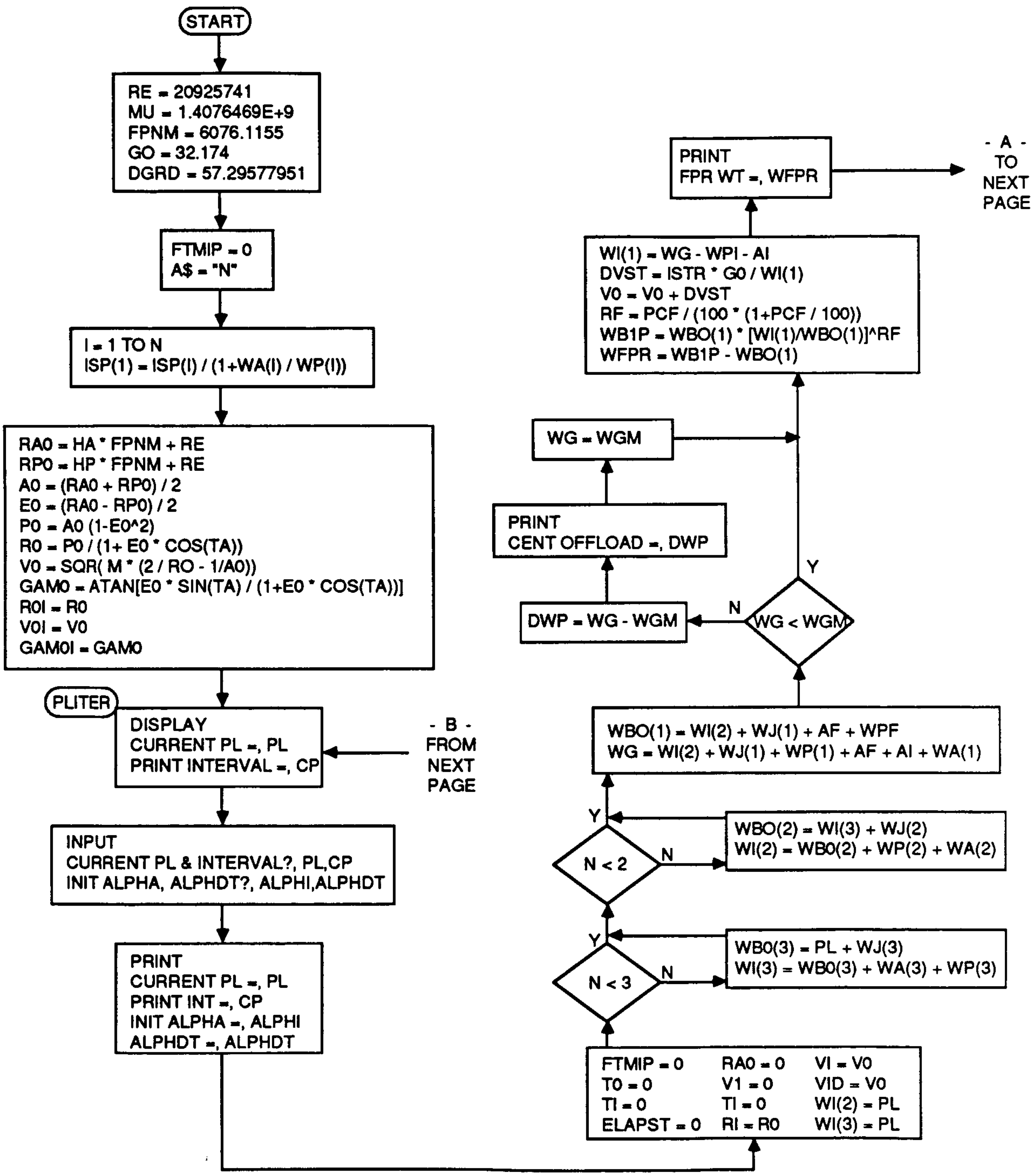
SUBROUTINE ITER

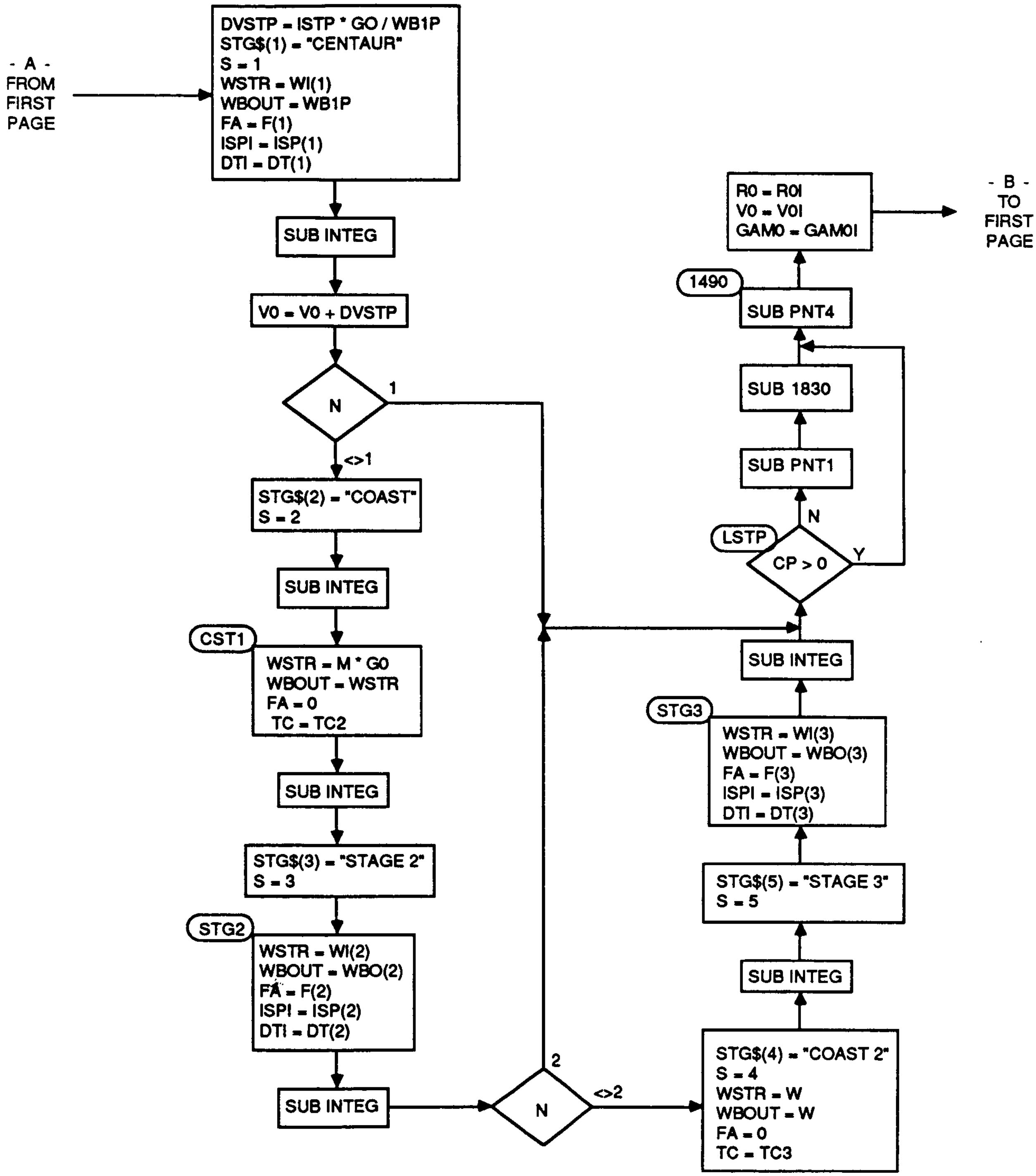


Variable	Description
A1	Radius of Reference Circular Orbit
A2	Semi-Major Axis of Transfer Orbit (Fast Transfer)
AC	Semi-Major Axis of Transfer Orbit (Slow Transfer)
ALPH	Pitch Angle of Attack for Transfer burn
DCA1	Central Angle Traversed in Ref. Circ. Orbit During Transfer Time
DCA2	Central Angle Traversed in Transfer Orbit During Transfer Time
DEPS	Delta Epsilon
DR	Delta Radius of 1st Transfer Apside from Ref. Orb. Radius
DS	Angular Separation After 1 Orbit (Slow Transfer)
DT	Drift Time During transfer (Fast Transfer)
DV	Velocity Increment
DVF()	Delta velocity of Ith Transfer Orbit (fps)
DVN()	Delta velocity of Ith Transfer Orbit (nmi/sec)
EA	Angle of Vector in Rect to Pol Conversion
EC2	Transfer Orbit Eccentricity (Fast Transfer)
EPS	Error in Desired Value of Dependent Variable
EPS1	Error in Desired Value of Dependent Variable
EPS2	Error in Desired Value of Dependent Variable
FPNM	Feet per Nautical Mile Conversion 6076.1155
GAM2	Flight Path Angle in T/O after 1st Burn (Fast Transfer)
H$	Print Flag
H1	Circular Orbit Altitude
HA()	Apogee of Ith Transfer Orbit
HA2	Transfer Orbit Apogee Altitude
HP()	Perigee of Ith Transfer Orbit
HP2	Transfer Orbit Perigee Altitude
IT	Iteration Counter
K	Flag for Iteration 1 = Continue 2 = Tol Met 3 = Too Many Iter
M	Magnitude of Vector in Rect to Polar Conversion
MU	Gravitational Parameter For Earth=1.4076469E+16 ft^3/sec^2
NP	Number of Transfer Orbit Passages
P2	Orbit Parameter (Fast Transfer)
PI	Constant 3.141592654
PRDH1	Orbital Period of Reference Circular Orbit (hr)
PRDH2	Transfer Orbit Period (hr)
R1	Radius of Slow Transfer Apside at Departure Burn
R2	Radius of 1st Transfer Apside (Fast Transfer)
R2	Radius of Opposite Apside (Slow Transfer)
RA2	Apogee Radius of Transfer Orbit (Fast Transfer)
RE	Radius of Earth 3443.934 Nmi
REFT	Ref. True Anomaly for Fast Transfer
RHO	Value of Dependent Variable
RHO1	Value of Dependent Variable
RHO2	Value of Dependent Variable
ROP2	Second Apside Radius (Fast Transfer)
RP2	Perigee Radius (Fast Transfer)
SCA	Separation Angle Between Spacecraft
SCAP	Angular Separation After N Orbits (Slow Transfer)

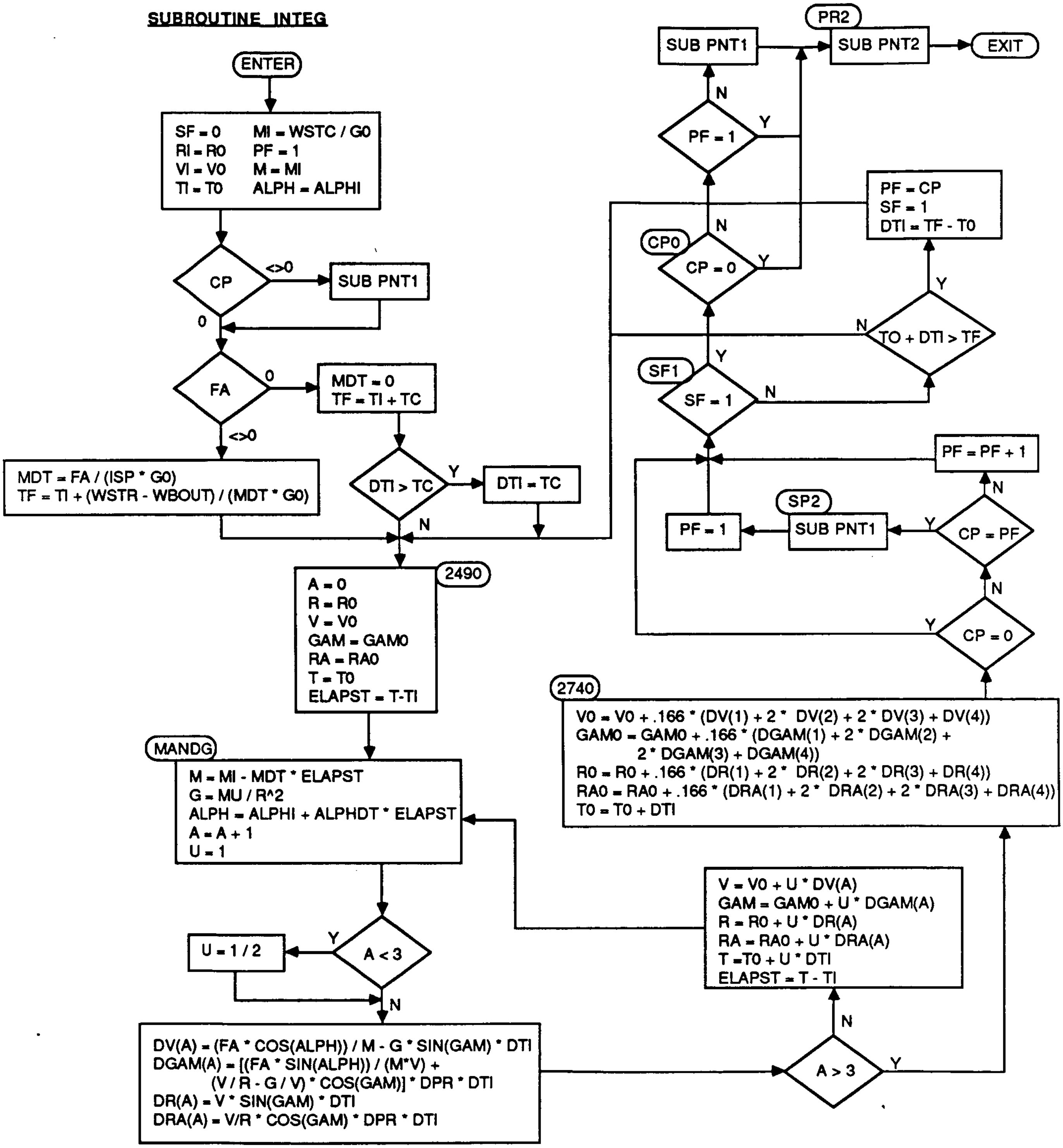
Variable	Description
T$	Flag for Tangential Transfer Burn (Slow Transfer)
T()	Drift
TA	True Anomaly
TANF	True Anomaly at 1st Burn (Fast Transfer)
TANI	True Anomaly at 2nd Burn (Fast Transfer)
TAU1	Period of Ref. Circular Orbit (sec)
TAU2	Transfer Orbit Period (sec)
TAUC	Period of Slow Transfer Orbit (sec)
TD	Drift Time (hr)
TOL	Tolerance in Dependent Variable
TP	Time from Perigee
TPF	Time from Perigee of T/O (2nd Burn)
TPI	Time from Perigee of T/O (1st Burn)
V0N	Velocity in Ref Orbit (Slow Transfer)
V1	Velocity in Ref Orbit (Fast Transfer)
V1N	Velocity after 1st Burn (Slow Transfer)
V2	Velocity in T/O after 1st Burn (Fast Transfer)
V2N	Velocity at Opposite Apside After 1st Burn (Slow Transfer)
X	X value in Cartesian Coords
Y	Y value in Cartesian Coords

TIP PROGRAM

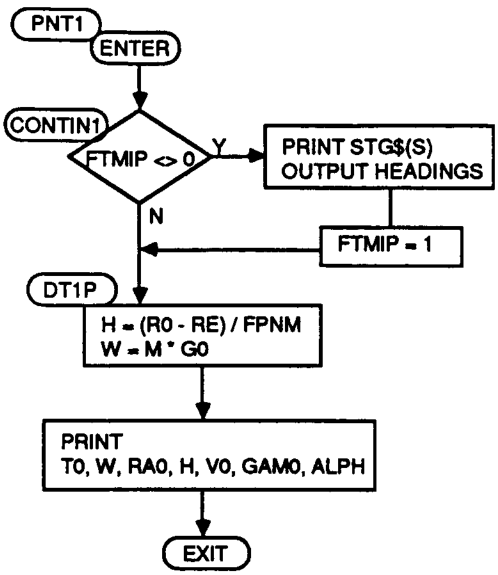

START
RE = 20925741
MU = 1.4076469E+9
FPNM = 6076.1155
GO = 32.174
DGRD = 57.29577951
FTMIP = 0
A$ = "N"
I = 1 TO N
ISP(1) = ISP(I) / (1+WA(I) / WP(I))
RA0 = HA * FPNM + RE
RP0 = HP * FPNM + RE
A0 = (RA0 + RP0) / 2
E0 = (RA0 - RP0) / 2
P0 = A0 (1-E0^2)
R0 = P0 / (1+ E0 * COS(TA))
V0 = SQR(M * (2 / RO - 1/A0))
GAM0 = ATAN[E0 * SIN(TA) / (1+E0 * COS(TA))]
R0I = R0
V0I = V0
GAM0I = GAM0
PLITER
DISPLAY
CURRENT PL =, PL
PRINT INTERVAL =, CP
- B -
FROM
NEXT
PAGE
INPUT
CURRENT PL & INTERVAL?, PL,CP
INIT ALPHA, ALPHDT?, ALPHI,ALPHDT
PRINT
CURRENT PL =, PL
PRINT INT =, CP
INIT ALPHA =, ALPHI
ALPHDT =, ALPHDT
FTMIP = 0
T0 = 0
TI = 0
ELAPST = 0
RA0 = 0
V1 = 0
TI = 0
RI = R0
VI = V0
VID = V0
WI(2) = PL
WI(3) = PL
N < 3
Y
N
WB0(3) = PL + WJ(3)
WI(3) = WB0(3) + WA(3) + WP(3)
N < 2
Y
N
WB0(2) = WI(3) + WJ(2)
WI(2) = WB0(2) + WP(2) + WA(2)
WB0(1) = WI(2) + WJ(1) + AF + WPF
WG = WI(2) + WJ(1) + WP(1) + AF + AI + WA(1)
WG < WGM
Y
N
DWP = WG - WGM
PRINT
CENT OFFLOAD =, DWP
WG = WGM
WI(1) = WG - WPI - AI
DVST = ISTR * G0 / WI(1)
V0 = V0 + DVST
RF = PCF / (100 * (1+PCF / 100))
WB1P = WB0(1) * [WI(1)/WB0(1)]^RF
WFPR = WB1P - WB0(1)
PRINT
FPR WT =, WFPR
- A -
TO
NEXT
PAGE


- A -
FROM
FIRST
PAGE
DVSTP = ISTP * GO / WB1P
STG$(1) = "CENTAUR"
S = 1
WSTR = WI(1)
WBOUT = WB1P
FA = F(1)
ISPI = ISP(1)
DTI = DT(1)
SUB INTEG
V0 = V0 + DVSTP
N
1
<>1
STG$(2) = "COAST"
S = 2
SUB INTEG
CST1
WSTR = M * G0
WBOUT = WSTR
FA = 0
TC = TC2
SUB INTEG
STG$(3) = "STAGE 2"
S = 3
STG2
WSTR = WI(2)
WBOUT = WBO(2)
FA = F(2)
ISPI = ISP(2)
DTI = DT(2)
SUB INTEG
N
2
<>2
STG$(4) = "COAST 2"
S = 4
WSTR = W
WBOUT = W
FA = 0
TC = TC3
SUB INTEG
STG$(5) = "STAGE 3"
S = 5
STG3
WSTR = WI(3)
WBOUT = WBO(3)
FA = F(3)
ISPI = ISP(3)
DTI = DT(3)
SUB INTEG
LSTP
CP > 0
Y
N
SUB PNT1
SUB 1830
1490
SUB PNT4
R0 = R0I
V0 = V0I
GAM0 = GAM0I
- B -
TO
FIRST
PAGE

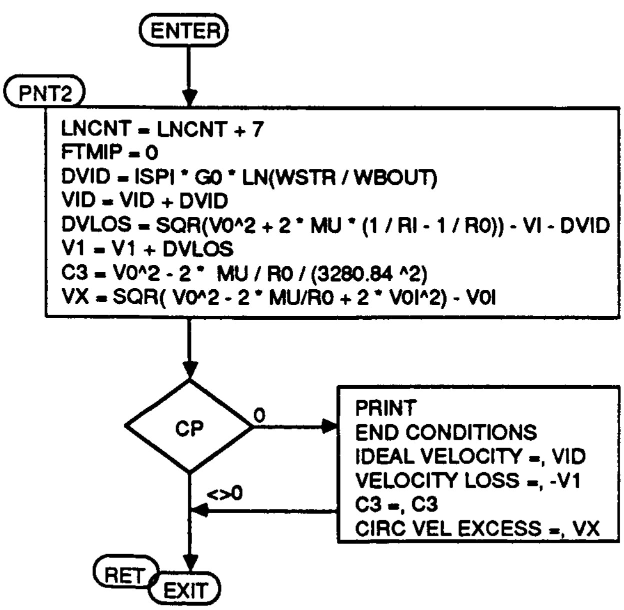
SUBROUTINE INTEG



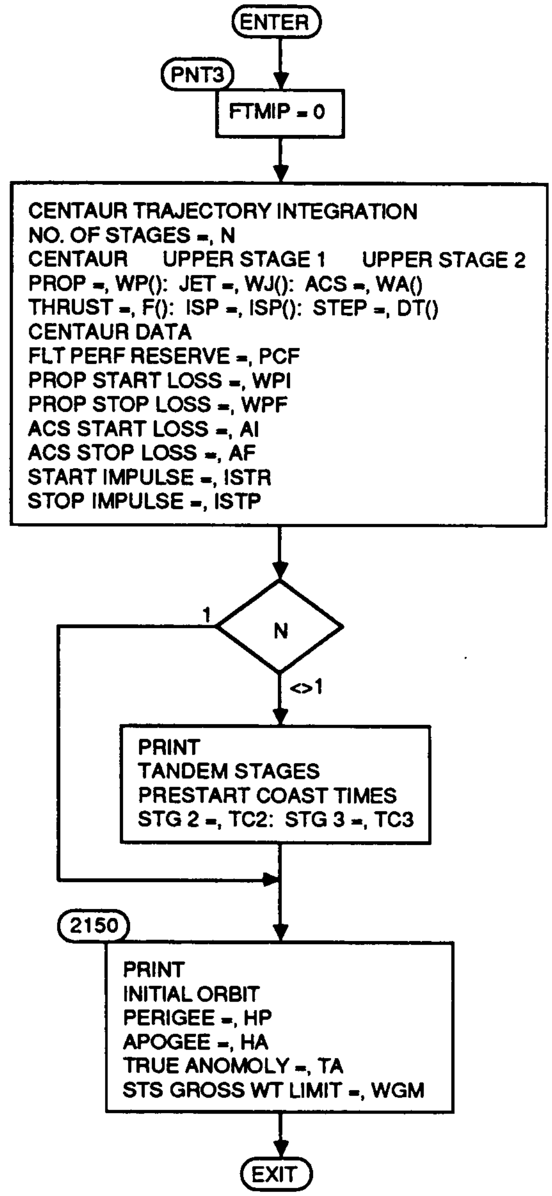
SUBROUTINE PNT1



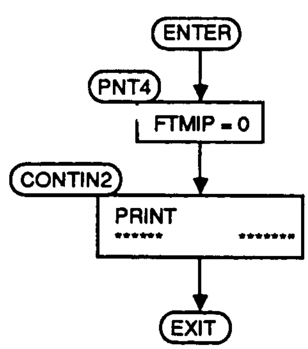
SUBROUTINE PNT2



SUBROUTINE PNT3



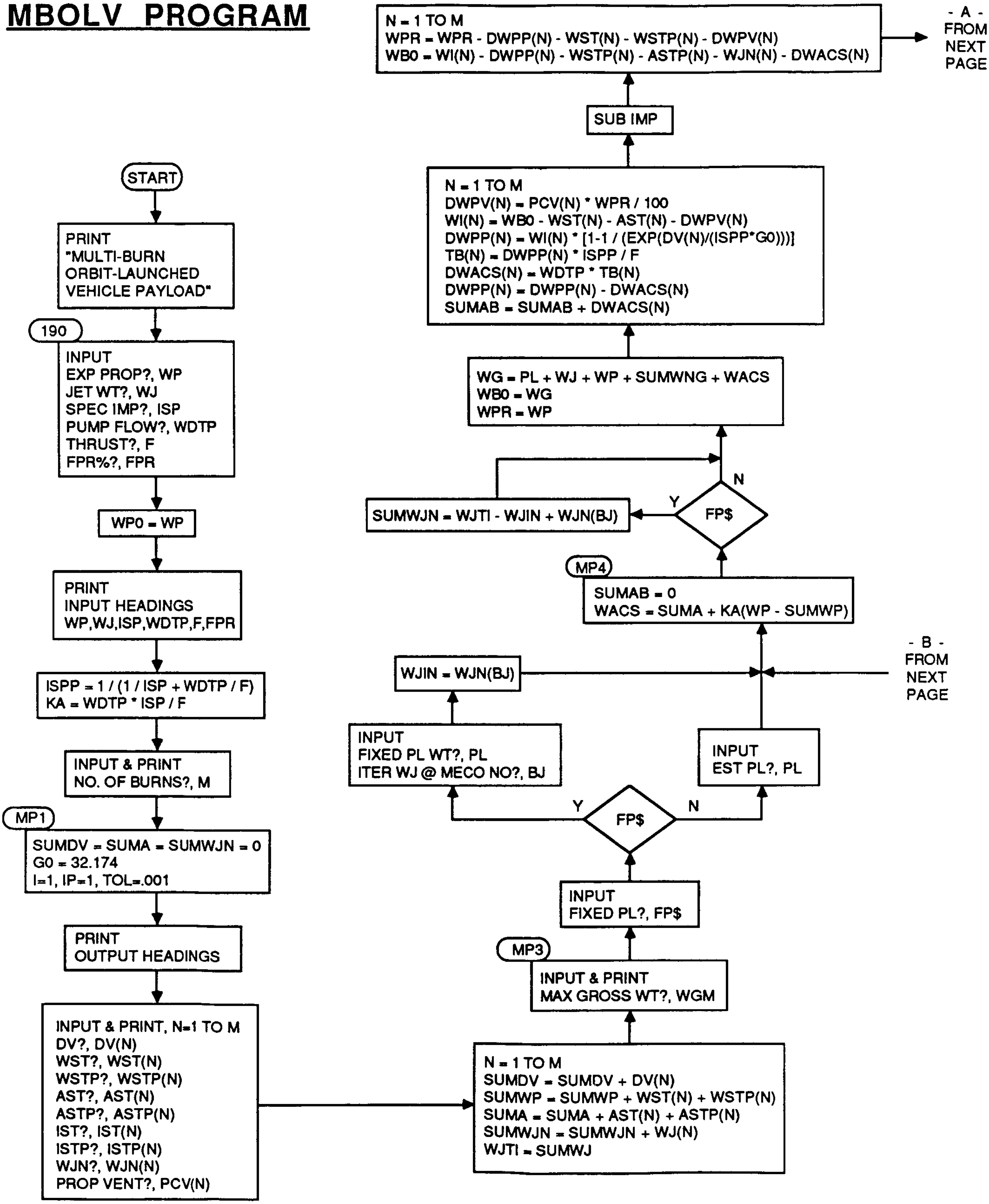
SUBROUTINE PNT4

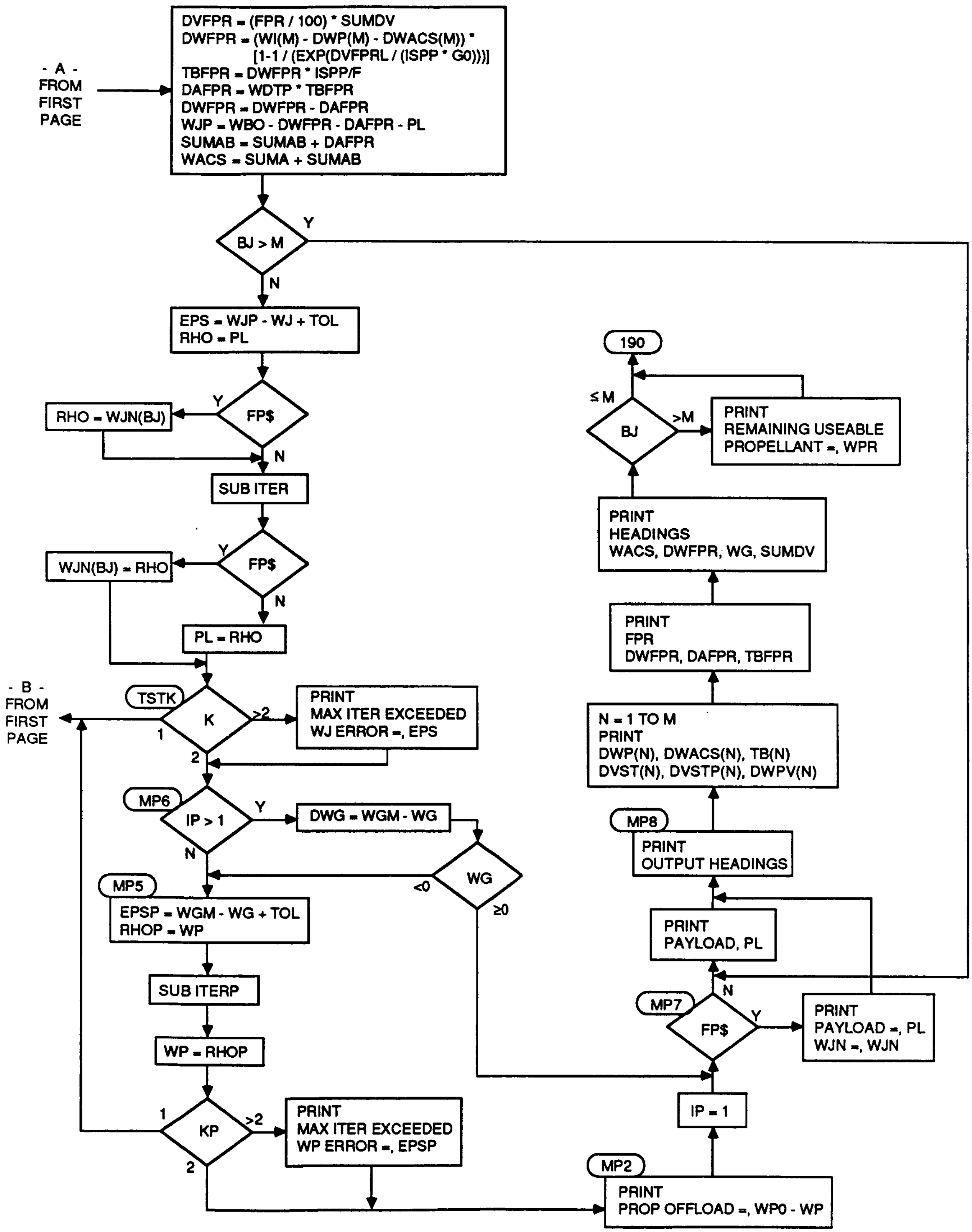


Variable	Description
A	Counter for Runge-Kutta Integration
A0	Initial Balue of Semi-Major Axis
AF	Auxiliary Propellant Loss During Engine Shutdown
AI	Auxiliary Propellant Loss During Engine Startup
ALPH	Pitch Angle of Attack
ALPHDT	Time Rate of Change of Angle of Attack
ALPHI	Initial Pitch Angle of Attack
C3	Orbital Energy Term
CP	Integration Cycles per Output Print
DGAM()	Delta Flight Path Angle for Ith Integration Step
DGRD	Deg to Rad Conversion 57.2957795 Deg/Rad
DR()	Delta Radius for Ith Integration Steop
DRA()	Delta Range Angle for I th Integration Step
DT()	Integration Stepsize in I th Section
DTI	Integration Stepsize in Current Section
DV()	Delta Velocity
DVID	Ideal Velocity Increment
DVLOS	Delta Velocity Loss
DVST	Delta Velocity During Engine Start Sequence
DVSTP	Delta Velocity During Engine Shutdown Sequence
DWP	Propellant Offloaded to Maintain Gross Weight
ELAPST	Time from start of Section
EO	Initial Value of Eccentricity
F()	Force or Thrust of Engines
FA	Thrust in Current Section
FPNM	Feet per Nautical Mile Conversion 6076.1155
FTMIP	Flag for Output Heading Print
GAM	Flight Path Angle
GAM0	Initial Flight Path Angle
GAM0I	Input Value of Initial Flight Path Angle
GO	Gravity at Sea Level 32.174 Ft/Sec^2 9.81 M/Sec^2
HA	Arrival Altitude for I th Transfer
HP	Perigee Altitude
ISP()	Specific Impulse
ISPI	Specific Impulse of Current Stage
ISTP	Engine Stop Impulse
ISTR	Engine Start Impulse
LNCNT	Printer Line Count
M	Current Value of Vehicle Mass
MDT	Time Rate of Mass Change
MU	Gravitational Parameter For Earth=1.4076469E+16 ft^3/sec^2
N	Number of Stages
P0	Initial Value of Orbit Parameter
PCF	Percentage of Total Velocity for Flight Performance Reserve
PF	Flag for Print Output
PL	Payload Weight Input
R	Radius from Earth Center
R0	Initial Orbit Radius

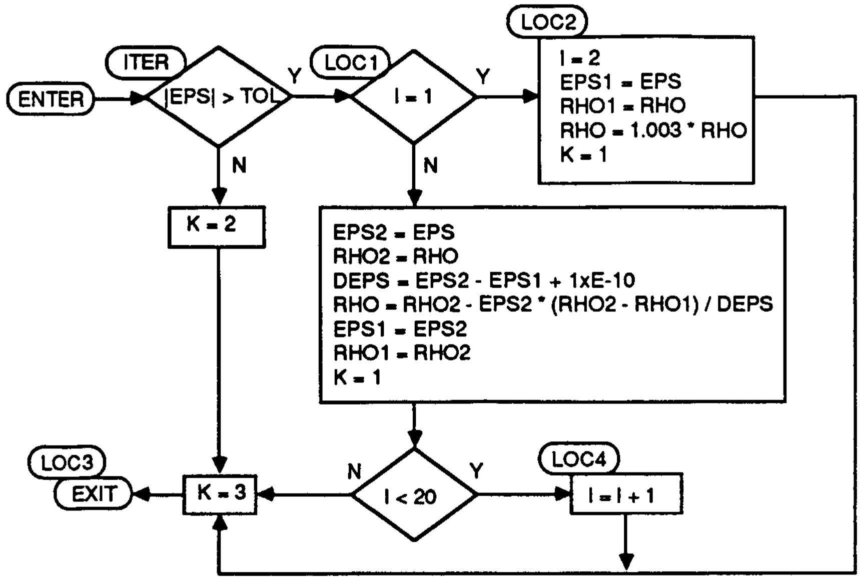
Variable	Description
RA0	Initial Apogee Radius
RE	Radius of Earth 3443.934 Nmi
RF	Mass Ratio to Provide Flight Performance Reserve
RI	Input Value of Orbit Radius
ROI	Input Value of Initial Orbit Radius
RP0	Initial Perigee Radius
S	Current Trajectory Section Number
SF	Flag for End of Section
STG$	Stage Heading
TA	True Anomaly in Departure Orbit
TC	Coast Time
TC2	Coast Time Before 2nd Stage Ignition
TC3	Coast Time Before 3rd Stage Ignition
TF	Time at End of Section
TI	Time at Start of Section
TO	Time from Start of First Burn
U	Multiplier for Runge-Kutta Summation
V	Velocity Output from Sub VEL
V0	Velocity at Current Time (or Initial Velocity)
V1	Velocity Before Departure Burn in Sub OPTPC
VI	Input Value of Velocity
VID	Ideal Velocity
VOI	Initial Value of Initial Velocity
VX	Velocity Increment in Excess of Circular Velocity
W	Vehicle Weight
WA()	Auxiliary Propellant of I th Stage
WB1P	Weight at Main Engine Cutoff
WBO()	Burnout Weight of I th Stage
WBOUT	Weight at End of Section
WFPR	Propellant Weight for Flight Performance Reserve
WG	Gross Weight of Stage at Separation (Including Payload)
WGM	Maximum Allowable Gross Weight
WI()	Weight at Start of I th Burn
WJ()	Jettison Weight of I th Stage
WP()	Expendable Propellant Weight of Ith Stage
WPF	Propellant Loss During Engine Shutdown
WPI	Propellant Loss During Engine Startup
WSTR	Weight at Start of Section

MBOLV PROGRAM

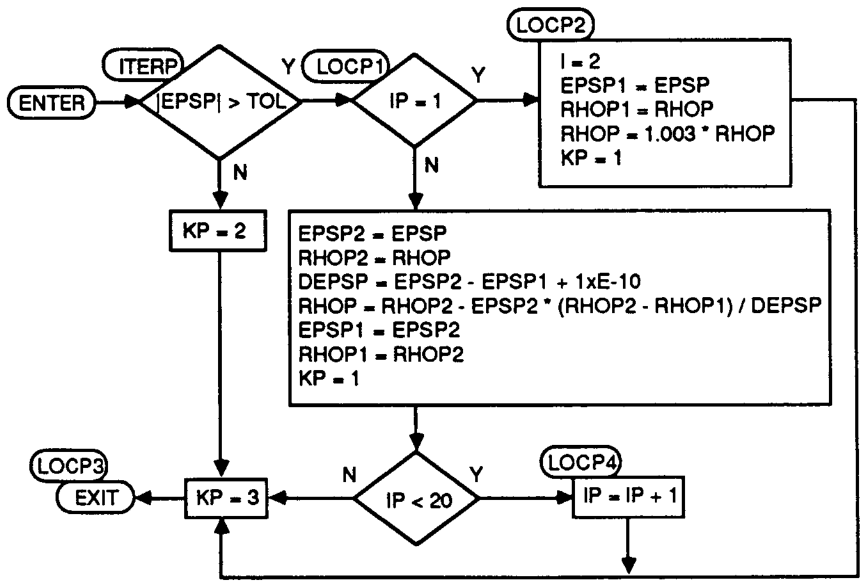



- A - FROM FIRST PAGE
DVFPR = (FPR / 100) * SUMDV
DWFPR = (WI(M) - DWP(M) - DWACS(M)) * [1-1 / (EXP(DVFPRL / (ISPP * G0)))]
TBFPR = DWFPR * ISPP/F
DAFPR = WDTP * TBFPR
DWFPR = DWFPR - DAFPR
WJP = WBO - DWFPR - DAFPR - PL
SUMAB = SUMAB + DAFPR
WACS = SUMA + SUMAB
BJ > M
Y
N
EPS = WJP - WJ + TOL
RHO = PL
FP$
RHO = WJN(BJ)
SUB ITER
WJN(BJ) = RHO
PL = RHO
TSTK
K
- B - FROM FIRST PAGE
PRINT MAX ITER EXCEEDED WJ ERROR =, EPS
MP6
IP > 1
DWG = WGM - WG
WG
<0
≥0
MP5
EPSP = WGM - WG + TOL
RHOP = WP
SUB ITERP
WP = RHOP
KP
PRINT MAX ITER EXCEEDED WP ERROR =, EPSP
MP2
PRINT PROP OFFLOAD =, WP0 - WP
IP = 1
MP7
PRINT PAYLOAD =, PL WJN =, WJN
PRINT PAYLOAD, PL
MP8
PRINT OUTPUT HEADINGS
N = 1 TO M
PRINT DWP(N), DWACS(N), TB(N) DVST(N), DVSTP(N), DWPV(N)
PRINT FPR DWFPR, DAFPR, TBFPR
PRINT HEADINGS WACS, DWFPR, WG, SUMDV
BJ
≤M
>M
PRINT REMAINING USEABLE PROPELLANT =, WPR
190

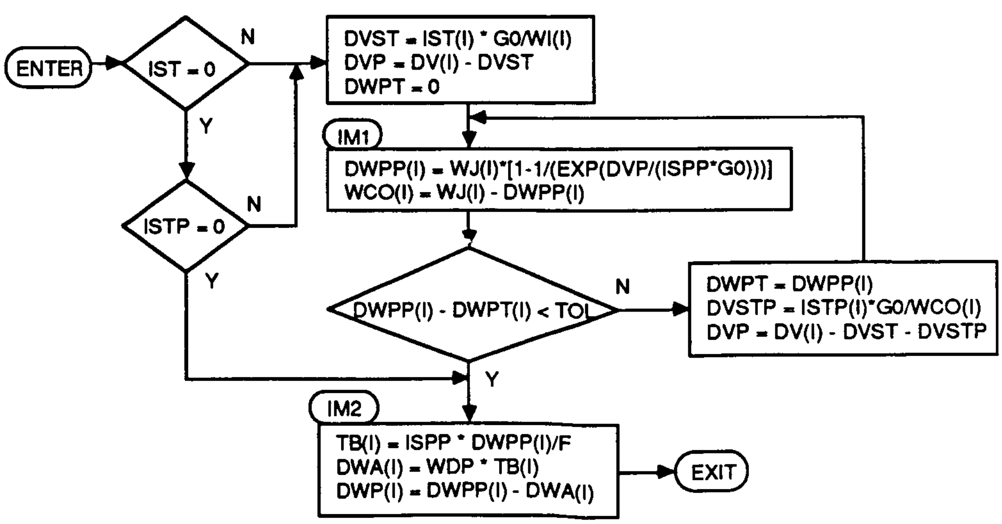
SUBROUTINE ITER



SUBROUTINE ITERP



SUBROUTINE IMPULSE

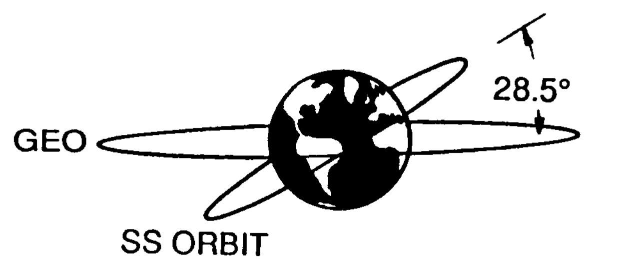


Variable	Description
AST()	RCS Start Loss
ASTP()	RCS Stop Loss
BJ	Which Burnout Iterated Jettison Weight will be Driven to
DAFPR	Auxiliary Propellant Used in Consuming FPR
DEPS	Delta Epsilon
DEPSP	Delta Epsilon
DV()	Delta Velocity
DVFPR	Delta Velocity Reserved for Dispersions (FPR)
DVP	Delta Velocity Supplied for Steady State Burn
DVST()	Delta Velocity During Engine Start Sequence
DVSTP()	Delta Velocity During Engine Shutdown Sequence
DWA()	Boost Pump Propellant used During Steady State Burn
DWACS()	Boost Pump Propellant used During Steady State Burn
DWFPR	Propellant Reserved for Dispersions
DWP()	Propellant Comsumed During Steady State Burn
DWPP()	Propellant (+RCS) Consumed During Steady State Burn
DWPT	Iterated Value of DWPP
DWPT()	Iterated Value of DWPP
DWPV()	Vented Propellant Weight
EPS	Error in Desired Value of Dependent Variable
EPS1	Error in Desired Value of Dependent Variable
EPS2	Error in Desired Value of Dependent Variable
EPSP	Iteration Error in Jettison Weight Versus Payload
EPSP1	Error in Desired Value of Dependent Variable
EPSP2	Error in Desired Value of Dependent Variable
F	Force or Thrust of Engines
FP$	Flag for Fixed Payload Weight
FPR	Flight Performance Reserve Propellant Ratio (% Δ Vt)
GO	Gravity at Sea Level 32.174 Ft/Sec^2 9.81 M/Sec^2
I	Iteration Counter (Variable Gross Weight)
IP	Iteration Counter (Fixed Gross Weight)
ISP	Specific Impulse
ISPP	Specific Impulse Corrected for Boost Pump Flow
IST()	Engine Start Impulse
ISTP	Engine Stop Impulse
ISTP()	Engine Stop Impulse
K	Flag for Iteration 1 = Continue 2 = Tol met 3 = Too Many Iter
KA	Flag for Iteration 1 = Continue 2 = Tol met 3 = Too Many Iter
KP	Flag for Iteration 1 = Continue 2 = Tol met 3 = Too Many Iter
M	Number of Engine Burns
PCV()	Percent of Remaining Propellant to be Vented
PL	Payload Weight Input
RHO	Value of Dependent Variable
RHO2	Value of Dependent Variable
RHOP	Payload Estimate for Jettison Weight Iteration
RHOP1	Payload Estimate for Jettison Weight Iteration
RHOP2	Payload Estimate for Jettison Weight Iteration
SUMA	Summation of Auxiliary Propellant Loss

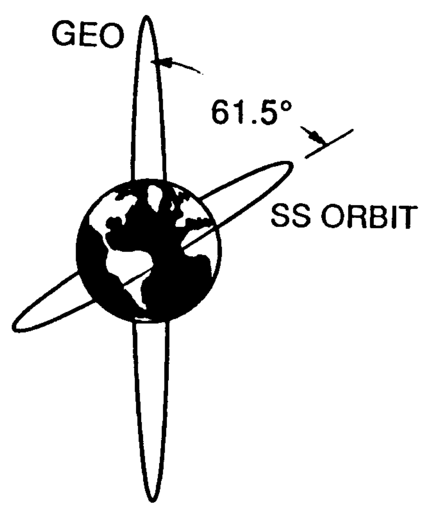
Variable	Description
SUMAB	Summation of Auxiliary Propellant Loss
SUMDV	Summation of Delta Velocities
SUMWJN	Summation of Weights Jettisoned During Coasts
SUMWP	Summation of Main Propellant Weights
TB()	Burn Time
TBFPR	Time Required to Consume FPR Propellants
TOL	Tolerance in Dependent Variable
WACS	Weight of Auxiliary Propellant
WB0	Burnout Weight
WDP	Boost Pump Flow Rate
WDTP	Boost Pump Flow Rate
WGM	Maximum Allowable Gross Weight
WI()	Weight at Start of I th Burn
WJ	Jettison Weight - Minimum Remaining Propellant
WJ()	Jettison Weight - Minimum Remaining Propellant
WJIN	Iterated Value of Jettison Weight During Coast
WJN()	Jettisoned Weight at end of I th Burn
WJP	Jettison Weight - Current Value
WJTI	Initial Value of SUMWJN
WP	Expendable Propellant Weight - Current Value
WPO	Initial Value of Expendable Propellant
WPR	Expendable Propellant Weight - Reference Value
WST()	Propellant Start Loss - I th Burn
WSTP()	Propellant Stop Loss - I th Burn

APPENDIX B
SBTC PERFORMANCE ANALYSIS RESULTS

SINGLE PAYLOAD MAXIMUM WEIGHT TO A GIVEN ALTITUDE FOR SBTC DEPENDS ON PLANE CHANGE (P/C) REQUIRED.



SBTC CAN PUT 9273 KGS (20,400 LBS) INTO A GEO ALTITUDE, 0° INCL ORBIT.



SBTC CAN PUT 7273 KGS (16,000) LBS INTO A GEO ALTITUDE, 90° INCL ORBIT.

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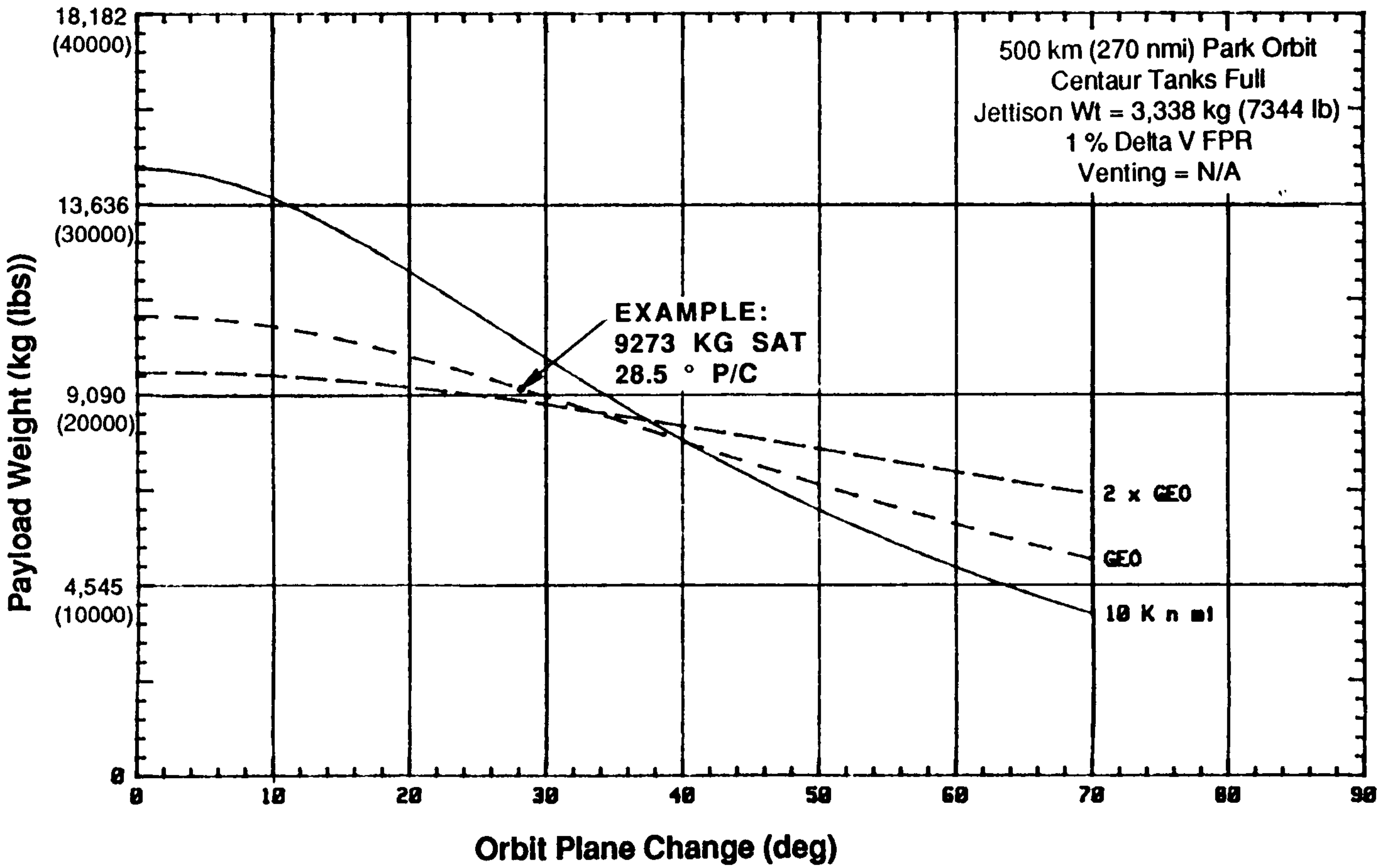
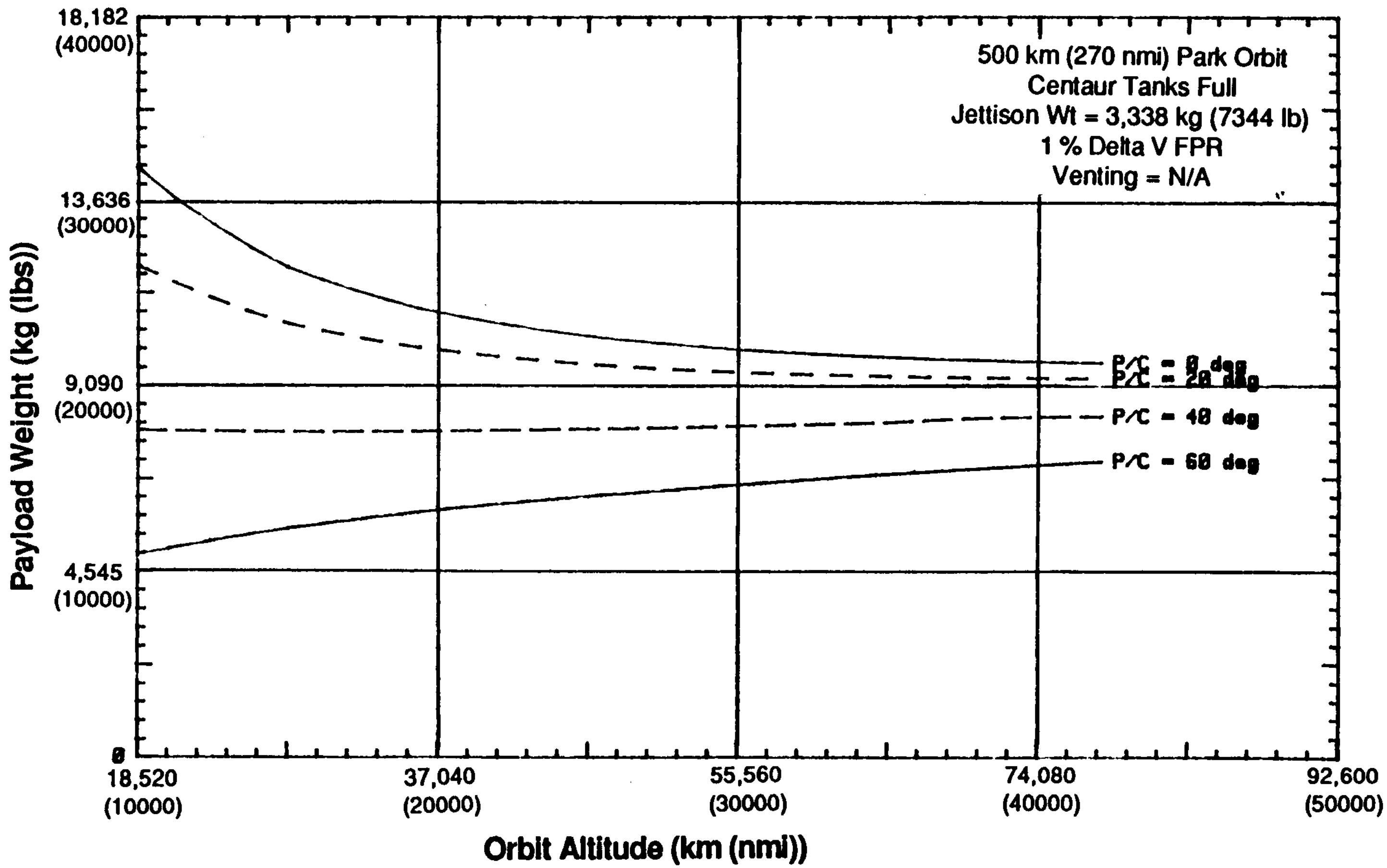
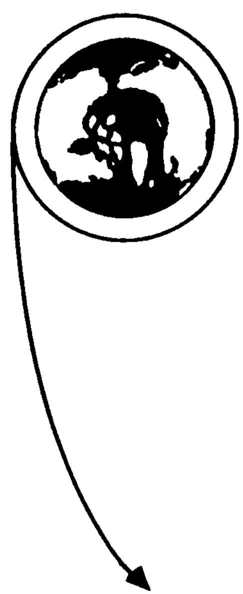
THE SBTC WILL HAVE A ROBUST PAYLOAD DELIVERY CAPABILITY FOR DIFFERENT PLANE CHANGE ANGLES



THE SBTC WILL HAVE A ROBUST PAYLOAD DELIVERY CAPABILITY TO DIFFERENT ORBIT ALTITUDES

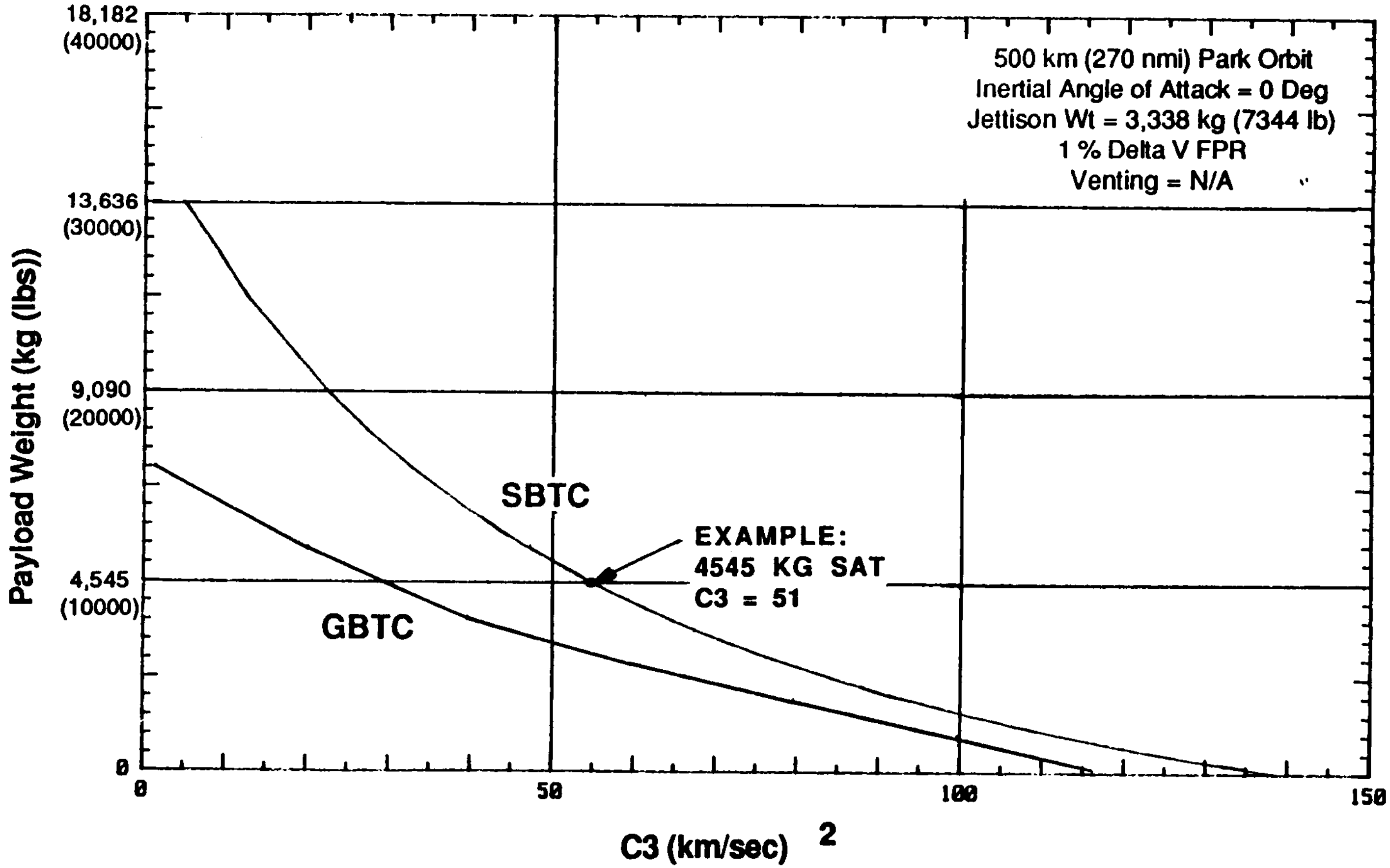


CENTAUR WILL HAVE THE CAPABILITY TO SEND LARGE SPACECRAFT TO INTERPLANETARY DESTINATIONS

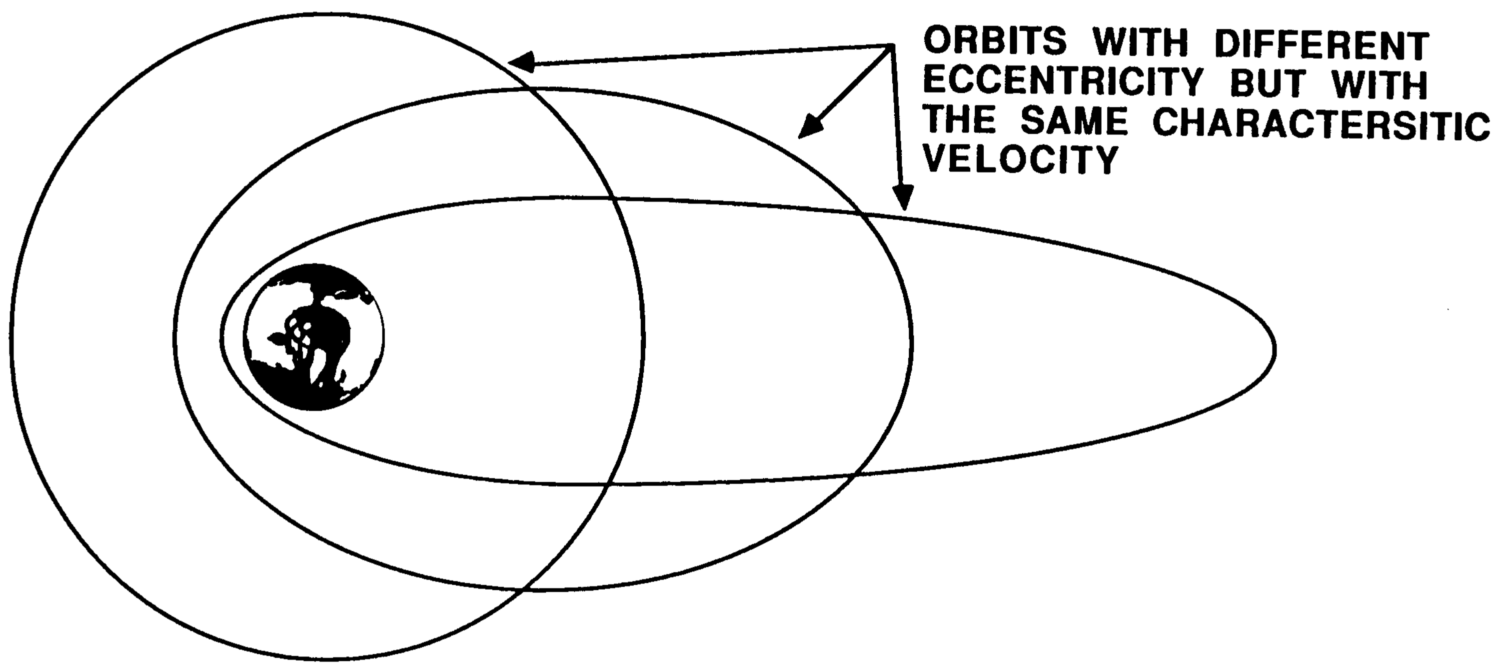


FOR EXAMPLE:
A 4545 KG SPACECRAFT COULD BE LAUNCHED ON AN INTERPLANETARY TRAJECTORY WITH A C3 OF 51 (KM/SEC)**2

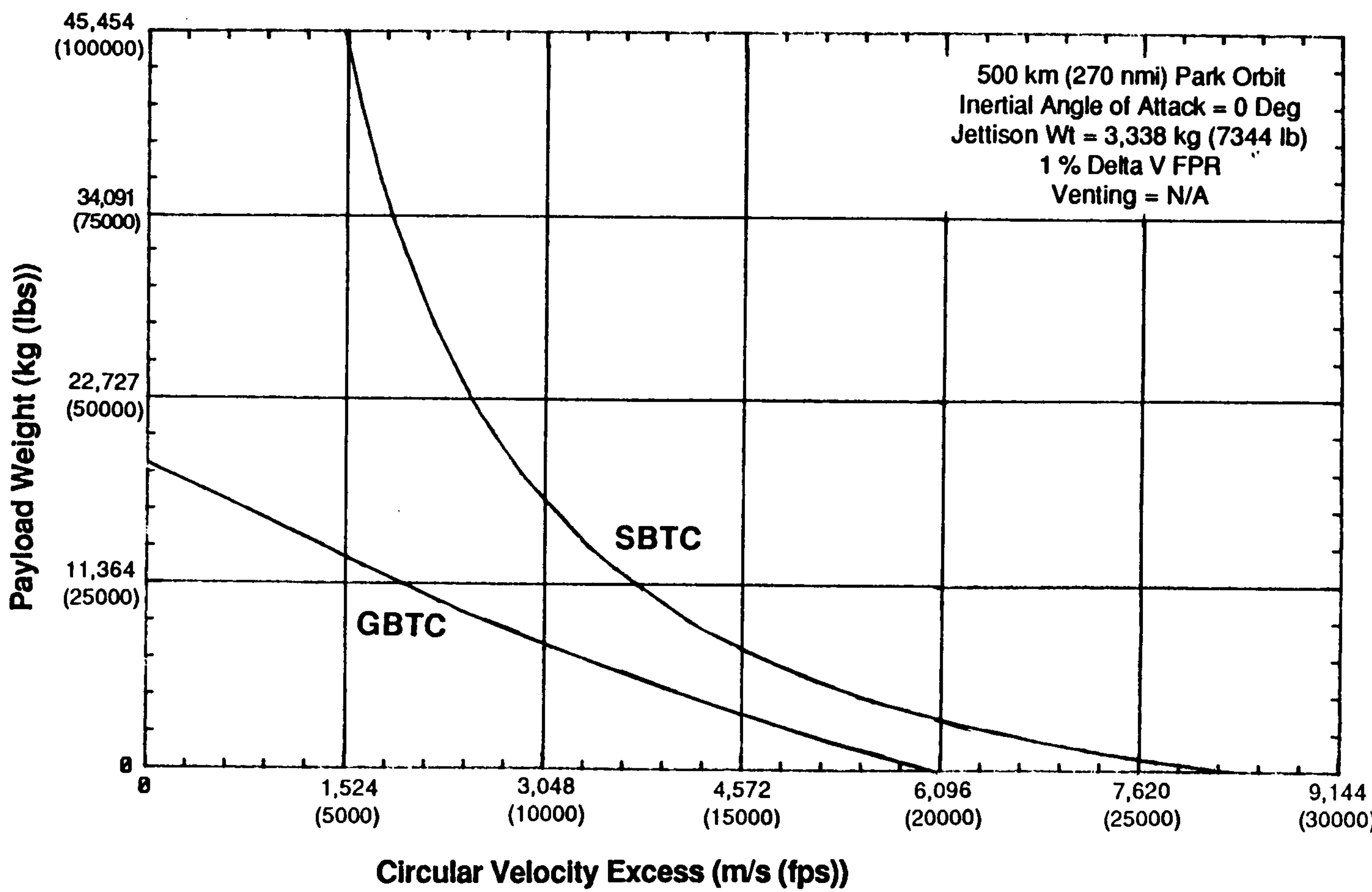
CENTAUR PLANETARY MISSION CAPABILITY INCREASES FROM THE SPACE STATION



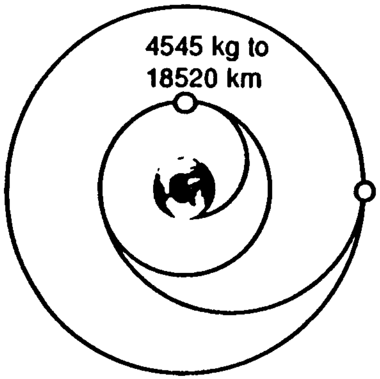
CENTAUR WILL BE CAPABLE OF PROVIDING A LARGE VARIETY OF ORBITS FOR A GIVEN SATELLITE WEIGHT.



CENTAUR PAYLOAD WEIGHT VS. CHARACTERISTIC VELOCITY FOR CLOSED ORBITS INCREASES AT THE STATION

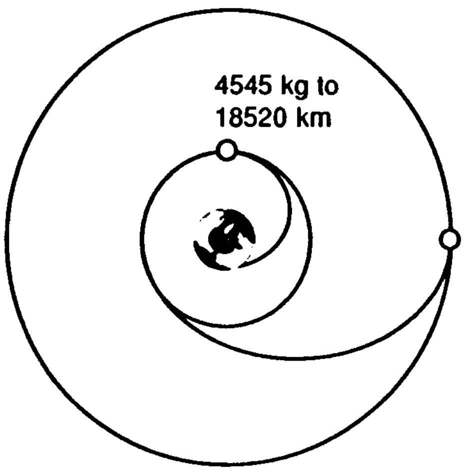


SBTC CAN DEPLOY TWO SPACECRAFT WHICH HAVE DIFFERENT ALTITUDE DELIVERY REQUIREMENTS.



- THE WEIGHT OF THE FIRST SATELLITE DETERMINES THE CAPABILITY FOR THE SECOND FOR A GIVEN DELIVERY ALTITUDE DIFFERENCE.

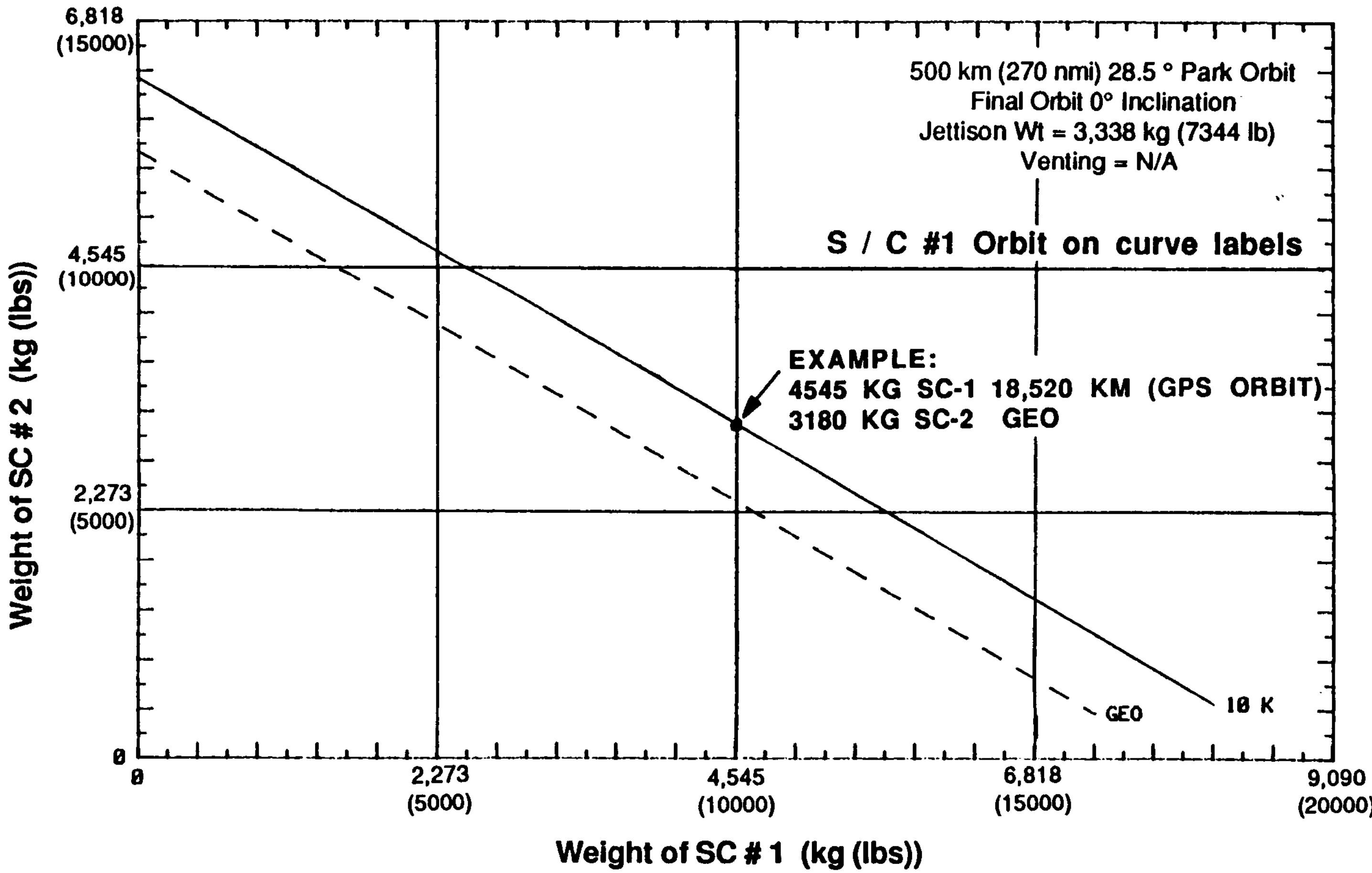
EXAMPLE 1: IF A 4545 KG S/C IS CIRCULARIZED AT 18,520 KM (GPS ORBIT), A 3,180 KG S/C CAN BE CIRCULARIZED AT GEO.



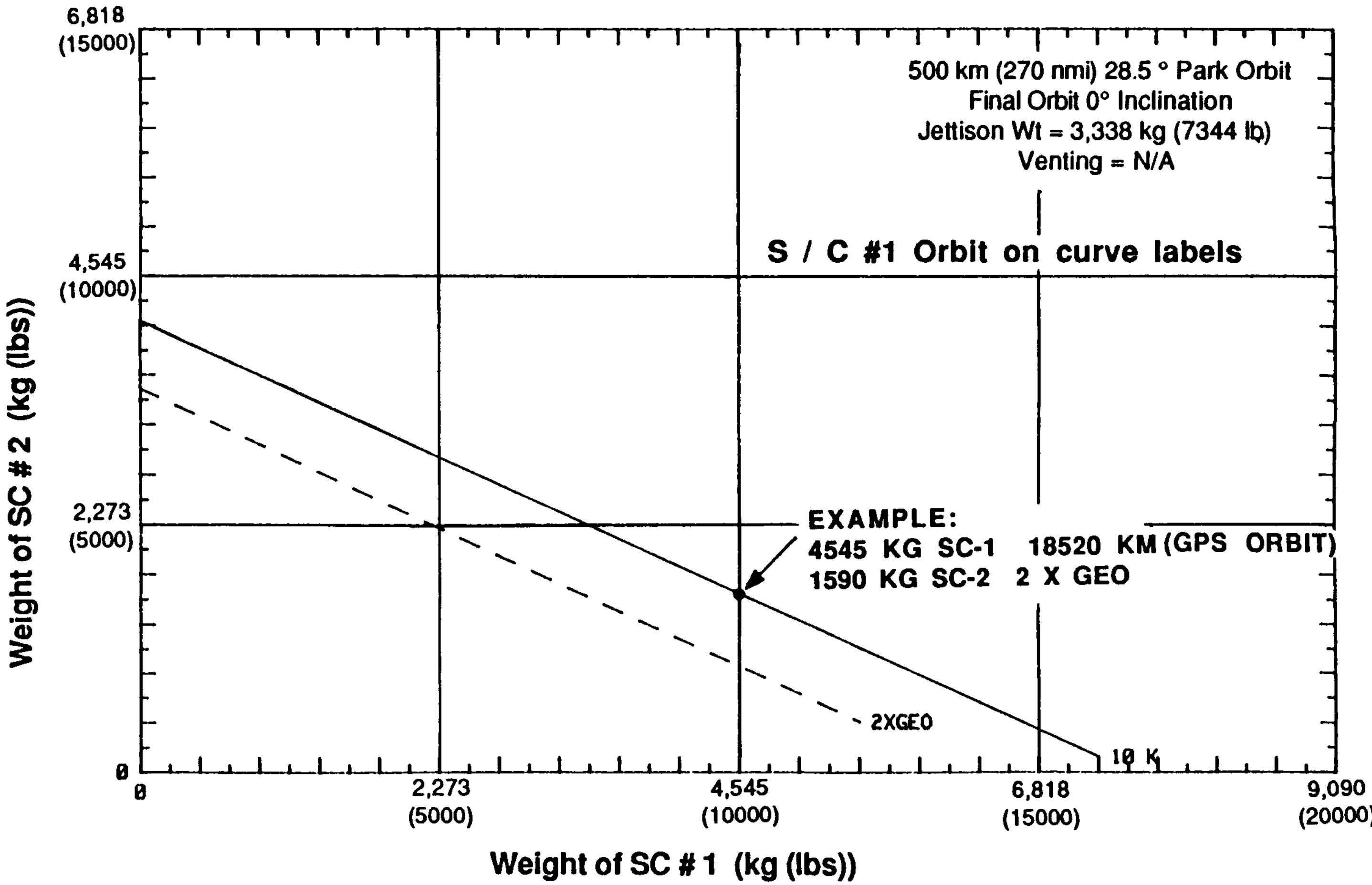
- ALTITUDE SEPARATION BETWEEN THE S/C AFFECTS TOTAL PAYLOAD CAPABILITY.

EXAMPLE 2: IF 4545KG S/C IS CIRCULARIZED AT 18,520 KM, A 1590 KG S/C CAN BE CIRC AT 2 X GEO.

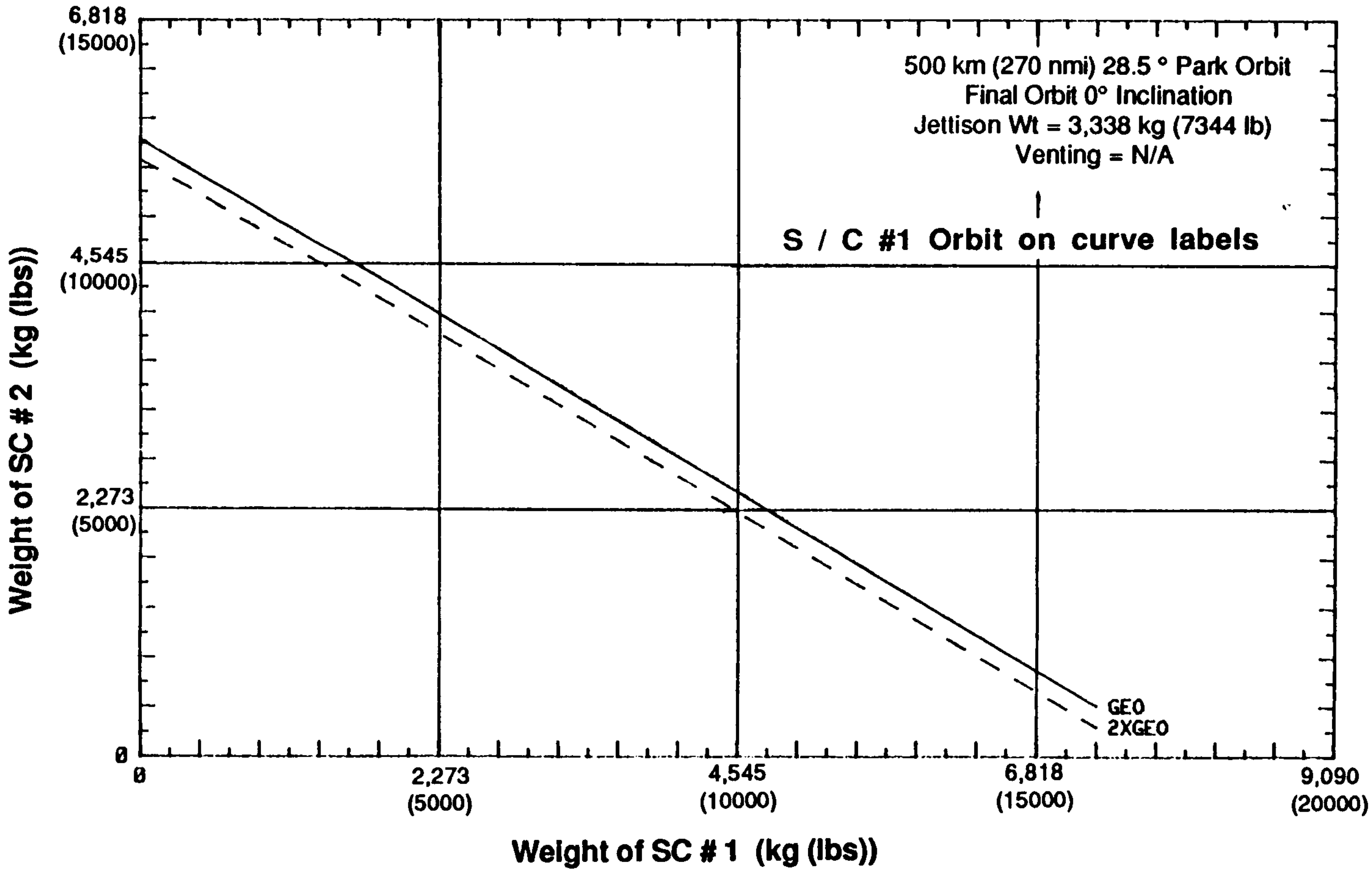
SBTC CAN DELIVER ONE SPACECRAFT TO 18520 KM AND ANOTHER TO GEO



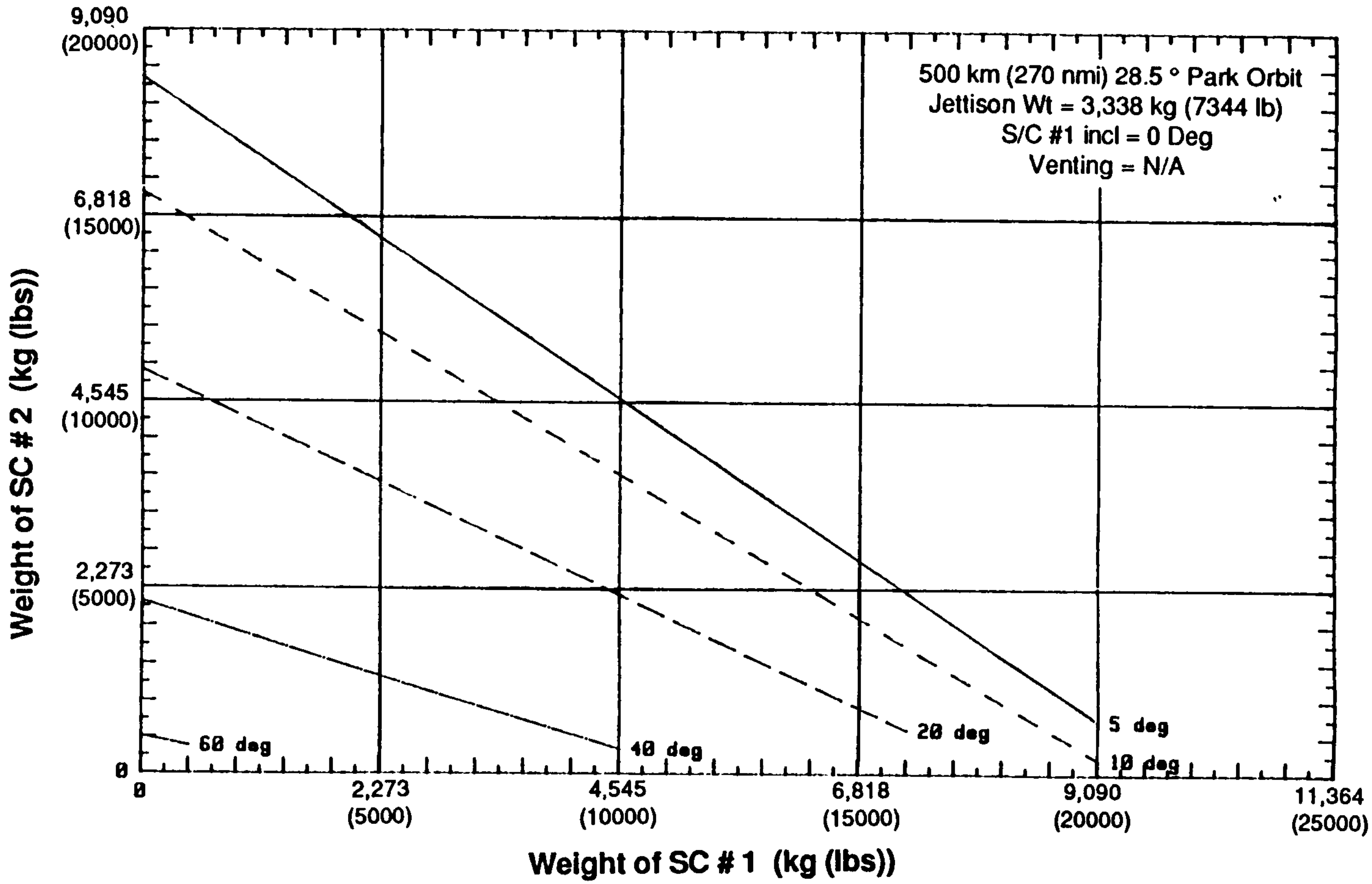
SBTC CAN DELIVER ONE SPACECRAFT TO 18520 KM AND ANOTHER TO 2 X GEO



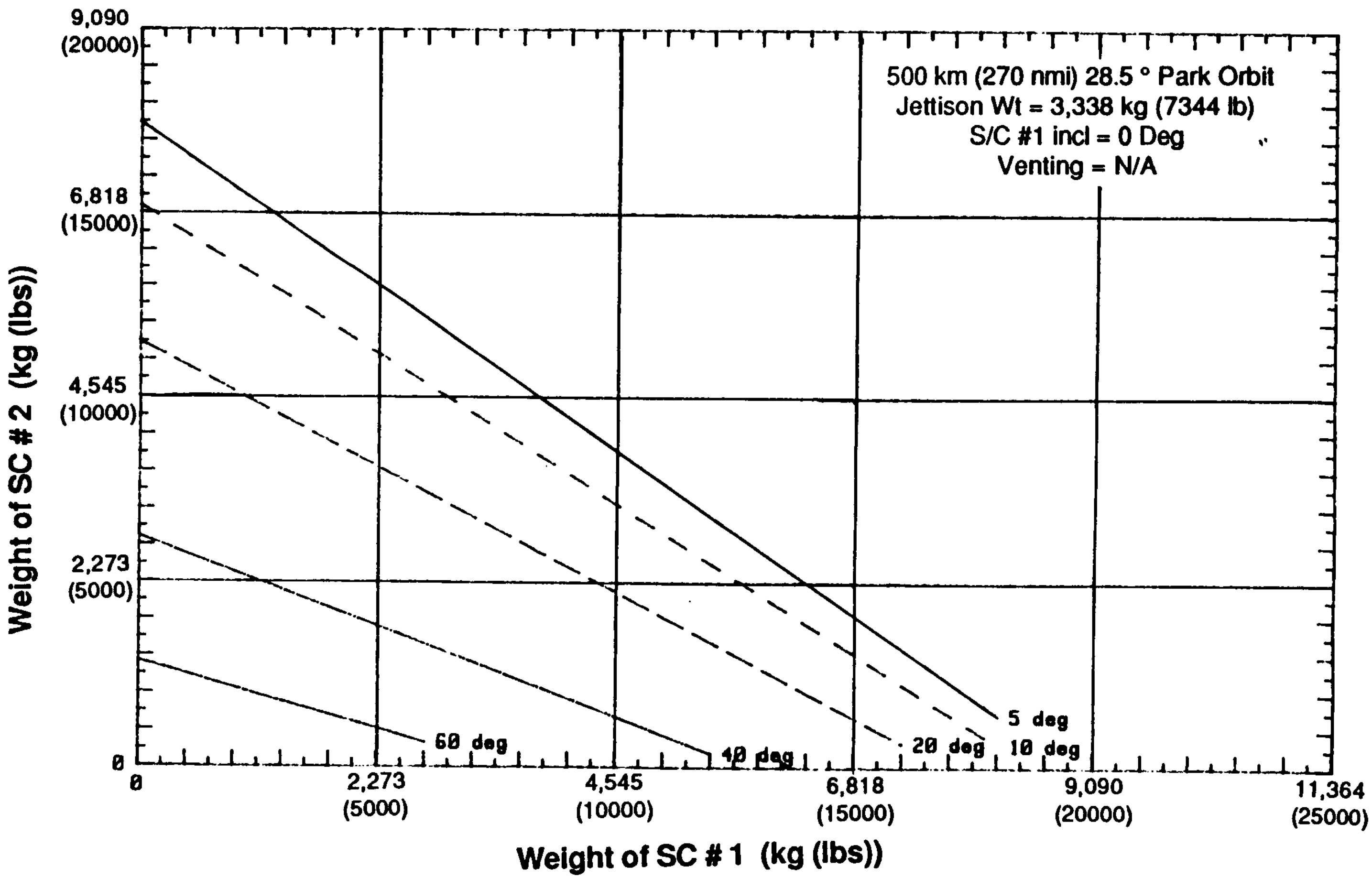
SBTC CAN DELIVER ONE SPACECRAFT TO GEO AND ANOTHER TO 2 X GEO



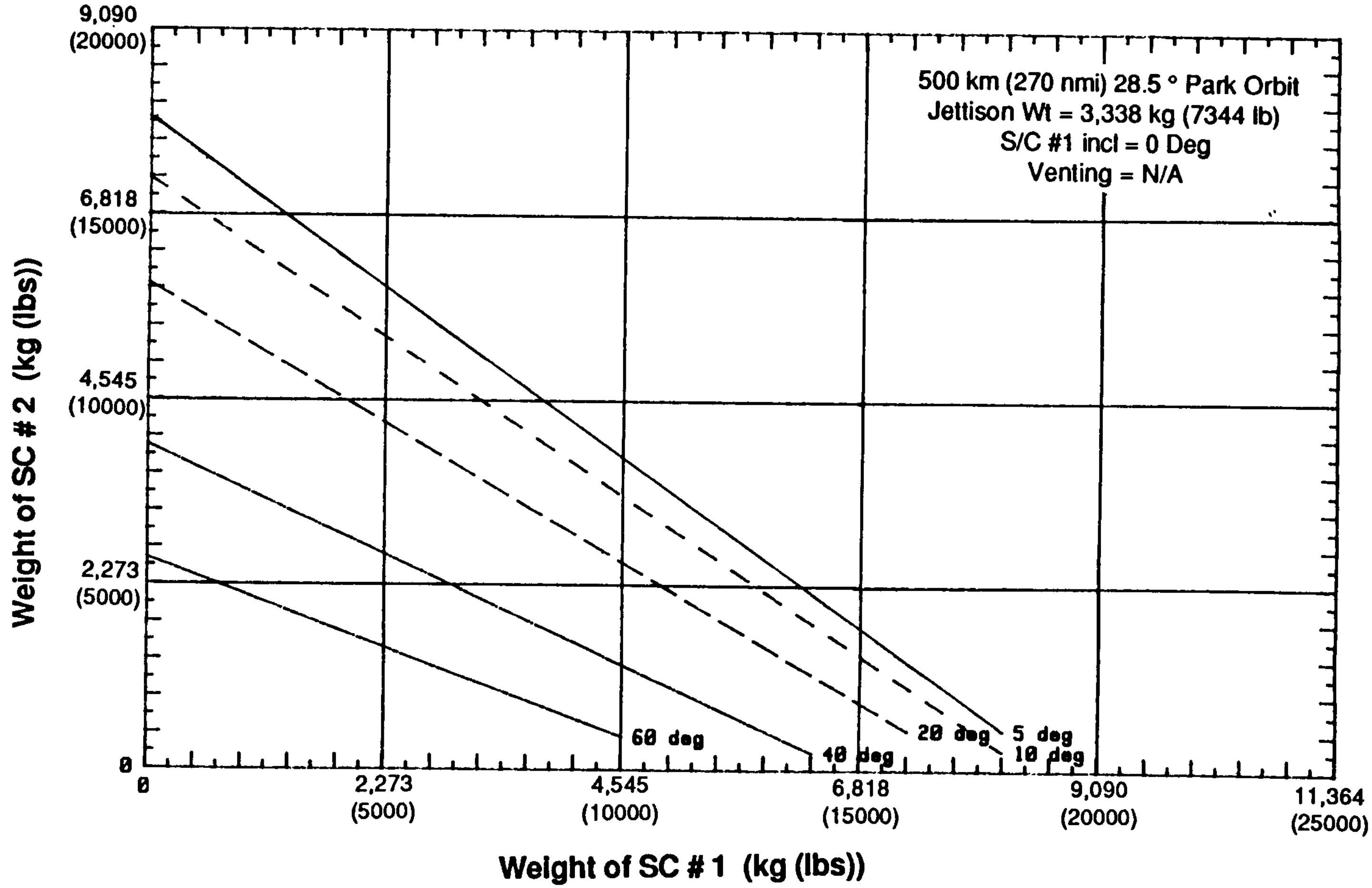
SBTC CAN DELIVER TWO SPACECRAFT TO 18,520 KM AT DIFFERENT INCLINATION ANGLES



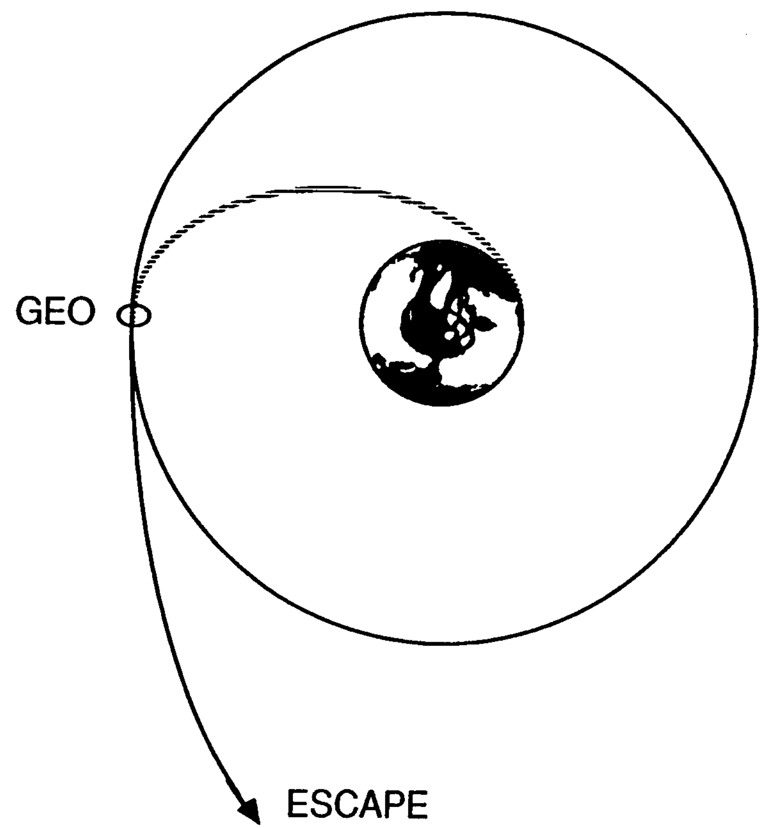
SBTC CAN DELIVER TWO SPACECRAFT TO GEO AT DIFFERENT INCLINATION ANGLES



SBTC CAN DELIVER TWO SPACECRAFT TO 2 X, GEO AT DIFFERENT INCLINATION ANGLES



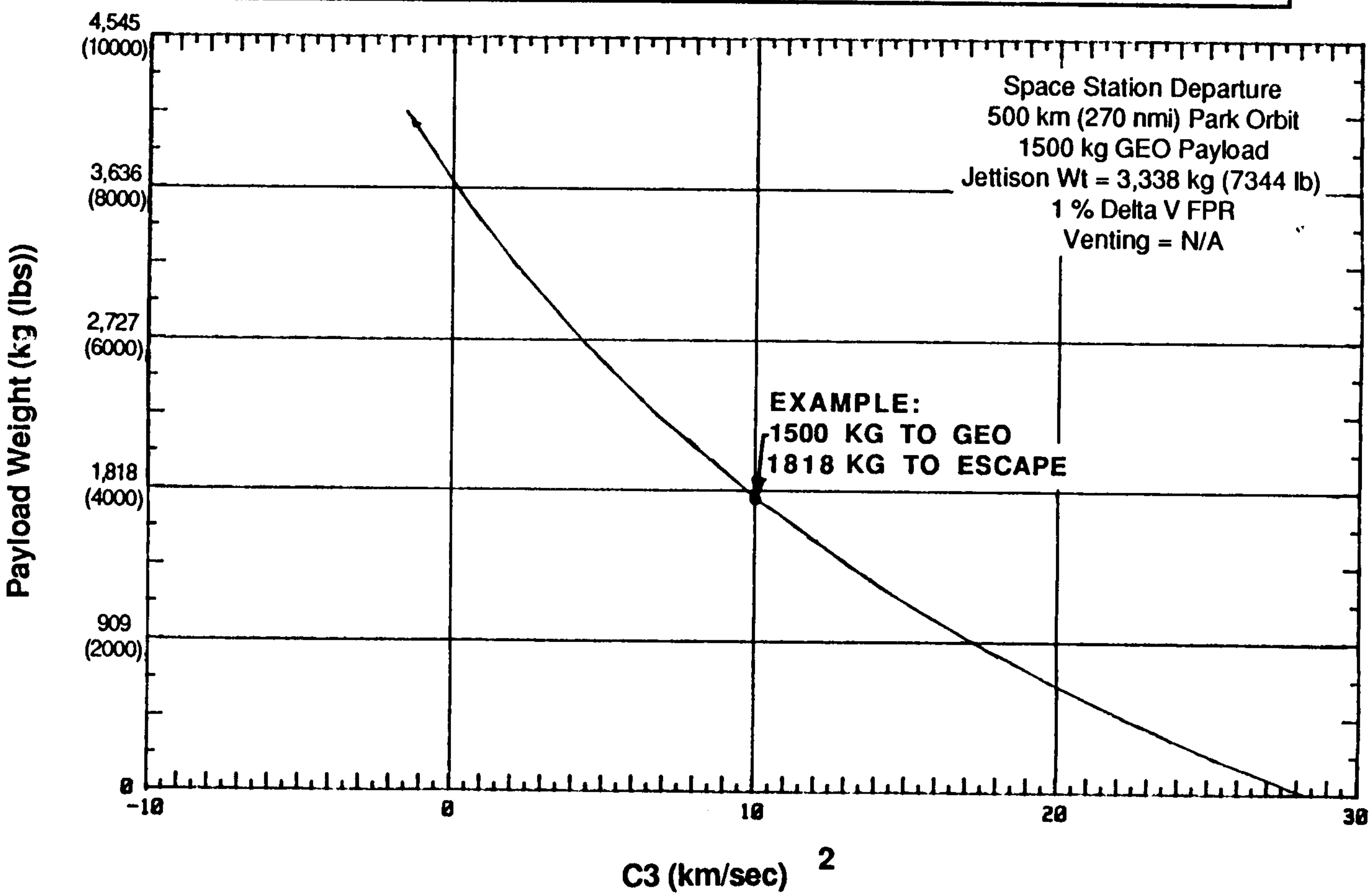
CENTAUR PERFORMANCE FROM THE SPACE STATION GEO PLUS ESCAPE DELIVERY MISSION



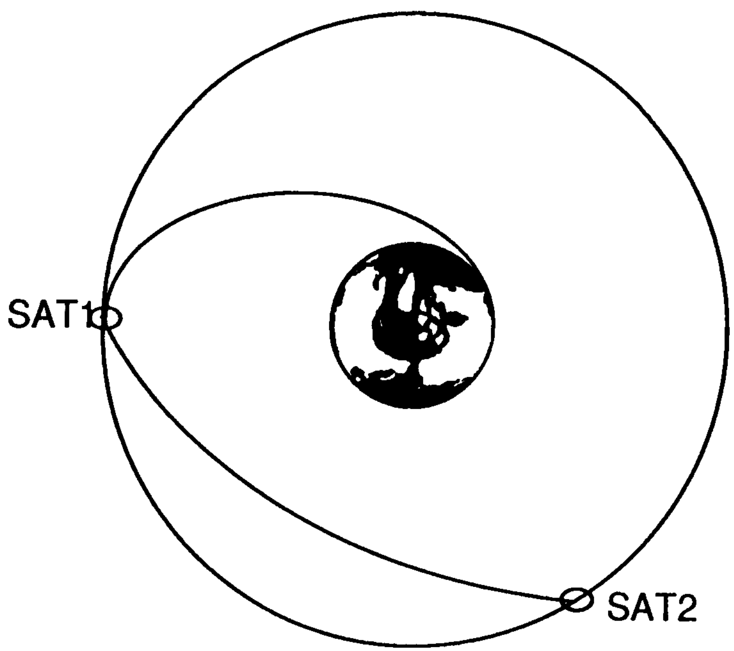
THE CENTAUR WILL HAVE THE CAPABILITY TO PLACE A SATELLITE INTO GEOSYNCHRONOUS ORBIT AND STILL HAVE ENOUGH PERFORMANCE TO PERFORM AN ESCAPE MISSION.

FOR EXAMPLE, THE CENTAUR WOULD LAUNCH FROM THE COP, CIRCULARIZE AT 0° INCLINATION GEO ORBIT AND DEPLOY A 1500 KG SPACECRAFT. IT WOULD THEN PERFORM AN EARTH ESCAPE BURN TO PROPEL A 1,818 KG SATELLITE AT A C3 OF +10.0.

SBTC COULD PERFORM A PLANETARY MISSION EVEN AFTER DELIVERING A 1500 KG PAYLOAD TO GEO



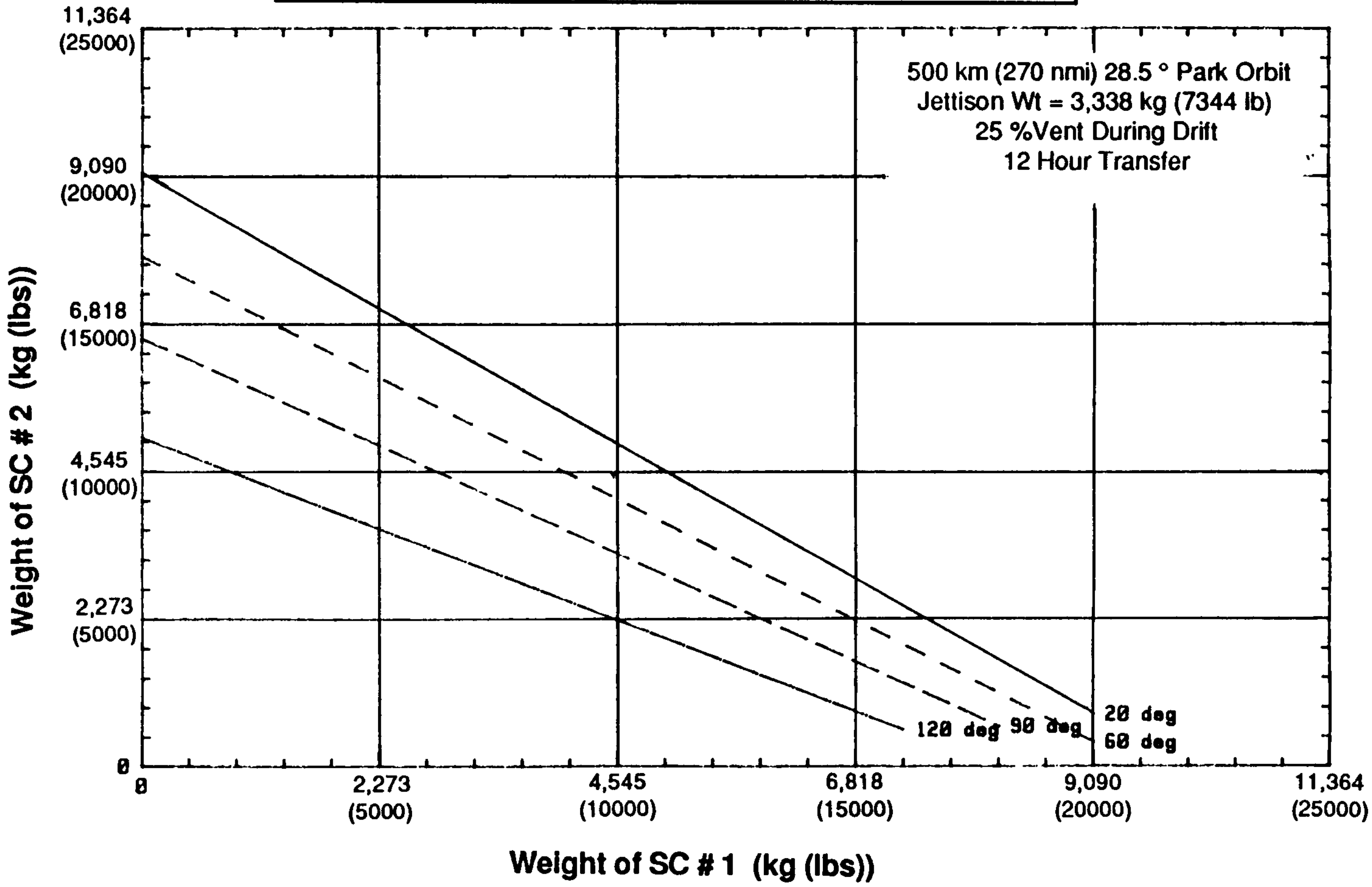
CENTAUR CAN DELIVER TWO COM-SATS TO GEO AND PROVIDE SPACING



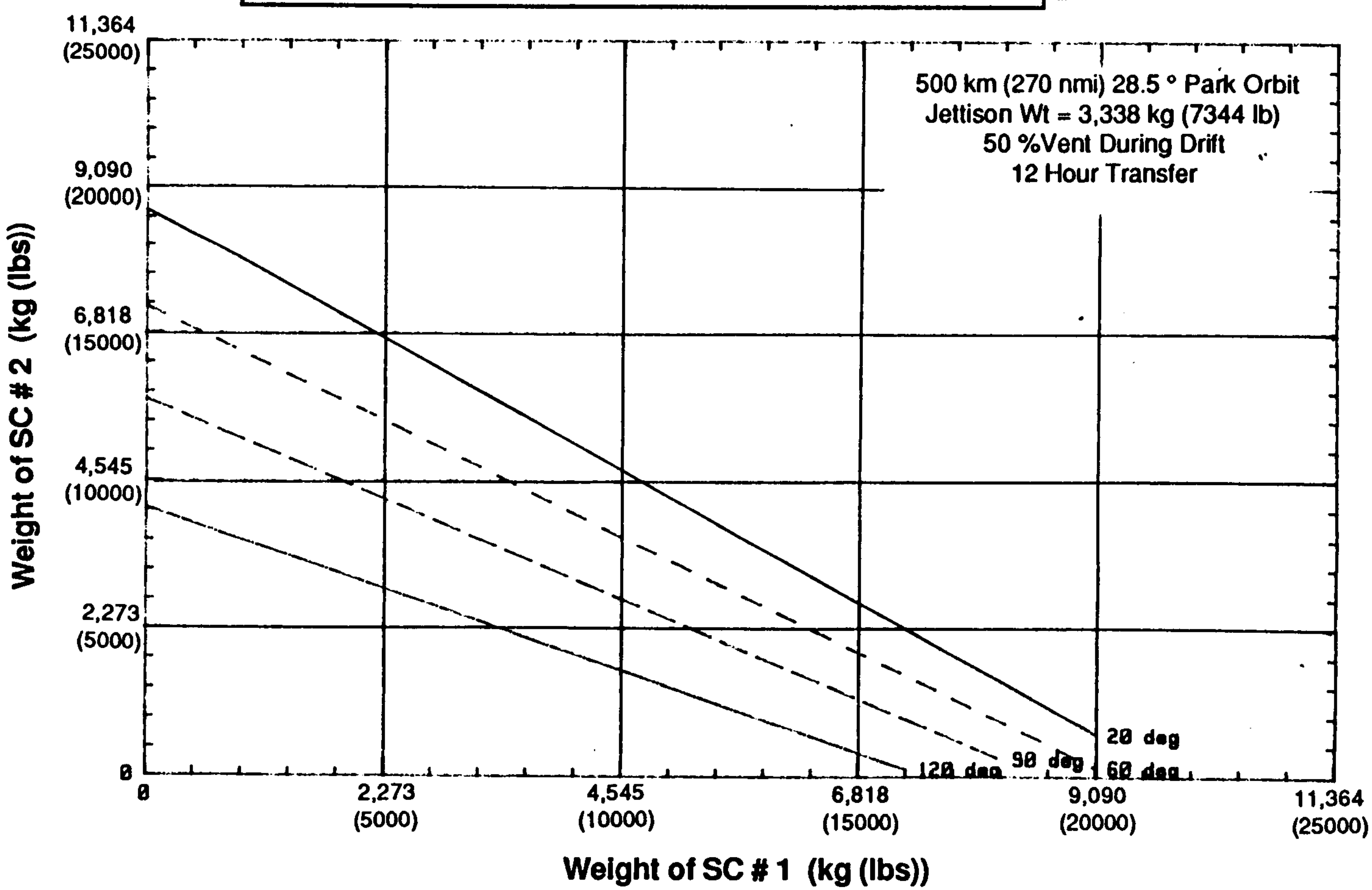
THE CENTAUR WILL HAVE THE CAPABILITY TO PLACE TWO SATELLITES INTO GEOSYNC ORBIT AT DIFFERENT PHASE ANGLES.

FOR EXAMPLE, THE CENTAUR COULD LAUNCH FROM THE COP, CIRCULARIZE AT 0° INCLINATION GEO ORBIT AND DEPLOY A 1818 KG SPACECRAFT. IT COULD THEN PERFORM A NON-HOHMAN TRANSFER TO PLACE A 2410 KG SATELLITE PHASED 120° AWAY IN 12 HRS WITH 25% PROP BOILOFF.

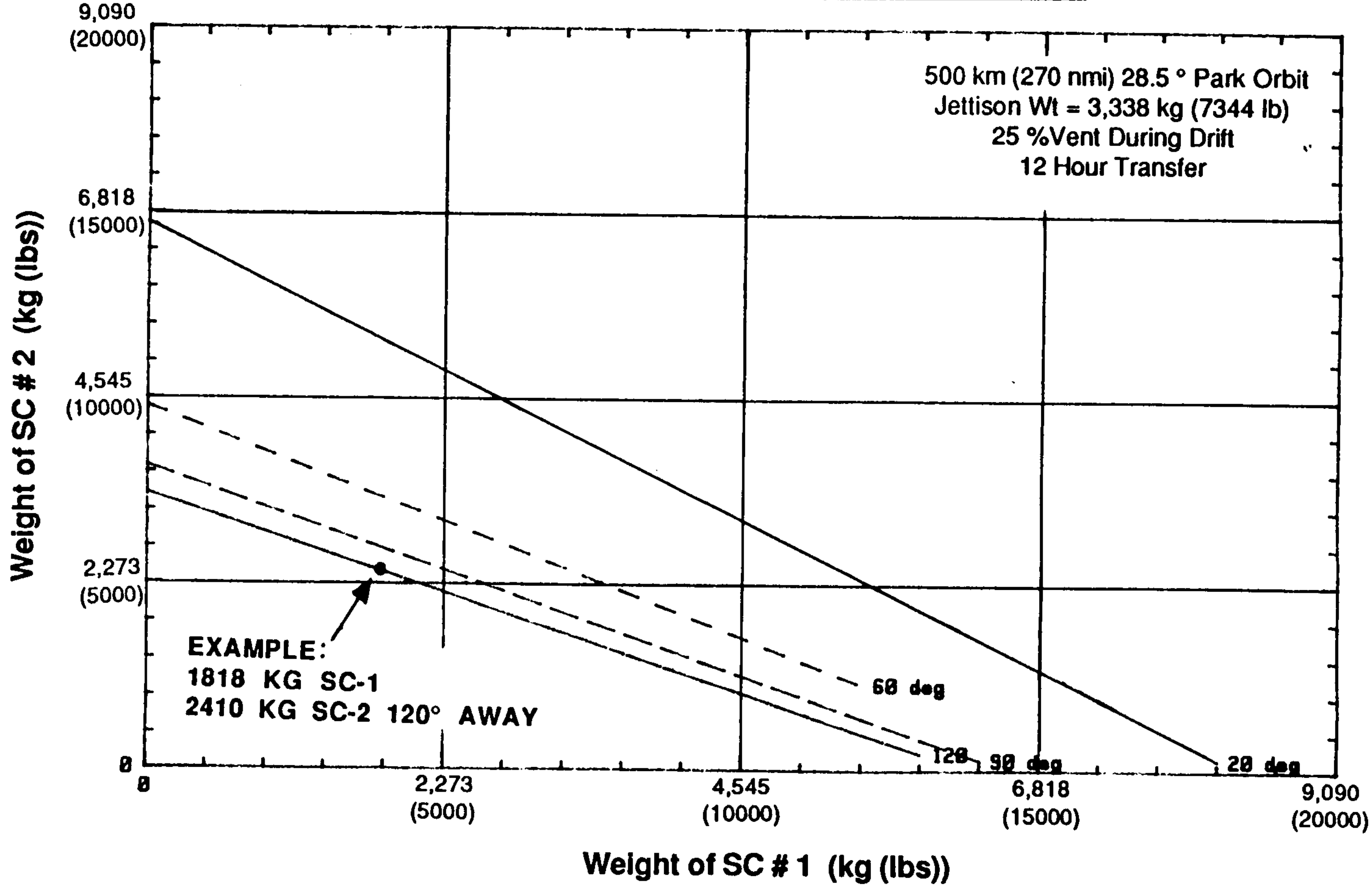
AT A 25% VENT RATE, SBTC CAN PLACE TWO SATELLITES IN 18520 KM ORBIT.



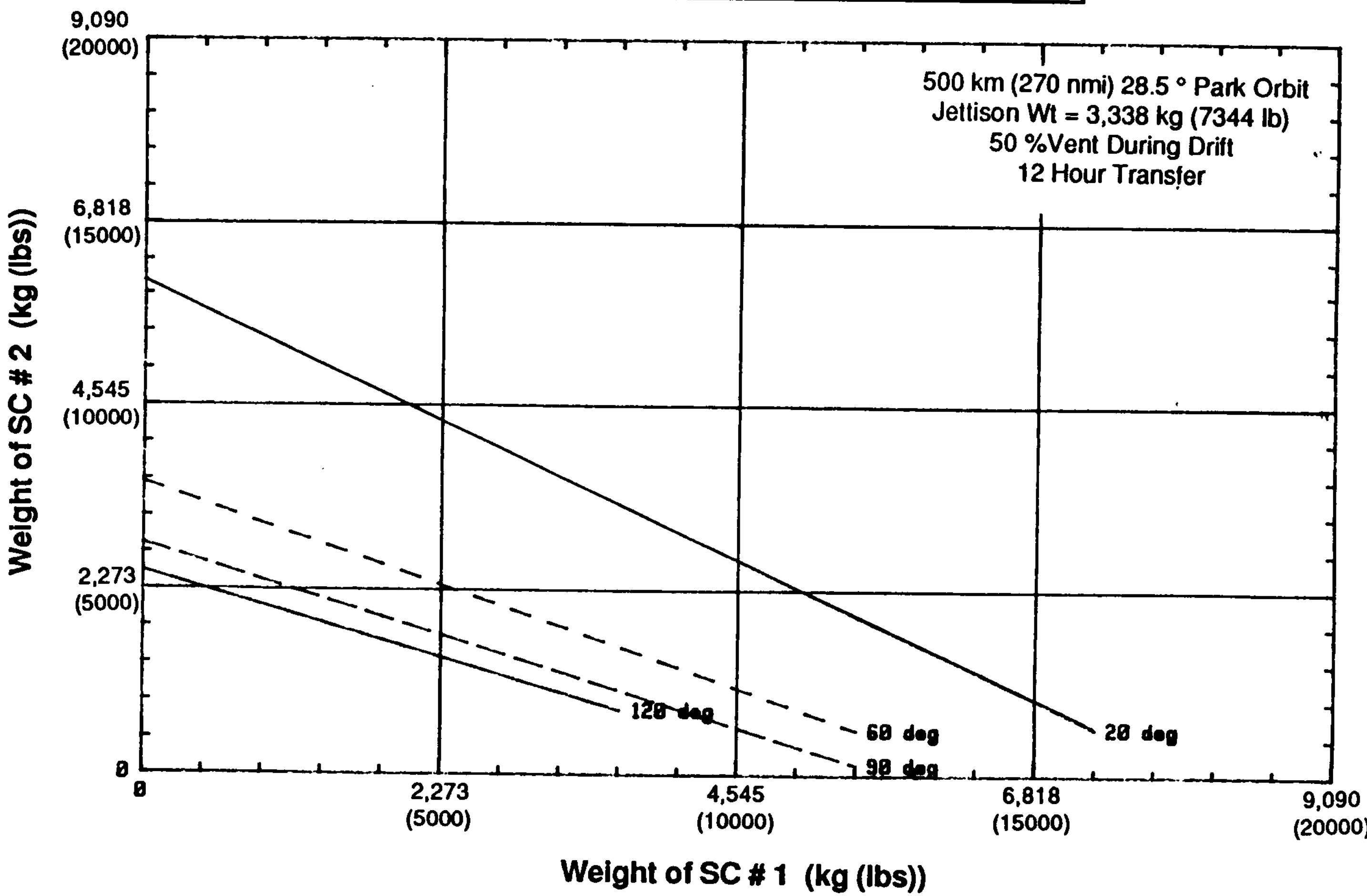
AT A 50% VENT RATE, SBTC CAN PLACE TWO SATELLITES IN 18520 KM ORBIT.



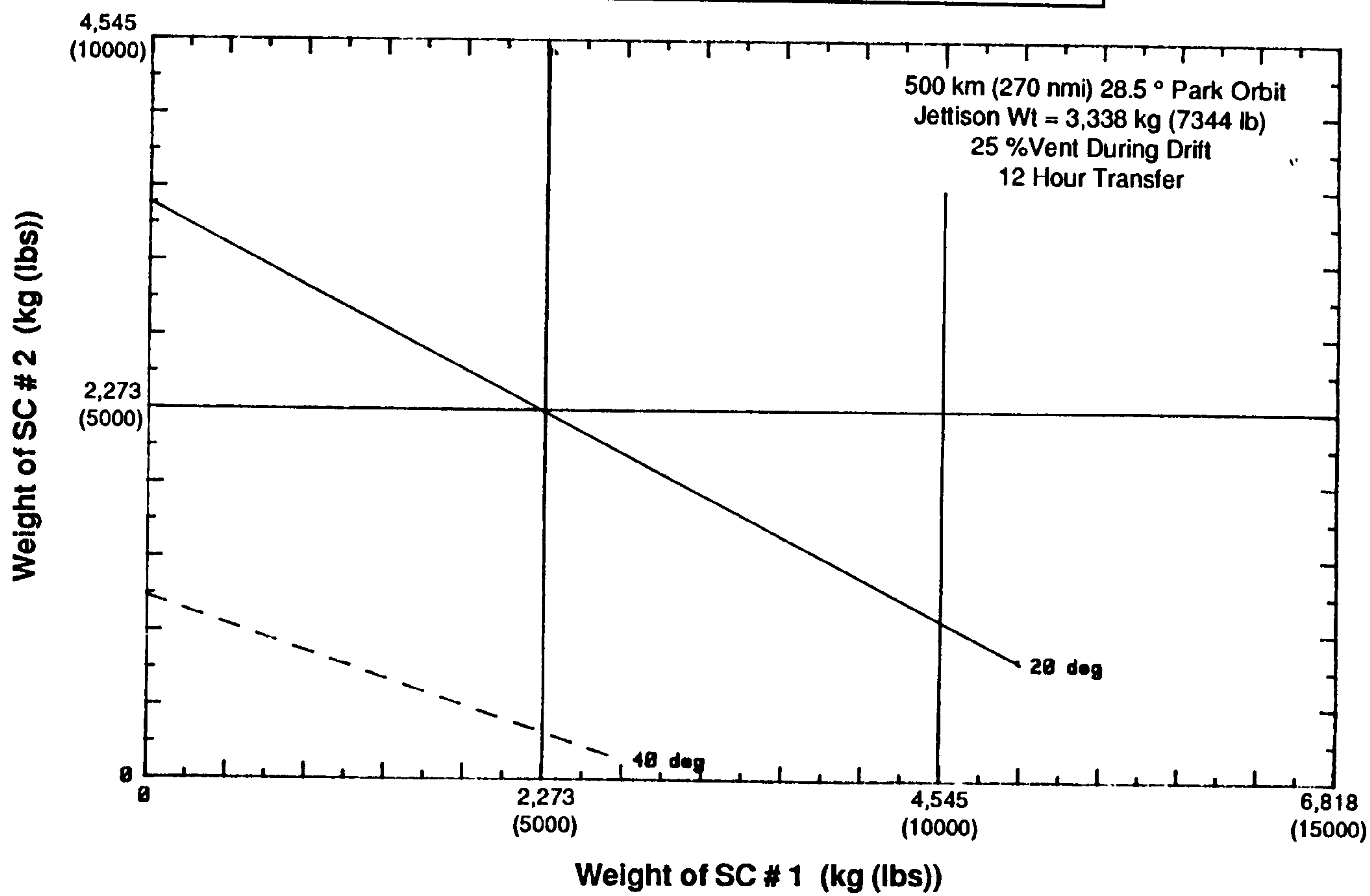
AT A 25% VENT RATE, SBTC CAN PLACE TWO SATELLITES IN GEO ORBIT.



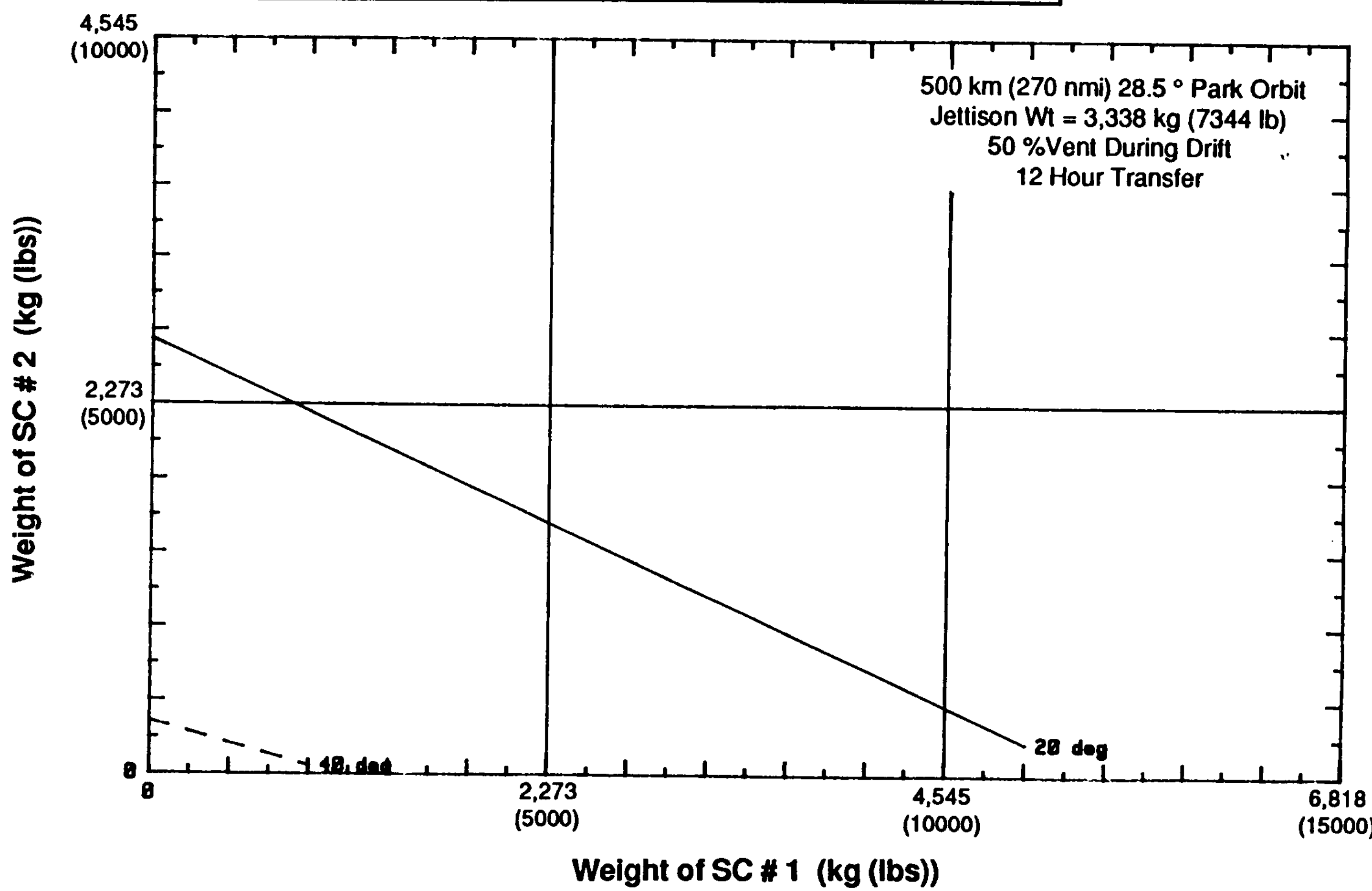
AT A 50% VENT RATE, SBTC CAN PLACE TWO SATELLITES IN GEO ORBIT.



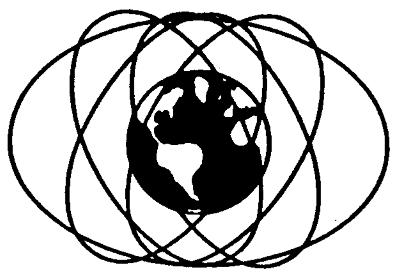
AT A 25% VENT RATE, SBTC CAN PLACE TWO SATELLITES IN 2 x GEO ORBIT.



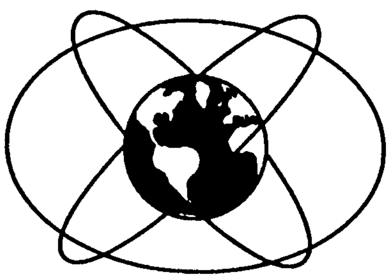
AT A 50% VENT RATE, SBTC CAN PLACE TWO SATELLITES IN 2 x GEO ORBIT.



SBTC COULD BENEFIT EITHER CURRENT OR PROPOSED GPS CONFIGURATIONS



THE PRESENT GPS CONFIGURATION CALLS FOR SIX PLANES OF SATELLITES WITH THREE SATELLITES SPACED 120° APART. THE SBTC COULD SUPPLY ALL 3 SATELLITES FOR ONE ORBIT (WITH SPACING) WITH A SINGLE LAUNCH.



A PROPOSED IMPROVED CONFIGURATION CONSISTS OF 3 ORBIT PLANES 120° APART WITH 6 SATELLITES AT 60° INTERVALS ON EACH. THE SBTC COULD SUPPLY UP TO 5 OF THE SATELLITES FOR ONE PLANE (WITH SPACING) WITH A SINGLE LAUNCH.

APPENDIX C
COSS SPACE STATION
TDM ANIMATION SEQUENCE LISTING

CSOD ANIMATION SEQUENCE

Commercial Space Operation Development

Space Flight Operations Animation

October 1987

Prepared for
NASA/Lewis Research Center
Cleveland, Ohio

Prepared by
GENERAL DYNAMICS
Space Systems Division
San Diego, CA

Contract NAS3-24900

CSOD PROGRAM GOALS

- Demonstrate Centaur launch of COM-SATs from Space Station
- Demonstrate/develop OTV accommodations & operations technology at Space Station
- Determine value of CSOD to NASA

Flight Operations Animation

- Illustrate operations for COM-SAT launch by Station-based Centaur
- Timeframe begins with growth Station incorporating Satellite Processing Facility
- Animation begins with the unfueled Centaur in the Orbiter

SEQUENCE OF EVENTS TO FOLLOW:

1. Shuttle docks with Station.
2. Centaur/CISS Assembly (CCA) removed from Cargo Bay.
3. CCA positioned in proximity of Centaur Hangar.
4. Tele-robotic Arm (TRA) mates CCA to hangar interface panel.
5. Centaur Support Structure rotates into position.

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1. **Shuttle docks with Station. ETE: start**

Two split screen images:

View 1 - Camera position should rotate around Space Station and Orbiter (like one of the views you had on the tape)

View 2 - Camera Position fixed with respect to station backed away some distance in an isometric view (similar to the isometric views you showed on the tape, but perhaps slightly farther away from the station so as to show the Orbiter's motions.

Action should show the Orbiter already oriented in a vertical position. As the Cargo Bay Doors open, the Orbiter should approach and dock with the Space Station.

2. **Centaur/CISS Assembly (CCA) removed from the Cargo Bay. ETE: 4 hrs**

Camera should be positioned in the isometric view as you show on the tape for this sequence, except that the camera should be move slightly so as to be able to view the entire Cargo Bay and Shuttle and less of the Satellite Processing Facility.

Action shown on the tape is fine.

LABELS: Centaur/CISS Assembly (CCA), RMS arm, Station RMS arm, Centaur Hangar

3. **CCA positioned in proximity of Centaur Hangar. ETE: 5 hrs**

Camera should be positioned in exactly the same location as was in sequence #2 (make sure the entire motion of the MRMS is within the view).

Action shown on the tape is OK, but arm motion needs to be slowed down.

LABELS: none required

4. **Tele-robotic Arm (TRA) mates CCA to hangar interface panel. ETE: 6 hrs**

Camera should be in the same position as in sequence #3.

Some of the motions of the arms seem too fast; slow these motions down. Make sure that the start of this sequence begins where tne previous sequence left off (positioning of the Centaur moved from sequence #3 to #4 in the tape). Also, don't move the MRMS arm away until the TRA has docked the CCA.

LABELS: Tele-robotic Arm (TRA), Hangar interface panel, Centaur Support Structure

5. Centaur Support Structure rotates into position. ETE: 12 hrs

Camera should be positioned in exactly the same location as was in sequence #4.

The TRA should be shown attached to the CCA before the rotation of the support structures begins. When these have moved into position, then the TRA should release the CCA and reposition itself to a stowed position.

LABELS none required.

REPOSITIONED CAMERA SEQUENCE

CAMERA POSITION: INSIDE CENTAUR HANGAR

SEQUENCE: CENTAUR POSITIONING IN HANGAR AND FINAL BERTHING

3a - 5a. Insert sequences #3, 4, and 5 but this time viewed with a camera position that starts at the lower corner of the Centaur hangar. This position will be kept during sequence 3a and at the start of 4a. It then pans the Centaur back to mating on the aft hangar wall, then translates towards the hangar front where sequence 5a is watched (this final camera position will be used in sequence 8a).

While the Centaur is berthed in its hangar, the accommodations TDMs will be performed:

- Vehicle checkout
- Vehicle maintenance
- Vehicle servicing
- Simulated payload integration

Operations TDMs will then be performed:

- Cryogenic propellant resupply
- Deployment from COP with payload

SEQUENCE OF EVENTS TO FOLLOW:

6. Station RMS moves satellite toward hangar.
7. TRA positions and mates satellite to Centaur.

6. Station RMS moves satellite toward hangar. ETE: 5 mon; 3 wks; 0 hrs

Camera should be positioned in exactly the same location as was in sequence #5, unless this field-of-view causes either the MRMS arm or the satellite to go off screen during the movements. If this occurs, reposition the camera back slightly.

When action starts, the satellite should be positioned parallel to and in line with the Satellite Processing Facility's center-line. The Orbiter should not be in any sequences from this point on. Also, slow down the motions of the satellite/arm combination.

LABELS Commercial Satellite (HS-393), Spacecraft Processing Facility (SPF)

7. TRA positions and mates satellite to Centaur. ETE: 5 mon; 3 wks; 1 hr

Camera should be positioned in exactly the same location as was in sequence #6.

Begin this sequence where the last sequence ended (with the satellite in the same position). No Orbiter should be shown.

LABELS none required.

REPOSITIONED CAMERA SEQUENCE

CAMERA POSITION: INSIDE CENTAUR HANGAR

SEQUENCE: SATELLITE POSITIONING AND MATING TO CENTAUR

6a, 7a. Insert sequences #6 and #7 but this time viewed with a camera position that starts from last position in 5a, then pans back towards the front of the hangar where it can watch the satellite handling done in sequences 6 and 7.

SEQUENCE OF EVENTS TO FOLLOW:

8. Interface panel disconnects from CCA, aft wall hinges open, and OMV is mated to CCA.
9. Centaur Support Structure rotates away from CCA.
10. TRA translates "stack" the out of hangar.

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8. **Interface panel disconnects from CCA, aft hangar wall hinges open, and OMV is positioned and mated to CCA. ETE: 5 mon; 3 wks; 72 hrs**

Camera should initially be positioned in exactly the same location as was in sequence #7, then, before the "action" starts, should zoom in on the Centaur Hangar until it and the attached satellite almost fill the screen.

Action should start with the rear hangar wall hinging open. When the door has fully opened, the OMV should be moved into the picture by the MRMS arm and mated to the aft end of the CCA. No Orbiter should be shown.

LABELS Aft hangar wall, OMV.

9. **Centaur Support Structure hinges away from CCA. ETE: 5 mon; 3 wks; 74 hrs**

Camera should be positioned in exactly the same location as it was at the end of sequence #8.

Start action with the TRA moving and attaching to the CCA. Then show the Centaur Support Structure rotating away from the CCA. No Orbiter should be shown.

LABELS none required.

10. **TRA translates "stack" out of hangar, then releases. ETE: 5 mon; 3 wks; 80 hrs**

Camera should initially be positioned in exactly the same location as was in sequence #9; then, before the "action" starts, should zoom out from the Centaur Hangar until the field-of-view will allow the entire "stack" and hangar to be viewed when the "stack has been translated out of the hangar.

Action should show the TRA translating the "stack" out of the Centaur Hangar, then stop, and release the CCA. No Orbiter should be shown.

LABELS none required.

REPOSITIONED CAMERA SEQUENCE

CAMERA POSITION: INSIDE CENTAUR HANGAR

SEQUENCE: OMV ATTACHMENT AND VEHICLE DEPLOYMENT

8a - 10a. Insert sequences #8 thru #10 but this time viewed with an initial camera position of that used at the end of sequence 5a. At this position, sequences 8 thru 9 will be viewed. Now, the camera will translate and pan in exactly the same way as was one in sequences 6a and 7a so that sequence 10 can be viewed.

SEQUENCE OF EVENTS TO FOLLOW:

11. "Stack" departs Space Station.
12. "Stack approaches Co-Orbiting Platform (COP).

11. "Stack" departs Space Station. ETE: 5 mon; 3 wks; 81 hrs

Camera should initially be positioned in exactly the same location as was in sequence #11; then, before the "action" starts, should translate from this position to a position on the upper surface of the moving CCA (so that a part of the OMV can be seen for reference). During this motion the camera should pan so that the Centuar Hangar remains in view.

With the camera attached to the moving CCA, the sequence should show the Station shrink in size and the Earth below rotating.

LABELS none required.

12. "Stack" approaches Co-Orbiting Platform (COP). ETE: 5 mon; 3 wks; 88 hrs

For this sequence, the camera should be positioned on the upper side of the "stack" facing in the direction of motion.

This sequence should show the "stack" approaching the COP. It should grow larger and the Earth below should be seen rotating.

LABELS: Co-Orbiting Platform (COP)

SEQUENCE OF EVENTS TO FOLLOW:

13. "Stack" rendezvous' with COP.
14. RMS grapples "stack"; OMV demates and departs.
15. CCA mated to COP.

13. "Stack" rendezvous' with COP. ETE: 5 mon; 3 wks; 89 hrs

Camera should translate and pan from the position in sequence 13 to a new position showing the COP in an isometric view (see sketch).

The "stack" should come into view and rendezvous in proximity to the COP awaiting RMS grappling.

LABELS COP RMS arm

14. RMS grapples "stack"; OMV demates and departs. ETE: 5 mon; 3 wks; 96 hrs

Camera should be in the same position as in ending of sequence 14.

The "stack" should be in the rendezvoused position (same as sequence 14 showed) and the RMS arm should grapple the CCA and the OMV should demate and depart off screen.

LABELS none required.

15. CCA mated to COP. ETE: 5 mon; 3 wks; 97 hrs

Camera should be in the same position as in sequence 15.

The RMS arm should mate the satellite and Centuar/CCA to the COP.

LABELS none required.

While attached to the COP, several events will occur:

- Cryogens will be transferred from COP to Centaur
- Final vehicle checkout
- Final satellite checkout

SEQUENCE OF EVENTS TO FOLLOW:

16. COP orients for Centaur deployment.
17. Centaur software & guidance update. (T=-xxx)
18. Deployment via CISS springs. (T=0)
19. Centaur orients itself and main engines fire.

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16. COP orients for Centaur deployment. (T=-2 hrs)

Camera should be in the same general position as in sequence 15 except zoomed away so the entire COP can be shown. The COP will orient itself (and the attached CCA) so its main axis is parallel to the Earth's surface and is facing the Space Station.

LABELS none required.

17. Centaur software & guidance update. (T=-1 hr)
18. Deployment via CISS springs. ETE: 6 mon; 0 wks; x hrs (T=0 hrs)

Camera should be in the same position as in sequence 16 and may have to zoom out and pan the Centaur. The title for 17 will appear, then title 18 will appear. Now the Centaur and satellite will deploy from the CISS and drift away and below the COP (see attached figure).

LABELS none required.

19. **Centaur orients itself and main engines fire.**
(T=4 hrs; distance=xxx miles)

Centaur and payload will orient itself as shown in the attached figure (the vehicle will rotate itself 180° from the direction it was facing previously.

LABELS none required.

APPENDIX D
COSS WBS AND WBS DICTIONARY

WBS NO.	WBS Level	WBS Description
1.0	4	CSOD TDM Program
1.1	5	Program Management
1.2	5	System Integration
1.3	5	Accommodations TDM
1.3.1	6	SE & I - Accommodations
1.3.2	6	PM - Accommodations
1.3.3	6	Berthing
1.3.3.1	7	Hangar Hardware
1.3.3.1.1	8	Truss Structure
1.3.3.1.2	8	Misc. Structure (Track)
1.3.3.1.3	8	Telerobotic Arms
1.3.3.1.4	8	Insulation & Debris Bumper
1.3.3.1.5	8	Electronics
1.3.3.1.6	8	Harness
1.3.3.2	7	Hangar Tooling
1.3.3.3	7	Hangar Assembly & C/O (Ground)
1.3.3.4	7	Hangar Assembly & C/O (Space)
1.3.3.5	7	Berthing Operation
1.3.4	6	Checkout, Maintenance, and Service
1.3.4.1	7	Tool Kits
1.3.4.2	7	ORU (Batteries, Avionics,etc)
1.3.4.3	7	C/O, Maintenance, and Service Operations
1.3.5	6	Payload Integration
1.3.5.1	7	UPA, MPA, and Interfaces
1.3.5.2	7	P/L Simulators
1.3.5.3	7	Integrate Actual P/L's
1.3.5.4	7	P/L Integration Operations
1.4	5	Operations TDM
1.4.1	6	SE & I - Operations
1.4.2	6	PM - Operations
1.4.3	6	Cryogenic Propellant Resupply
1.4.3.1	7	OMV Service
1.4.4.2	7	Ground Monitoring/Control
1.4.3.3	7	Space Station Monitoring/Control
1.4.4	6	SBTC Deployment
1.4.4.1	7	OMV Service
1.4.4.2	7	Ground Monitoring/Control
1.4.4.3	7	Space Station Monitoring/Control
1.5	5	SBTC Vehicle and Modifications
1.5.1	6	SE & I - SBTC
1.5.2	6	PM - SBTC
1.5.3	6	Titan Centaur Vehicle (dry w/ RCS)
1.5.4	6	Support Structure Modification
1.5.4.1	7	Mod. Fwd Support Structure

WBS NO.	WBS Level	WBS Description
1.5.4.2	7	Mod. Aft Adapter
1.5.5	6	Fluid & Mechanical System Modification
1.5.5.1	7	Mod. Fluid Lines & Interfaces
1.5.5.2	7	Add Liquid Acquisition Device (both tanks)
1.5.5.3	7	Add Mass Gauge (both tanks)
1.5.5.4	7	Add Diffuser/dissipator (LOX only)
1.6	5	CISS Modification
1.6.1	6	SE & I - CISS Modification
1.6.2	6	PM - CISS Modification
1.6.3	6	CISS Support Structure Modification
1.6.3.1	7	OMV, COP, and Hangar Bolt-on Structure
1.6.3.2	7	Space Handling Fixture
1.6.4	6	Fluid Lines Modification & Disconnect Panels
1.6.5	6	Mod. Electrical Disconnect Mechanism
1.6.6	6	CISS (S/C)
1.7	5	Space Station Modifications (Scars)
1.7.1	6	SE & I - Scars
1.7.2	6	PM - Scars
1.7.3	6	CCLS
1.7.4	6	Fluid, Electrical Lines, and Interfaces
1.7.5	6	Software (CCLS)
1.8	5	Co-Orbiting Platform
1.8.1	6	SE & I - COP
1.8.2	6	PM - COP
1.8.3	6	Structures
1.8.3.1	7	Core
1.8.3.2	7	CCA Interface Adapter
1.8.3.3	7	LH2 Tank & Debris Shield
1.8.3.4	7	LOX Tank & Debris Shield(2)
1.8.4	6	Power System
1.8.4.1	7	Solar Arrays
1.8.4.2	7	Array Support Structure
1.8.4.3	7	Battery
1.8.5	6	Attitude Control System
1.8.5.1	7	Tanks
1.8.5.2	7	Feed System
1.8.5.3	7	Thrusters
1.8.6	6	MRMS Modules (2)
1.8.6.1	7	MRMS Adapter (2)
1.8.6.2	7	Arms (2)
1.8.6.3	7	Movement Control Electronics
1.8.7	6	Fluid System
1.8.7.1	7	Compressors

WBS NO.	WBS Level	WBS Description
1.8.7.2	7	Accumulator
1.8.7.3	7	Pumps
1.8.7.4	7	Pressurization System
1.8.7.5	7	Thermodynamic Vent System
1.8.7.6	7	Mass Gauging
1.8.7.7	7	Liquid Acquisition Device
1.8.7.8	7	Plumbing
1.8.8	6	Passive Thermal Control
1.8.8.1	7	LOX Tank (2)
1.8.8.2	7	LH2 Tank
1.8.9	6	Emergency Jettison System
1.8.10	6	Data Management
1.8.10.1	7	Guidance, Navigation and Control
1.8.10.2	7	Electrical Equipment
1.8.10.3	7	R.F. Systems
1.8.10.4	7	Instrumentation & Data Acquisition
1.8.10.5	7	Tracking System
1.8.10.6	7	CCLS
1.8.11	6	Software
1.8.11.1	7	Launch Operation Software
1.8.11.2	7	Systems Control Software
1.8.12	6	Tooling
1.8.13	6	Ground Assembly & C/O
1.8.14	6	Space Assembly
1.9	5	Delivery Transportation
1.9.1	6	SE & I - Transportation
1.9.2	6	PM - Transportation
1.9.3	6	Logistics ASE
1.9.4	6	TDM Equipment (COP, Hangar, CCLS, etc.)
1.9.5	6	SBTC and P/L

COST ELEMENT	DEFINITION
DDT&E PHASE	This cost elements refers to the total cost of developing the Commercial Space Operations Development (CSOD) program, begining with the conceptual and definition activities and concluding when the system element are ready for operational use. Included is design, development, ground test, and initial spares of the manufacturing hardware.
DESIGN & DEVELOPMENT	This element includes the cost of interpreting the CSOD system requirements and translating these requirements into the generation of design drawings, models, and other written and constructed representations that guide the manufacture and test of the CSOD hardware. This involves the successive eration of designs and models throughout the DDT&E phase, from conceptual design through full-scale development.
GROUND TEST	This element includes the cost of manufacturing major subsystems and complete the ground test needed for thermal, structural, dynamic testing, avionics system tests, and all systems tests of the CSOD program.
INITAL SPARES	This is the costs of manufacturing the initial spares that must be available at system prior transportation delivery.
FLIGHT HARDWARE	Included in this element are the costs of manufacturing production hardware for the CSOD, excluding the manufacture of any components produced and refurbished for use in ground test and validation in the DDT&E phase.

1.0 CSOD TDM PROGRAM	This element is the total cost of developing, assembling, and demonstrate the new technologies required for the CSOD program. Included are all labor, material, and overhead required for the design, development, fabrication, assembly, testing, operation, and additional one commercial satellite launch at the COP.
1.1 PROGRAM MANAGEMENT	This element includes the costs associated with program administration and management, planning and scheduling , and financial and administrative support for major system or for total CSOD program.
1.2 SYSTEM ENGINEERING & INTEGRATION	This is the costs of the systems engineering effort that directly supports manufacturing. Included is the coordination of the various manufacturing activities on an inter-departmental basis and with subcontractors and vendors. Also included are continued engineering such as design changes, product improvement, and associated technology evelopment program for major system or total CSOD program.
1.3 ACCOMMODATION TDM	This element is one of the seven major system of CSOD program which includes the total cost of developing, manufacture flight hardware, groundassembling, and operating of the three major accommodations technology demonstration mission (TDM): Berthing, Checkout and maintenance service, and Payload ntegration.
1.3.1 BERTHING	This WBS element refers to the total cost of the hangar which will be attached to the Space Station (SS) and the associated operation costs to bring the hangar to the Initial Operating capacity (IOC) stage.

1.3.1.1 HANGAR HARDWARE	This element is the cost of the principal hardware elements of the berthing hangar. Included are the components designed to protect, Checkout Maintenance & Service, and Payload Integration for he Space Base Titan/Centaur (SBTC), including truss structure, misc. structure, telerobototic arms, insulation & debris bumper, electronics, and harness.
1.3.1.1.1 TRUSS STRUCTURE	This element is the cost of the principal structural elements of the hangar which will be attached to the Space Station (SS).
1.3.1.1.2 TRACK	This refers to the cost of the structural tracks which two telerobotic arms will be traveled on and performed all the accommodation operation.
1.4 OPERATION TDM	This element is the cost of operation TDM for CSOD program which includes the total cost of developing, manufacture flight hardware, ground assembling, and operating of the cryogenic propellant resupply and a SBTC final commercial satellite deployment.
1.5 SBTC VEH. & MOD.	This is the costs of Space Based Titan Centaur (SBTC) vehicle which includes the total modification costs to the current Titan/Centaur and manufacture a SBTC flight hardware. No cost of space base maintainability allowance for this SBTC.
1.6 CISS MODIFICATION	This refers to the cost of modifiication of the existing Shuttle Centaur (S/C) Centaur Integration Support Structure (CISS) to fit in new CSOD SBTC assumed no cost for the existing S/C CISS. No cost of space base maintainability allowance for this Space Based CISS.

1.7 SS SCARS	This is the cost of modification of the Space Station which includes the costs of the CCLS, fluid line, electrical monitoring system and interface, and the software for CCLS.
1.8 CO-ORBITING PLATFORM	This refers to the total cost of developing, manufacture flight hardware & software, ground assembling, space assembling and ground test for Co-orbiting platform. Included are all labor, material, and overhead required for the design, development, fabrication, assembly, and testing for the COP.
1.9 DELIVERY TRANSPORTATION	This is the total cost of developing and manufacturing all the Airbone Support Equipment (ASE) and all the launch service costs for delivery transportation required by the CSOD components.

APPENDIX E
COSS AND STV COST MODELS AND INPUTS

CSOD Program Cost Model

WBS NO.	WBS Level	WBS Description	Wt. (lb)/ TFU Key	Key Para./ DDT&E Key	Dev't Thruput	% New Dev't	Dev't Cplxty	T1 Mfg. Thruput	Mfg. Cplxty	Grd. Test SS	D&D Coef.	D&D Exp.	T1 Coef.	T1 Exp.	DDT&E Phase: Design & Dev't	DDT&E Phase: Ground Test	DDT&E Phase: Initial Spares	DDT&E Phase: Total DDT&E	Flt. Hardware /Mfg. TFU
1.1	5	Program Management	13.2 M$	43.8 M$	0.00	100%	1.0	0.00	1.0	1.1	0.100	1.000	0.100	1.000	4.38	4.82	0.88	10.07	4.38
1.2	5	System Integration	14.0 M$	75.71 $	0.00	100%	1.0	0.00	1.0	1.1	0.100	1.000	0.100	1.000	7.57	8.33	1.51	17.41	7.57
1.3	5	Accommodations TDM	10843 lb	10865 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	101.56	27.15	5.87	134.58	38.32
1.3.1	6	SE & I - Accommodations	31.9 M$	76.2 M$	0.00	100%	1.0	0.00	1.0	1.1	0.421	0.840	0.220	0.779	16.02	3.59	0.65	20.26	3.27
1.3.2	6	PM - Accommodations	31.9 M$	76.2 M$	0.00	100%	1.0	0.00	1.0	1.1	0.528	0.662	0.235	0.750	9.31	3.47	0.63	13.41	3.15
1.3.3	6	Berthing	5968 lb	5968 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	61.29	14.50	3.56	79.36	26.71
1.3.3.1	7	Hangar Hardware	5968 lb	5968 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	41.07	12.72	3.56	57.35	17.82
1.3.3.1.1	8	Truss Structure	1526 lb	1526 lb	0.00	100%	1.2	0.00	1.5	1.1	0.407	0.430	0.008	0.667	11.84	1.64	0.30	13.79	1.49
1.3.3.1.2	8	Misc. Structure (Track)	300 lb	300 lb	0.00	100%	1.2	0.00	1.5	1.1	0.407	0.430	0.008	0.667	5.88	0.56	0.10	6.54	0.51
1.3.3.1.3	8	Telerobotic Arms	1150 lb	1150 lb	0.00	50%	1.0	0.00	1.0	1.1	0.580	0.500	0.069	0.700	6.96	6.48	2.36	15.79	11.78
1.3.3.1.4	8	Insulation & Debris Bumper	2892 lb	2892 lb	0.00	100%	1.0	0.00	1.0	1.1	0.045	0.700	0.003	0.907	12.03	3.99	0.73	16.75	3.63
1.3.3.1.5	8	Electronics	50 lb	50 lb	0.00	100%	0.3	0.00	1.0	0.1	1.379	0.579	0.008	0.917	3.32	0.03	0.05	3.40	0.27
1.3.3.1.6	8	Harness	50 lb	50 lb	0.00	100%	1.0	0.00	1.0	0.1	0.092	0.620	0.005	0.870	1.04	0.01	0.03	1.08	0.14
1.3.3.2	7	Hangar Tooling	1826 lb	1826 lb	0.00	100%	1.0	0.00	1.0	0.0	0.251	0.574	0.000	1.000	18.69	0.00	0.00	18.69	0.00
1.3.3.3	7	Hangar Assembly & C/O(Ground)	17.8 M$	41.1 M$	0.00	100%	1.0	0.00	1.0	1.0	0.033	1.000	0.100	1.000	1.36	1.78	0.00	3.14	1.78
1.3.3.3	7	Hangar Assembly & C/O(Space)	80 U-hr	0.0 M$	0.00	100%	1.0	7.10	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	7.10
1.3.3.5	7	Berthing Operation	N/A	9 U-hr	0.00	100%	1.0	0.00	1.0	0.0	0.020	1.000	0.000	1.000	0.18	0.00	0.00	0.18	0.00
1.3.4	6	Checkout, Maintenance, and Service	700 lb	722 lb	N/A	N/A	N/A	N/A	N/A	N/A	0.000	N/A	N/A	N/A	3.76	0.77	0.14	4.66	0.00
1.3.4.1	7	Tool Kits	100 lb	100 lb	0.00	100%	1.0	0.00	1.0	1.1	0.407	0.430	0.008	0.667	2.96	0.18	0.03	3.17	0.16
1.3.4.2	7	ORU (Batteries, Avionics,etc)	600 lb	600 lb	0.00	0%	1.0	0.00	1.0	1.1	0.000	0.430	0.008	0.667	0.00	0.59	0.11	0.70	0.53
1.3.4.3	7	C/O, Maintenance, and Service Operations	N/A	22 U-hr	0.80	100%	1.0	0.00	1.0	0.0	0.000	1.000	0.000	1.000	0.80	0.00	0.00	0.80	0.00
1.3.5	6	Payload Integration	4175 lb	4175 lb	N/A	N/A	N/A	N/A	N/A	N/A	0.000	N/A	N/A	N/A	11.19	4.83	0.86	16.89	5.19
1.3.5.1	7	UPA, MPA, and Interfaces	1175 lb	1175 lb	0.00	100%	1.5	0.00	1.5	1.1	0.237	0.430	0.006	0.667	8.51	1.92	0.35	10.78	1.74
1.3.5.2	7	P/L Simulators	3000 lb	3000 lb	0.00	10%	1.0	0.00	1.0	1.1	0.407	0.430	0.008	0.667	1.28	2.91	0.53	4.72	2.65
1.3.5.3	7	Integrate Actual P/L's	22 U-hr	N/A	0.00	0%	1.0	0.80	1.0	0.0	1.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	0.80
1.3.5.4	7	P/L Integration Operations	N/A	52 U-hr	1.39	100%	1.0	0.00	1.0	0.0	0.000	1.000	0.000	1.000	1.39	0.00	0.00	1.39	0.00
1.4	5	Operations TDM	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	7.25	2.42	0.44	10.11	10.08
1.4.1	6	SE & I - Operations	7.9 M$	4.4 M$	0.00	100%	1.0	0.00	1.0	1.1	0.421	0.840	0.220	0.779	1.46	1.21	0.22	2.88	1.10
1.4.2	6	PM - Operations	7.9 M$	4.4 M$	0.00	100%	1.0	0.00	1.0	1.1	0.528	0.662	0.235	0.750	1.40	1.22	0.22	2.84	1.11
1.4.3	6	Cryogenic Propellant Resupply	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	4.38	0.00	0.00	4.38	3.55
1.4.3.1	7	OMV Service	1 Flt	1 Flt	0.00	0%	1.0	3.55	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	3.55
1.4.4.2	7	Ground Monitoring/Control	1 Flt	1 Flt	0.00	100%	1.0	0.00	1.0	0.0	0.440	1.000	0.000	1.000	0.44	0.00	0.00	0.44	0.00
1.4.3.3	7	Space Station Monitoring/Control(Tanking)	N/A	200 U-hr	3.95	100%	1.0	0.00	1.0	0.0	0.000	1.000	0.000	1.000	3.95	0.00	0.00	3.95	0.00
1.4.4	6	SBTC Deployment	1 Flt	1 Flt	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	0.00	0.00	0.00	0.00	4.33
1.4.4.1	7	OMV Service	1 Flt	1 Flt	0.00	0%	1.0	3.55	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	3.55
1.4.4.2	7	Ground Monitoring/Control	1 Flt	1 Flt	0.00	100%	1.0	0.00	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	0.00
1.4.4.3	7	Space Station Monitoring/Control	40 U-hr	N/A	0.00	10%	1.0	0.79	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	0.79
1.5	5	SBTC Vehicle and Modifications	8429 lb	8429 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	20.57	5.45	1.52	27.54	37.70
1.5.1	6	SE & I - SBTC	5.8 M$	13.8 M$	0.00	100%	1.0	0.00	1.0	0.0	0.421	0.840	0.220	0.779	3.81	0.00	0.17	3.98	0.87
1.5.2	6	PM - SBTC	5.8 M$	13.8 M$	0.00	100%	1.0	0.00	1.0	0.0	0.528	0.662	0.235	0.750	3.00	0.00	0.18	3.17	0.88
1.5.3	6	Titan Centaur Vehicle (dry w/ RCS)	6720 lb	6720 lb	0.00	0%	1.0	30.10	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	30.10
1.5.4	6	Support Structure Modification	653 lb	653 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	2.97	0.70	0.14	3.82	0.72
1.5.4.1	7	Mod. Fwd Support Structure	711 lb	711 lb	2.69	100%	1.0	0.08	1.0	0.0	0.237	0.430	0.006	0.667	2.69	0.00	0.02	2.71	0.08
1.5.4.2	7	Mod. Aft Adapter	-58 lb	-58 lb	0.28	100%	1.0	0.64	1.0	1.1	0.000	1.000	0.000	1.000	0.28	0.70	0.13	1.11	0.64
1.5.5	6	Fluid & Mechanical System Modification	1056 lb	1056 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	10.80	4.74	1.02	16.57	5.12
1.5.5.1	7	Mod. Fluid Lines & Interfaces	836 lb	836 lb	0.00	100%	1.0	0.00	1.0	1.1	0.266	0.500	0.048	0.667	7.68	4.74	0.86	13.29	4.31
1.5.5.2	7	Add Liquid Acquisition Device (both tanks)	205 lb	205 lb	0.00	50%	1.0	0.00	1.0	1.1	0.139	0.500	0.015	0.667	1.00	0.00	0.10	1.10	0.51
1.5.5.3	7	Add Mass Gauge (both tanks)	5 lb	5 lb	0.00	50%	1.0	0.00	1.0	1.1	1.318	0.500	0.069	0.667	1.47	0.00	0.04	1.51	0.20
1.5.5.4	7	Add Diffuser/dissipator (LOX only)	10 lb	10 lb	0.00	50%	1.0	0.00	1.0	1.1	0.412	0.500	0.021	0.667	0.65	0.00	0.02	0.67	0.10
1.6	5	CISS Modification	7669 lb	7669 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	14.58	2.37	0.44	17.39	2.21
1.6.1	6	SE & I - CISS Modification	1.6 M$	9.5 M$	0.00	100%	1.0	0.00	1.0	1.0	0.421	0.840	0.220	0.779	2.78	0.31	0.06	3.15	0.31
1.6.2	6	PM - CISS Modification	1.6 M$	9.5 M$	0.00	100%	1.0	0.00	1.0	1.0	0.528	0.662	0.235	0.750	2.34	0.33	0.07	2.73	0.33
1.6.3	6	CISS Support Structure Modification	460 lb	460 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	4.87	0.53	0.10	5.49	0.48
1.6.3.1	7	OMV, COP, and Hangar Bolt-on Structure	180 lb	180 lb	0.00	100%	1.0	0.00	1.0	1.1	0.250	0.430	0.006	0.667	2.33	0.22	0.04	2.60	0.20
1.6.3.2	7	Space Handling Fixture	280 lb	280 lb	0.00	100%	1.0	0.00	1.0	1.1	0.407	0.430	0.006	0.667	2.54	0.30	0.05	2.89	0.27
1.6.4	6	Fluid Lines Modification & Disconnect Panels	80 lb	80 lb	0.00	100%	1.0	0.00	1.0	1.1	0.266	0.500	0.048	0.667	2.38	0.99	0.18	3.55	0.90
1.6.5	6	Mod. Electrical Disconnect Mechanism	160 lb	160 lb	0.00	100%	1.0	0.00	1.0	1.1	0.250	0.430	0.006	0.667	2.22	0.21	0.04	2.46	0.19
1.6.6	6	CISS (S/C)	6969 lb	6969 lb	0.00	0%	1.0	0.00	0.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	0.00
1.7	5	Space Station Modifications (Scars)	210 lb	210 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	25.01	5.69	1.03	31.73	5.17
1.7.1	6	SE & I - Scars	3.9 M$	17.0 M$	0.00	100%	1.0	0.00	1.0	1.1	0.421	0.840	0.220	0.779	4.55	0.70	0.13	5.37	0.83
1.7.2	6	PM - Scars	3.9 M$	17.0 M$	0.00	100%	1.0	0.00	1.0	1.1	0.528	0.662	0.235	0.750	3.45	0.72	0.13	4.29	0.65
1.7.3	6	CCLS	150 lb	150 lb	0.00	50%	1.0	0.00	1.0	1.1	0.271	0.813	0.025	0.961	7.96	3.46	0.63	12.04	3.14
1.7.4	6	Fluid, Electrical Lines, and Interfaces	60 lb	60 lb	0.00	100%	1.0	0.00	1.0	1.1	0.266	0.500	0.048	0.667	2.06	0.82	0.15	3.02	0.74

1.7.5	6	Software (CCLS)	0 klc	20 klc	0.00	100%	1.0	0.00	1.0	0.0	0.354	1.000	0.000	1.000	7.00	0.00	0.00	7.00	0.00
1.8	5	Co-Orbiting Platform	32173 lb	32173 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	350.43	100.28	20.13	470.82	113.66
1.8.1	6	SE & I - COP	86.8 M$	214.1 M$	0.00	100%	1.0	0.00	1.0	1.1	0.421	0.840	0.220	0.779	38.11	7.84	1.43	47.38	7.13
1.8.2	6	PM - COP	86.8 M$	214.1 M$	0.00	100%	1.0	0.00	1.0	1.1	0.528	0.662	0.235	0.750	18.44	7.35	1.34	27.12	6.68
1.8.3	6	Structures	19463 lb	19463 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	79.52	28.94	5.26	113.72	26.31
1.8.3.1	7	Core	4800 lb	4800 lb	0.00	100%	1.0	0.00	1.0	1.1	1.381	0.313	0.030	0.667	19.60	9.26	1.68	30.55	8.42
1.8.3.2	7	CCA Interface Adapter	350 lb	350 lb	0.00	100%	1.0	0.00	1.0	1.1	1.381	0.313	0.030	0.667	8.64	1.62	0.29	10.55	1.47
1.8.3.3	7	LH2 Tank & Debris Shield	8538 lb	8538 lb	0.00	100%	1.0	0.00	1.0	1.1	1.658	0.309	0.022	0.667	27.19	10.20	1.85	39.24	9.27
1.8.3.4	7	LOX Tank & Debris Shield(2)	5775 lb	5775 lb	0.00	100%	1.0	0.00	1.0	1.1	1.658	0.309	0.022	0.667	24.09	7.86	1.43	33.38	7.15
1.8.4	6	Power System	1170 lb	1170 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	37.19	4.62	0.84	42.65	4.20
1.8.4.1	7	Solar Arrays	320 lb	8000 w	0.00	100%	1.0	0.00	1.0	1.1	0.017	0.742	0.207	0.473	13.28	3.48	0.63	17.39	3.16
1.8.4.2	7	Array Support Structure	200 lb	200 lb	0.00	100%	1.0	0.00	1.0	1.1	1.381	0.313	0.030	0.667	23.00	1.11	0.20	24.32	1.01
1.8.4.3	7	Battery	650 lb	2.4 kwh	0.00	50%	1.0	0.00	1.0	1.1	1.705	0.078	0.011	0.143	0.91	0.03	0.01	0.95	0.03
1.8.5	6	Attitude Control System	302 lb	302 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	3.56	1.93	0.35	5.84	1.75
1.8.5.1	7	Tanks	100 lb	100 lb	0.00	100%	1.0	0.00	1.7	1.1	0.031	0.500	0.002	0.667	0.31	0.08	0.01	0.40	0.07
1.8.5.2	7	Feed System	150 lb	150 lb	0.00	100%	1.0	0.00	1.0	1.1	0.266	0.500	0.046	0.667	3.25	1.43	0.26	4.94	1.30
1.8.5.3	7	Thrusters	52 lb	52 lb	0.00	0%	1.0	0.04	1.0	1.1	0.000	1.000	0.000	1.000	0.00	0.42	0.08	0.50	0.38
1.8.6	6	MRMS Modules (2)	6050 lb	6050 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	17.93	19.80	7.08	44.80	35.39
1.8.6.1	7	MRMS Adapter (2)	650 lb	650 lb	0.00	100%	1.0	0.00	1.0	1.1	0.237	0.430	0.006	0.667	2.86	0.67	0.12	3.64	0.61
1.8.6.2	7	Arms (2)	5300 lb	5300 lb	0.00	50%	1.0	0.00	1.0	1.1	0.580	0.500	0.069	0.700	15.07	19.13	6.96	41.16	34.78
1.8.6.3	7	Movement Control Electronics	100 lb	100 lb	0.00	0%	1.0	0.00	1.0	1.1	0.000	0.500	0.000	0.700	0.00	0.00	0.00	0.00	0.00
1.8.7	6	Fluid System	1946 lb	1946 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	20.08	7.37	1.34	28.79	6.70
1.8.7.1	7	Compressors	72 lb	72 lb	0.00	50%	1.0	0.00	1.0	1.1	0.412	0.500	0.021	0.667	1.75	0.40	0.07	2.22	0.37
1.8.7.2	7	Accumulator	750 lb	750 lb	0.00	50%	1.0	0.00	1.0	1.1	0.412	0.500	0.021	0.667	5.64	1.92	0.35	7.91	1.74
1.8.7.3	7	Pumps	552 lb	552 lb	0.00	50%	1.0	0.00	1.0	1.1	0.412	0.500	0.021	0.667	4.84	1.56	0.28	6.69	1.42
1.8.7.4	7	Pressurization System	48 lb	48 lb	0.00	50%	1.0	0.00	1.0	1.1	0.412	0.500	0.021	0.667	1.43	0.31	0.06	1.79	0.28
1.8.7.5	7	Thermodynamic Vent System	31 lb	31 lb	0.00	50%	1.0	0.00	1.0	1.1	0.279	0.500	0.031	0.667	0.78	0.33	0.06	1.17	0.30
1.8.7.6	7	Mass Gauging	17 lb	17 lb	0.00	50%	1.0	0.00	1.0	1.1	1.318	0.500	0.069	0.667	2.72	0.50	0.09	3.31	0.45
1.8.7.7	7	Liquid Acquisition Device	312 lb	312 lb	0.00	50%	1.0	0.00	1.0	1.1	0.139	0.500	0.015	0.667	1.23	0.75	0.14	2.11	0.68
1.8.7.8	7	Plumbing	164 lb	164 lb	0.00	50%	1.0	0.00	1.0	1.1	0.266	0.500	0.048	0.667	1.70	1.60	0.29	3.59	1.46
1.8.8	6	Passive Thermal Control	2592 lb	2592 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	11.59	3.84	0.70	16.12	3.49
1.8.8.1	7	LOX Tank (2)	836 lb	836 lb	0.00	100%	1.0	0.00	1.0	1.1	0.045	0.700	0.003	0.907	3.11	1.30	0.24	4.64	1.18
1.8.8.2	7	LH2 Tank	1756 lb	1756 lb	0.00	100%	1.0	0.00	1.0	1.1	0.045	0.700	0.003	0.907	8.48	2.54	0.46	11.48	2.31
1.8.9	6	Emergency Jettison System	50 lb	50 lb	0.00	100%	1.0	0.00	1.0	1.1	0.237	0.430	0.006	0.667	1.28	0.10	0.02	1.39	0.09
1.8.10	6	Data Management	600 lb	600 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	29.95	9.80	1.78	41.53	8.91
1.8.10.1	7	Guidance, Navigation and Control	80 lb	80 lb	0.00	100%	1.0	0.00	1.0	1.1	0.271	0.813	0.025	0.961	9.54	1.89	0.34	11.78	1.72
1.8.10.2	7	Electrical Equipment	200 lb	200 lb	8.64	100%	1.0	0.88	1.0	1.1	0.000	1.000	0.000	1.000	8.64	0.97	0.18	9.79	0.88
1.8.10.3	7	R.F. Systems	80 lb	80 lb	0.00	100%	1.0	0.00	1.0	1.1	0.281	0.579	0.034	0.917	3.56	2.06	0.37	5.99	1.87
1.8.10.4	7	Instrumentation & Data Acquisition	60 lb	60 lb	0.00	100%	1.0	0.00	1.0	1.1	0.328	0.579	0.010	0.917	3.51	0.48	0.09	4.07	0.43
1.8.10.5	7	Tracking System	80 lb	80 lb	0.00	100%	1.0	0.00	1.0	1.1	0.281	0.579	0.034	0.917	3.56	2.06	0.37	5.99	1.87
1.8.10.6	7	CCLS	100 lb	100 lb	0.00	10%	1.0	0.00	1.0	1.1	0.271	0.813	0.025	0.961	1.14	2.34	0.43	3.91	2.13
1.8.11	6	Software	0 klc	37 klc	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	13.01	0.00	0.00	13.01	0.00
1.8.11.1	7	Launch Operation Software	0 klc	14 klc	0.00	100%	1.0	0.00	1.0	0.0	0.354	1.000	0.000	1.000	5.00	0.00	0.00	5.00	0.00
1.8.11.2	7	Systems Control Software	0 klc	23 klc	0.00	100%	1.0	0.00	1.0	0.0	0.354	1.000	0.000	1.000	8.00	0.00	0.00	8.00	0.00
1.8.12	6	Tooling	19663 lb	19663 lb	0.00	100%	1.0	0.00	1.0	0.0	0.251	0.574	0.000	1.000	73.14	0.00	0.00	73.14	0.00
1.8.13	6	Ground Assembly & C/O	86.8 M$	201.1 M$	0.00	100%	1.0	0.00	1.0	1.0	0.033	1.000	0.100	1.000	6.64	8.68	0.00	15.32	8.68
1.8.14	6	Space Assembly	80 U-hr	0 U-hr	0.00	100%	1.0	4.33	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	4.33
1.9	5	Delivery Transportation	9000 lb	9000 lb	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	N/A	53.22	6.79	1.24	61.25	235.81
1.9.1	6	SE & I - Transportation	4.7 M$	38.3 M$	0.00	100%	1.0	0.00	1.0	1.1	0.421	0.840	0.220	0.779	8.99	0.81	0.15	9.94	0.73
1.9.2	6	PM - Transportation	4.7 M$	38.3 M$	0.00	100%	1.0	0.00	1.0	1.1	0.528	0.662	0.235	0.750	5.90	0.82	0.15	6.88	0.75
1.9.3	6	Logistics ASE	9000 lb	9000 lb	0.00	100%	1.0	0.00	1.0	1.1	0.407	0.430	0.008	0.667	38.32	5.16	0.94	44.43	4.69
1.9.4	6	TDM Equipment (COP, Hangar, CCLS, etc.)	3 Flt	0 Flt	0.00	100%	1.0	76.55	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	229.64
1.9.5	6	SBTC and P/L	2 Flt	0 Flt	0.00	100%	1.0	103.00	1.0	0.0	0.000	1.000	0.000	1.000	0.00	0.00	0.00	0.00	0.00
1.0	4	CSOD TDM Program Total Cost	68324 lb	68346 lb											584.56	163.29	33.06	780.91	454.91
		Fee @ 10%													58.46	18.33	3.31	78.09	45.49
		Total CSOD TDM Price													643.02	179.61	36.36	859.00	500.40

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CSOD Program Technical Requirements Input

SOFTWARE REQUIREMENTS FOR COST REPORTING

DESCRIPTION: LAUNCH OPERATIONS SOFTWARE
APPLICATION: CCLS COMPUTER AT COP, STATION AND GROUND

LANGUAGE:	ADA
INSTRUCTIONS:	14,000 LINES
% TOTAL EFFORT:	25%

DESCRIPTION: AVIONICS/FLUIDS CHECKOUT SOFTWARE
APPLICATION: CCLS COMPUTER AT COP, STATION AND GROUND

LANGUAGE:	ADA
INSTRUCTIONS:	20,000 LINES
% TOTAL EFFORT:	35%

DESCRIPTION: SYSTEMS SOFTWARE
APPLICATION: CCLS COMPUTER AT COP, STATION AND GROUND

LANGUAGE:	ADA
INSTRUCTIONS:	23,000 LINES
% TOTAL EFFORT:	40%

COP AVIONICS REQUIREMENTS FOR COST REPORTING

UNIT	WEIGHT	UNIT	WEIGHT
Flt Control Processor	90	S-Band Transmitter	10
IMU	108	S-Band Antennas (3)	3
Rate Gyros	43	S-Band Pre-Amp and Amp	5
Remote Voter Unit	84	C-Band Transponder	4
MDU	22	C-Band Antennas (2)	12
RDU	29	C-Band Amp	5
Sensors	120	TWTA	30
Harnessing	78	TIU	30
CCLS Computer	100	Signal Conditioners (3)	50
Propellant Control	45		

Structure Input Sheet

Description Multiple Payload Adapter

Structure Input Sheet

Description Hangar - Truss Structure

Accom, - TDM

Structure Input Sheet

Description Universal Payload Adapter

Application Payload Integration to MRA - Accom, P/L Integ. TDM

1. Diameter (max in inches)		50"		
2. Length (feet)		2'		
3. Gross weight (k-lb)		95#/UPA		
4. Primary Material (Enter composition of each per stage by percent)				
Aluminum Alloy		100%		
Titanium				
Steel				
Other				
	Total	100%	100%	100%
5. Construction (Enter composition of each per stage by percent)				
Skin-Stringer-Frame		100%		
Isogrid				
Carbonfiber Sandwich				
Other				
	Total	100%	100%	100%
6. Fabrication (Enter % of each per stage)				
Conventional Welding		30%		
Conventional Riveting		30%		
One-way Bolts		40%		
Other				
	Total	100%	100%	100%

Coments:

APPENDIX F
COSS AND STV TEST PLANS

...AL PAGE IS
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GENERAL DYNAMICS
SPACE SYSTEMS DIVISION

Technology Demontration Missions

Test Plan Outline

For

Commercial Space Operations

Development Program

February 22, 1988
838-0-88-063

Prepared by

Clifton B. Phillips
Test Planning -Advanced Programs
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Table of Contents

Acronym Table

CCA	Centaur/CISS Assembly
CCLS	Computer Controlled Launch Set
CISS	Centaur Integrated Support System
COP	Co-Orbiting Platform
CPOCC	Centaur Payload Operations Control Center
EVA	Extra-Vehicular Activity
FCP	Flight Control Processor
GDSSD	General Dynamics Space Systems Division
GHe	Gaseous Helium
GPS	Global Positioning System
IVA	Inter-Vehicular Activity
LH_2	Liquid Hydrogen
LO_2	Liquid Oxygen
MCCH	Mission Control Center Houston
MPA	Multiple Payload Adapter
MRMS	Mobile Remote Manipulating System
OMV	Orbital Maneuvering Vehicle
ORU	Oribital Replaceable Unit
OTV	Orbital Transfer Vehicle
POCC	Payload Operations Control Center
RCS	Reaction Control System
RMS	Remote Manipulating System
SBTC	Space-Based Titan/Centaur
SPF	Spacecraft Processing Facility
STS	Space Transportation System
TBD	To Be Determined
TDM	Technology Demonstration Mission
TDRS	Tracking and Data Relay Satellite
TDRSS	Tracking and Data Relay Satellite System
TRA	Telerobotic Arm
UPA	Universal Payload Adapter
WSGS	White Sands Ground Station

1.0 INTRODUCTION

The Space-Based Titan/Centaur (SBTC) rocket with a Centaur Integrated Support System (CISS) will utilize the Space Station to perform a mission and demonstrate the new technologies required for space-based Orbital Transfer Vehicle (OTV) mission success.

Technology Demonstration Missions (TDM's) are experiments at or in the vicinity of the Space Station. Their purpose is to improve the Space Station accommodations and operations for the space based OTV.

2.0 FOREWORD

Five experiments have been incorporated into two TDM's using a (SBTC) Centaur/CISS assembly (CCA) over a nine month period. These TDM's will develop and demonstrate accommodations and operations required by an OTV at the Space Station, using the (SBTC) as an OTV simulator.

The two identified TDM's are structured as follows:

1. Accommodations TDM
   - Berthing
   - Checkout, Maintenance and Servicing
   - Payload Integration
2. Operations TDM
   - Cryogenic Propellant Resupply
   - Centaur Launch Deployment

A hanger shall be constructed at the Space Station to be used for berthing and storing the SBTC. A "dry" (no loaded cryogens) CCA with full gaseous helium (GHe) and hydrazine bottles will be delivered to the Space Station by the Space Transportation System (STS) to complete the Accommodations TDM. An advanced booster, such as ALS or shuttle C, could also be used for delivery.

After the Accommodations TDM has been completed, the SBTC will undergo a system checkout and a real payload will be ready for deployment on a mission. The Orbital Maneuvering Vehicle (OMV) will have been attached to the aft end of the CCA and the Operations TDM will start.

The Operations TDM will include transport from the Space Station to a co-orbiting platform (COP) where cryogenic tanking and deployment operations will occur.

JSC Mission Control Center Houston (MCCH) will be at the hub of launch operations. After payload and Centaur payload operation control centers (POCC and CPOCC) give MCCH the "go" signal, the deployment sequence begins. After Centaur deployment, the Space Station will supply ranging information until the Centaur is out of range.

After Centaur deployment, the CSOD program is over. Reusable items shall be returned to the Space Station. Non-reusables and the CISS shall be returned to the ground by the STS.

3.0 TECHNOLOGY DEMONSTRATION MISSIONS

3.1 Accommodations TDM

3.1.1. Berthing

3.1.1.1 **Summary**

The STS shall deliver a hangar kit to the Space Station. The hangar will be constructed and attached to the Space Station. Electrical power, data communication, a helium interface, and micrometeriod shields will be added to prepare the hangar for accommodating SBTC.

A dry SBTC with no loaded cryogens, containing fully charged helium and hydrazine bottles will be delivered to the Space Station by the STS. The Space Shuttle Oribiter RMS will remove the Centaur/CISS assembly (CCA) from the docked Orbiter and hand the assembly off to the Space Station MRMS arm. The CCA will then be transferred to the hangar, where it will be handed off to the hangar TRA and berthed at the back of the hangar to complete the berthing portion of the Accommodations TDM.

3.1.1.2 **Objectives**

3.1.1.2.1 Demonstrate that berthing can be accomplished in the low-g space environment at the Space Station.

3.1.1.2.2 Gain experience in usage of required tools, grappling fixtures, remote manipulators, and telerobotic arms while performing berthing sequence.

3.1.1.2.3 Develop required procedures for berthing space-based OTV.

3.1.1.3 **Requirements**

3.1.1.3.1 Construct hangar in the low-g environment at the Space Station to protect Centaur and provide a captive environment to perform EVA and teleoperations while performing the Accommodations TDM.

3.1.1.3.2 Conduct the berthing maneuvers in the low-g environment at the Space Station.

3.1.1.4 **Configuration**

3.1.1.4.1 The CCA assembly shall be modified to accommodate the berthing TDM.

3.1.1.4.2 The CCA assembly shall remain in the mated position throughout the berthing sequence.

3.1.1.5 **Special Instrumentation Requirements**

Video, voice, TDRSS, and CCLS Data link to Space Station.

3.1.1.6 **Berthing Sequence**

3.1.1.6.1 Remove hangar components from STS, attach segments to station truss, deploy hangar walls and complete structural assembly.

3.1.1.6.2 Hook up interface connections and verify assembly before power up.

3.1.1.6.3 Perform checkout of hangar assembly lights, electrical, data communication and helium interface connection.

3.1.1.6.4 Using the Orbiter RMS, remove the CCA from the Orbiter cargo bay and hand off to the Space Station MRMS.

3.1.1.6.5 After visual inspection of CCA, transport the vehicle to the hangar using the Space Station MRMS.

3.1.1.6.6. Transfer control of the CCA to the hangar telerobotic arm (TRA).

3.1.1.6.7 After visual inspection, mate the Centaur/CISS assembly with the berthing fixture and engage the latching mechanisms.

3.1.1.7 **Remarks**

During periods of storage the checkout portion of the Accommodations TDM will be performed every seven days. The initial berthed storage period will be approximately $2^1/_2$ months. Shorter berthing periods of approximately two weeks will occur after the cryogenic propellant transfer experiment.

3.1.2 <u>Checkout, Maintenance and Servicing</u>

3.1.2.1 **Summary**

The checkout, maintenance and servicing aspect develops the procedures and tooling required to perform these operations on a space-based OTV.

Centaur checkout will be accomplished by the Space Station computer controlled launch set (CCLS) data link through the CISS. Umbilicals connect the Centaur/CISS assembly (CCA) to the Space Station through the fluid and electrical interface panels.

On-orbit replaceable units (ORU's) will be removed and replaced to gain experience performing space-based maintenance and servicing functions.

The ORU's to be removed and replaced during this aspect of the Accommodations TDM are as follows:

a. Avionics Flight Control Processor (FCP)

b. Battery

c. CISS Helium Bottle

3.1.2.2 **Objectives**

3.1.2.2.1 Demonstrate checkout, maintenance and servicing can be accomplished in the low-g space environment at the Space Station.

3.1.2.2.2 Gain experience in usage of software, required tools, grappling fixtures, remote manipulators, telerobotic arms, and procedures while performing the checkout, maintenance and servicing sequence.

3.1.2.2.3 Develop required procedures for checkout, maintenance and servicing of space-based OTV.

3.1.2.3 <u>Requirements</u>

3.1.2.3.1 During CCA residence at the Space Station, continuously monitor all tank pressures and temperatures, power to avionics, and temperatures at avionics

3.1.2.3.2 Perform space-based checkout procedures and relay data to ground via the tracking and data relay satellite system (TDRSS).

3.1.2.3.3 A checkout procedure shall be accomplished before and after every ORU remove and replace operation. During periods when no operations are performed on the CCA, a checkout procedure shall be executed once every seven days. The data shall be relayed to the ground via TDRSS.

3.1.2.3.4 Perform space-based maintenance/servicing operation to remove and replace ORU's by EVA and IVA.

3.1.2.3.5 Refill GHe bottle through CISS interface at Space Station.

3.1.2.4 **Configuration**

3.1.2.4.1 Checkout, maintenance and servicing shall be performed on the mated CCA while resident in the berthing fixture.

3.1.2.4.2 Umbilicals will connect the CCA data communication, electrical power, and helium interface to the Space Station through the interface panel at outside of hangar rear wall.

3.1.2.4.3 The IVA activity will utilize the hangar telerobotic arms (TRA's).

3.1.2.5 **Special Instrumentation Requirements**

Video, voice, TDRSS and CCLS.

3.1.2.6 **Checkout, Maintenance and Servicing Sequence**

3.1.2.6.1 Perform checkout procedure immediately after berthing operation.

3.1.2.6.2 Perform checkout procedure immediately before avionics ORU remove and replace operation.

3.1.2.6.3 Remove and replace avionics Flight Control Processor ORU by EVA.

3.1.2.6.4 Perform checkout procedure.

3.1.2.6.5 Remove and replace avionics Flight Control Processor ORU using telerobotic arm.

3.1.2.6.6 Perform checkout procedure.

3.1.2.6.7 Remove and replace Battery by EVA.

3.1.2.6.8 Perform checkout procedure.

3.1.2.6.9 Remove and replace battery using telerobotic arm.

3.1.2.6.10 Perform checkout procedure.

3.1.2.6.11 Remove and replace one CISS helium pressure bottle ORU by EVA.

3.1.2.6.12 Perform checkout procedure.

3.1.2.6.13 Remove and replace one CISS helium pressure bottle ORU by telerobotic arm.

3.1.2.6.14 Perform checkout procedure.

3.1.2.6.15 This concludes checkout aspect of Accommodations TDM. Perform continuous monitoring with complete checkout procedure every seven days during CCA storage.

3.1.2.7 **Remarks**

A checkout procedure shall be performed once every seven days during periods of storage.

3.1.3 **Payload Integration**

3.1.3.1 **Summary**

The forward end of the Centaur will be fitted with an OTV universal payload adapter to facilitate the payload integration aspect of the Accommodations TDM and to fulfill a payload deployment mission. One TDRS-class and four GPS-class dummy payloads will be brought to the Space Station with STS and utilized, along with Centaur, to gain experience and develop procedures for mating payloads to the OTV. The dummy payloads shall be placed in storage at the Spacecraft Processing Facility (SPF) upon arrival at the Space Station. At least one dummy payload will have sufficient instrumentation to verify interface connections during the payload integration operations. Low-g handling and maneuverability will be experienced with dummy payloads to develop technology and procedures to integrate universal OTV-payloads to launch vehicle upper stages.

3.1.3.2 **Objectives**

3.1.3.2.1 Demonstrate that single or multiple payloads can be mated to OTV spacecraft payload adapters in the low-g environment at the Space Station.

3.1.3.2.2 Develop the technology and operational requirements for a common payload adapter for use on OTV.

3.1.3.2.3 Gain risk free experience with dummy payloads while detecting payload integration difficulties in the low-g environment at the Space Station.

3.1.3.3 **Requirements**

3.1.3.3.1 Perform payload integration aspect of Accommodations TDM in the low-g environment at the Space Station.

3.1.3.3.2 Perform single payload integration using dummy TDRS-class payload with the universal payload adapter (UPA) and perform multiple payload integration using 4 dummy GPS-class payloads with UPA and the multiple payload adapter (MPA).

3.1.3.3.3 Perform IVA payload integration maneuvers by using telerobitic arm.

3.1.3.3.4 Verify the status of mechanical and electrical interfaces using one of the instrumented dummy payloads.

3.1.3.4 **Configuration**

3.1.3.4.1 The Centaur payload interface shall be modified to accommodate the universal payload adapter (UPA) for OTV payload compatability.

3.1.3.4.2 The multiple payload adapter and dummy payloads shall be designed for compatability with the UPA interface.

3.1.3.5 **Special Instrumentation Requirements**

Video, voice, TDRSS and TBD.

3.1.3.6 **Payload Integration/Mating Sequence**

3.1.3.6.1 **Single Payload-Dummy TDRS Class**

3.1.3.6.1.1 Remove dummy TDRS payload from the SPF and install the UPA.. Transport dummy payload/UPA assembly to vehicle.payload interface. Install payload assembly on vehicle UPA and mate electrically and mechanically.

3.1.3.6.1.2 Conduct payload interface checkout.

3.1.3.6.1.3 Demate electrical/mechanical interfaces. Transport payload/UPA assembly to storage facility. Demate the UPA and secure dummy TDRS class payload and UPA in the SPF.

3.1.3.6.2 **Multiple Payloads-Four Dummy GPS Class**

3.1.3.6.2.1 Set up multiple payload adapter (MPA) in fixture for payload mounting. Attach the appropriate number of UPA's at locations on the MPA.

3.1.3.6.2.2 Remove one GPS class dummy payload from SPF storage and mate to UPA/MPA Connect mechanical and electrical interfaces.

3.1.3.6.2.3 Repeat 3.1.2.6.2.2 for remaining three dummy GPS class payloads.

3.1.3.6.2.4 Using the Space Station MRMS, transport the loaded MPA assembly to the Centaur Hangar and transfer assembly to hangar TRA's. Align and mate the MPA to the Centaur payload mount and connect mechanical and electrical interfaces.

3.1.3.6.2.5 Perform payload interface checkout.

3.1.3.6.2.6 Using Centaur Hangar TRA, disconnect and remove the loaded MPA from the CCA, transfer to Space Station MRMS, and return to the SPF.

3.1.3.6.2.7 For each dummy payload/UPA demate mechanical interfaces and electrical umbilicals and return payloads to SPF storage. Disconnect UPA's from MPA and secure both MPA and UPA's in the SPF.

3.1.3.6.3 **Integrate Mission Payload** (TBD joint activity after propellant transfer TDM).

3.1.3.7 **Remarks**

The checkout performed during dummy payload experiments is limited to the payload interface.

3.2 Operations TDM

3.2.1 Cryogenic Propellant Resupply

3.2.1.1 Summary

The cryogenic propellant resupply aspect of the Operations TDM will utilize a separate unmanned Co-Orbiting Platform (COP) to perform tanking and de-tanking in the low-g environment of space. A small scale technology demonstration for storing propellants in the low-g environment of space will have already been accomplished by COLDSAT. The COP will demonstrate propellant storage and transfer on a full scale upper stage vehicle at a location remote from the Space Station.

STS will deliver the COP core to the Space Station where solar panels will be installed and final checkout will occur. The COP core will be placed in final orbit by the OMV. Titan IV launch vehicles will deliver the LH_2 and LO_2 COP segments to orbit fully tanked. The OMV and the COP MRMS will be utilized to assemble COP components.

3.2.1.2 Objectives

3.2.1.2.1 Demonstrate cryogenic propellant storage and transfer can be accomplished in the low-g environment of space.

3.2.1.2.2 Gain experience in performing mating of large zero-leak disconnects, tank chilldown, no-vent fill, and draining of vehicle back into propellant tank.

3.2.1.2.3 Develop cryogenic propellant handling procedures for use in space based OTV turn-around operations.

3.2.1.3 Requirements

3.2.1.3.1 Mate/de-mate large zero-leak disconnects. Perform low-g chilldown, no-vent fill, and low-g tank drain.

3.2.1.4 Configuration

The CCA/payload assembly will be mated to the OMV for transport to the COP. Upon arrival at the COP, the COP MRMS will grapple the CCA. The OMV will demate and return to the Space Station. The remainder of the propellant operations will be performed without the OMV, utilizing the CCA grappling fixtures and the COP MRMS. The COP will have an interface panel compatable with the CCA fluid disconnect panel. The COP shall also have a Computer Controlled Launch Set (CCLS) for automated fluids control, to monitor CCA, and react to telemetry.

3.2.1.5 Special Instrumentation Requirements

CCLS, hardline, video, voice, TDRSS and TBD.

3.2.1.6 Cryogenic Tanking Sequence

3.2.1.6.1 Prepare CCA for transport to COP by attaching hangar TRA to CCA and disconnect the aft support panel from the CCA.

3.2.1.6.2 Open rear hangar door. Using Space Station MRMS, remove the OMV from SPF storage and mate with the aft end of the CCA.

3.2.1.6.3 Perform final checkout procedure at hangar.

3.2.1.6.4 Disengage and retract hangar CCA support structure. Using the TRA move the OMV/CCA asembly out of hangar to release position. Release the CCA grappling fixtures. Clear space station using OMV cold gas thrusters.

3.2.1.6.5 Using the OMV RCS propulsion system, transport the CCA to the vicinity of the COP MRMS. Use cold gas thrusters in the vicinity of the COP.

3.2.1.6.6 Grapple the OMV/CCA assembly with the COP MRMS. Disengage the OMV from the CCA. Using the COP MRMS, mate the CCA to the COP while the OMV returns to the Space Station.

3.2.1.6.7 Perform three propellant transfers per the following sequence. Two propellant transfers are to occur with warm Centaur tanks and one while the tanks are still chilled.

3.2.1.6.7.1 Perform LO_2 tanking operation.

3.2.1.6.7.2 Perform LH_2 tanking operation

2.2.1.6.7.3 After tanking operations, lock-up fill and drain valves. Using COP MRMS seperate the CCA from the COP and leak check disconnects, then re-connect CCA to COP.

3.2.1.6.7.4 Perform LH_2 de-tanking operation.

3.2.1.6.7.5 Perform LO_2 de-tanking operations.

3.2.1.6.8 Safe all COP/CCA interfaces.

3.2.1.6.9 Transport the CCA back to the Space Station using the OMV.

3.2.1.6.10 Berth and store the CCA utilizing experience gained from the berthing operation performed in the Accommodations TDM.

3.2.1.6.11 Perform a checkout procedure on the CCA once every seven days during periods of storage.

3.2.1.7 **Remarks**

The capability to jettison the Centaur will be maintained to minimize COP risk in case of uncontrollable events on the vehicle.

CCLS override capability shall be maintained by the ground.

3.2.2 <u>Launch Deployment</u>

3.2.2.1 **Summary**

After cryogenic tanking operations, the CCLS will perform a final checkout of the CCA/payload assembly. Upon satisfactory completion of the final checkout, Centaur internal power will be activated. Seconds later, the super-zip separation system on the CISS will fire and the Centaur will seperate from the CISS at approximately 0.5 m/s. After sufficient time has passed to allow appropriate separation distance, the Centaur FCP will issue commands to the Centaur for main engine start (MES). The payload will be deployed according to mission profile.

3.2.2.2 **Objectives**

3.2.2.2.1 Gain experience and refine procedures for performing launch operations from a space-based platform.

3.2.2.2.2 Demonstrate satisfactory results from the previous TDM operations.

3.2.2.2.3 Launch a payload to increase the cost effectiveness of the TDM program.

3.2.2.3 **Requirements**

Space Station and ground shall have telemetry coverage. Abort shall initiated by Space Station CCLS, ground CCLS, or COP CCLS.

3.2.2.4 **Configuration**

No special equipment required beyond that required for previous TDM's.

3.2.2.5 **Special Instrumentation Requirements**

Hardline, video, voice, TDRSS, S-band and KU-band operational capability.

3.2.2.6 **Launch Sequence**

3.2.2.6.1 Mate payload to SBTC payload adapter utilizing experience gained while performing payload integration Accommodations TDM.

3.2.2.6.2 Transport and attach the CCA/payload assembly to the COP using the experience gained in the cryogenic resupply Operations TDM.

3.2.2.6.3 Perform cryogenic tanking procedure.

3.2.2.6.4 Perform final CCA/payload checkout procedure.

3.2.2.6.5 Analyze data at Space Station and Ground Stations. Return go/no-go status to Space Station via MCCH.

3.2.2.6.6 CCLS deployment sequence shall be initiated from Space Station. on MCCH command.

3.2.2.6.7 Monitor deployment sequence.

3.2.2.6.7.1 Verify Centaur switchover to internal power.

3.2.2.6.7.2 Verify super-zip system fires.

3.2.2.6.7.3 Monitor telemetry during coast.

3.2.2.6.7.4 Verify RCS system activation.

3.2.2.6.7.5 Verify engines enabled.

3.2.2.6.7.6 Continuously monitor telemetry during remainder of mission.

3.2.2.7 **Remarks**

The CISS rotation feature is not required for this mission and shall be suppressed during ground preparation before deployment by STS.

The Space Station shall provide radar ranging information until the SBTC is out of range.

4.0 **Division of Responsibility**

Item	Organization
CCA	GDSSD/CPOCC
Delivery of hardware to Space Station	NASA (STS or STS-C)
Delivery of propellants	NASA (Titan IV or STS-C)
Dummy payloads	GDSSD
Payload	Payload Vendor/POCC
TDM support hardware	GDSSD
Telemetry, voice, video	SS/WSGS

5.0 **Data Requirements**

Record all telemetry, video, and voice communications.

6.0 **Analysis**

Analysis shall be completed within 180 days of completion of TDM's.

7.0 **Analysis Report.**

The analysis report shall be completed within 270 days of completion of TDM's.

GENERAL DYNAMICS
SPACE SYSTEMS DIVISION

Technology Demonstration Missions

Test Plan Outline

For

OTV Servicing Missions Demonstration

February 22, 1988
838-0-88-064

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ACRONYM TABLE

ACS	Attitude Control System
GDSSD	Genaral Dynamics Space Systems Division
EVA	Extra Vehicular Activity
G	Gravity
GPS	Global Positioning System
I/R	Install and Replace
LAD	Liquid Aquisition Device
LH_2	Liquid Hydrogen
LTCSF	Long Term Cryogenic Storage Facility
MRMS	Mobile Remote Manipulator System
OMV	Orbital Maneuvering Vehicle
OTV	Orbital Transfer Vehicle
R/R	Remove and Replace
TBD	To Be Determined
TDM	Technology Demonstration Mission
TDRS	Tracking Data Relay Satellite
TDRSS	Tracking Data Relay Satellite System
TVS	Thermodynamic Vent System

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1.0 INTRODUCTION

Technology demonstration missions (TDM's) are experiments at, or in the vicinity of the Space Station. Their purpose is to test and verify Space Station accommodations and operation concepts for the orbital transfer vehicle (OTV).

2.0 FOREWORD

Technology demonstration missions (TDM's) are performed to gain experience and establish procedures utilizing the Space Station for space-based Orbital Transfer Vehicle (OTV) missions. Demonstrating new technologies used for OTV turnaround operations will increase saftey and confidence of final designs and minimize risk to the Space Station. A simulated OTV shall be deployed in the space shuttle orbiter and delivered to the Space Station where the TDM's will occur.

Four TDM's have been identified to verify equipment, control algorithms, hardware, and life support systems required for operation in the space environment before full commitment to Space Station operations.

Four TDM's shall be performed to verify satisfactory implementation of new technologies. The four TDM's are as follows:

a. Docking and Berthing
b. Maintenance and Servicing
c. Payload Mating / Interface
d. Cryogenic Propellant Transfer, Storage, and Reliquefaction

Data and experience gained from TDM's will be used as input for final design of equipment and definition of operations used at the Space Station.

3.0 TECHNOLOGY DEMONSTRATION MISSIONS

3.1 **Docking and Berthing TDM**

3.1.1 **Summary**

Docking and berthing TDM's will be performed using a simulated OTV and berthing fixture. The simulated OTV consists of a core open box truss with aerobrake, attitude control system (ACS), avionics, docking/payload attachment adapter, engine, and tank modules attatched to the core.

3.1.2 **Objectives**

3.1.2.1 Obtain data base to be used as input for final designs.

3.1.2.2 To demonstrate the OTV can perform docking and berthing maneuvers in a low-g environment with minimum risk to the Space Station.

3.1.2.3 Practice and gain experience performing docking and berthing maneuvers while performing turnaround operations at the Space Station.

3.1.2.4 Establish adequate procedures for performing OTV docking and berthing maneuvers at the Space Station.

3.1.3 **Requirements**

Accomplish docking and berthing maueuvers in the low-g environment at the Space Station.

3.1.3.1 **Test Conditions**

The test will be conducted in the low-g environment at the Space Station.

3.1.3.2 **Required Data**

Acceleration, velocity and position transducers, video and voice.

3.1.4 **Configuration**

The OTV as described in summary shall be mounted in berthing fixture attached to Space Station

3.1.4.1 **Number of Specimens**

One.

3.1.4.2 **Description of Specimens**

The Demonstration specimen consist of the following items:

a. Fixed truss frame (stays with shuttle)
b. Deployable truss frames
c. EVA manipulator
d. Motorized carriage
e. Berthing / support system
f. Simulated OTV
g. Truss frames berthing systems
h. Electrical and instrumentation
i. OMV (available at Space Station)
j. Space Station MRMS
k. Control Station in Space Station

3.1.5 Special Instrumentation Requirements

Cameras from teleoperation, TDRSS, and TBD.

3.1.6 Demonstration Sequence

a. Transfer simulated OTV from docked space shuttle orbiter to Space Station.

b. Mate OMV with simulated OTV.

c. Remove mated OMV/OTV from berthing fixture.

d. Release OMV/simulated OTV assembly.

e. OMV performs free flight manuvers while mated to OTV.

f. OMV brings the simulated OTV back to the Space Station and performs docking maneuver.

g. OMV/simulated OTV assembly will be berthed.

h. OMV is placed in storage.

3.1.7 Remarks

Docked is defined as the simulated OTV being in proximity close enough to be engaged by the Space Station MRMS.

Berthed is defined as securing the simulated OTV in the berthing fixture.

3.2 **Maintenance and servicing TDM**

3.2.1 **Summary**

Maintenance and servicing TDM's will be performed using a berthed simulated OTV. Remove and replace operations will be conducted on the ACS, engine, avionics, tank, and aerobrake modules. Maintenance and servicing operations will be performed using both EVA and teleoperation.

3.2.2 **Objectives**

3.2.2.1 Obtain data base to be used as input for final designs.

3.2.2.2 Demonstrate that maintenance and servicing operations can be performed in a low-g environment by teleoperation as the primary means and by EVA for backup.

3.2.2.3 To practice and gain experience performing maintenance and servicing operations in a low-g environment while conducting OTV turnaround operations at the Space Station.

3.2.2.4 Establish adequate procedures for conducting maintenance and servicing operations at the Space Station.

3.2.3 **Requirements**

Perform the maintenance and servicing in the low-g environment at the Space Station.

3.2.3.1 **Test Conditions**

The demonstration will be conducted in the low-g environment at the Space Station. Operations shall take place when the simulated OTV is secured to the berthing fixture. The demonstration will be performed both EVA and by teleoperator.

3.2.3.2 **Required Data**

Video, voice and applicable instrumentation recording.

3.2.4 **Configuration**

The simulated OTV will reside in the berthing fixture during maintenance and servicing operations.

3.2.4.1 **Number of Specimens**

Five

3.2.4.2 **Discription of Specimens**

ACS

Aerobrake

Avionics Modules

Engine

Tank

3.2.5 **Special Instrumentation**

Cameras from teleoperation, TDRSS.

3.2.6 **Demonstration Sequence**

The test shall be conducted in compliance with Table I in accordance to the following procedure.

a. Berth OTV and adjust berthing fixture for accessibility to required component.

b. Remove component.

c. Transport component to holding fixture and attach to fixture.

d. Remove component from holding fixture and return to OTV.

e. Re-install component onto OTV.

f. Align and retract berthing fixture.

Table I

OPERATIONS	NO.	EVA	TELEOPERATION	COMPONENT
R/R[1]	3	X		AEROBRAKE
R/R	3		X	AEROBRAKE
R/R	3	X		ACS
R/R	3		X	ACS
R/R	3	X		AVIONICS MODULES
R/R	3		X	AVIONICS MODULES
R/R	3	X		ENGINE
R/R	3		X	ENGINE
R/R	3	X		TANK
R/R	3		X	TANK

1. R/R-Remove and replace.

3.2.7 Remarks

The actual avionics modules for remove and replace operations will probably consist of a battery, fuel cell or guidance set.

3.3.0 **Payload Mating/Interface TDM**

3.3.1 **Summary**

The payload mating and interface TDM will be accomplished using two dummy payloads with universal interfaces. The payloads will have been fit-checked on the ground prior to performing this TDM. Payload matiing and interfacing will be accomplished using both EVA and teleoperator.

3.3.2 **TDM Objectives**

3.3.2.1 To obtain data base to be used as input for final design.

3.3.2.2 To demonstrate payload mating can be performed in the low-g environment at the Space Station by both EVA and teleoperation.

3.3.2.3 To practice and gain experience installing payloads on the OTV during OTV turnaround operations.

3.3.2.4 Establish adequate procedures for conducting payload mating/demating operations.

3.3.3 **Requirements**

Perform payload mating operations in the low-g environment at the Space Station.

3.3.3.1 **Test Conditions**

The demonstration will be conducted at the Space Station in the low-g environment of space. The operation will be performed with the simulated OTV rotated 90° in the berthing carriage. The task will be performed by both EVA and teleoperation.

3.3.3.2 **Required Data**

Voice, video, payload interface instrumentation.

3.3.4 **Configuration**

The simulated OTV will remain in the berthing fixture rotated 90° during the payload mating TDM.

3.3.4.1 **Number of Specimens**

Two different dummy payloads will be utilized for this TDM.

3.3.4.2 **Description of Specimens**

A dummy tracking data relay satellite (TDRS) and global positioning system (GPS) payload will be used to demonstrate the payload mating and interface TDM.

3.3.5 **Special Instrumentation Requirements**

Cameras, TDRSS, payload interface checkout kit.

3.3.6 **Demonstration Sequence**

The test shall be conducted in compliance with Table II in accordance with the following procedure.

a. Translate the berthed simulated OTV to payload mating orientation using the berthing fixture.

b. Remove payload from storage facility and transport to simulated OTV.

c. Mechanically install payload to OTV interfaces and connect umbilicals.

d. Perform payload checkout.

e. Disconnect and remove payload.

f. Transport to storage fixture and store.

Table II

Payload Integration TDM Operations

OPERATIONS	NO.	EVA	TELEOPERATION	COMPONENT
I/R [2]	3	X		TDRS
I/R	3		X	TDRS
I/R	3	X		GPS
I/R	3		X	GPS

2. I/R-Install and remove.

3.3.7 **Remarks**

Both dummy payloads will be sufficiently instrumented to return an interface connection pass/fail status to the Space Station.

3.4 Cryogenic Propellant Transfer, Storage, and Reliquefaction TDM

3.4.1 Summary

Cryogenic propellant transfer, storage and reliquefaction TDM will be performed at the Space Station. The LH_2 receiver tank will be delivered by the space shuttle orbiter and secured to the Space Station. A full LH_2 supply depot tank will be deployed on a Titan III launch vehicle. The OMV will bring the LH_2 supply tank to the Space Station where it will be secured and the appropiate lines connected.

3.4.2 Objectives

3.4.2.1 Obtain a data base to be used as input for final design of a LTCSF.

3.4.2.2 Establish and confirm which parameters correlate with analysis and data from scale models.

3.4.2.3 Practice, gain experience, and develope procedures for performing low-g cryogenic tanking and detanking, mass gaging, boiloff reliquefaction, and long duration storage operations required for OTV turnaround operations.

3.4.2.4 Evaluate the performance of cryogenic tanking/detanking and storage operations in a low-g environment.

3.4.3 Requirements

Perform cryogenic operations in the low-g environment at the Space Station.

3.4.3.1 Test Conditions

The cryogenic transfer experiment will occur in the low -g environment at the Space Station.

3.4.3.2 Data Requirements

Pressure, temperature, flowrates, acceleration, voltage, current, voice, and video.

3.4.4 Configuration

The cryogenic transfer will occur while the depot and receiver tanks are mated and secured to the Space Station.

3.4.6 Demonstration Sequence

The demonstration will be conducted per the following sequence:

a. Passive, low-g cryogenic tank pressure control (TVS).

b. Active, low-g cryogenic tank pressure control (TVS & mixer).

c. Cryogenic tank chilldown in low-g (fluid injected spray).

b. No-vent fill/refill of cryogenic tanks in low-g.

c. Fill of LAD (liquid aquisition device) in low-g.

d. Low-g liquid mass gaging of cryogenic tanks.

e. Cryogenic liquid slosh dynamics and control in a low-g environment.

3.4.7 Remarks

The supply and receiver tanks will be mated IVA utilizing the Space Station MRMS and the OTV.

4.0 Division of Responsibility

Item	Organization
Data Aquisition	WSGS
Delivery of hardware to SS	NASA (STS or STS-C)
Dummy Payload	Payload contractor
OMV	NASA
Operations	NASA (JSC)
Propellants	NASA (Titan III)
Simulated OTV	Simulated OTV contractor

5.0 Data

All data shall be relayed to the ground through the TDRSS. The data station on the ground shall save all data on magnetic tape and produce backup copies. The data station shall strip out appropiate data and deliver two copies to Genaral Dynamics Space Systems Division (GDSSD).

6.0 Data Analysis

Engineering groups shall analyze the data. The data analysis shall be completed 90 days after completion of the individual TDM.

7.0 Final Report

The responsible engineering groups shall release an analysis report within 60 days of analysis.

APPENDIX G
IMPORTANT MISCELLANEOUS DATA

3.2.3. PLATFORMS

The SSP includes platforms that provide exterior space and attach mechanisms, the resource needs (power, thermal control, attitude control and data management), and the operational needs (orbit, g-level, cleanliness, etc.) of the missions allocated to them.

3.2.3.1. CO-ORBITING PLATFORM

This platform(s) will have a 28.5 degree inclination.

3.2.3.2. POLAR PLATFORM

This platform(s) will have a nominal Sun-synchronous polar orbit.

3.2.4. DEPLOYMENT, ASSEMBLY, AND CONSTRUCTION

The SSP will provide support capability for construction, assembly, and deployment which implies providing payload service devices such as manipulators, Manned Maneuvering Unit (MMU)'s, Extravehicular Activity (EVA) capability, and standard tool kits. The manned element will have facilities to support the assembly and disassembly of large structures including attachment provisions, a storage area for components, a remote manipulator system, and an orbital maneuvering system. Power, thermal, and data system interfaces will be available to the payload undergoing assembly or disassembly. The platforms will also be designed to facilitate on-orbit assembly and disassembly.

3.2.5. PAYLOAD CHECKOUT, INTEGRATION, AND DEPLOYMENT

All classes of payloads and satellites requiring transfer to other orbits may be brought to the station by the NSTS or other vehicles and then integrated with a transfer stage, checked out and launched. The SSP will provide the facilities to checkout payloads and satellites after receipt from the NSTS and provide the necessary launch support prior to deployment. These stages could be either expendable or reusable. Reusable transfer stages will be based, serviced, maintained and refueled at the station. Expendable stages will be stored and serviced. The growth station will also provide the capability for payload deployment to high energy orbits.

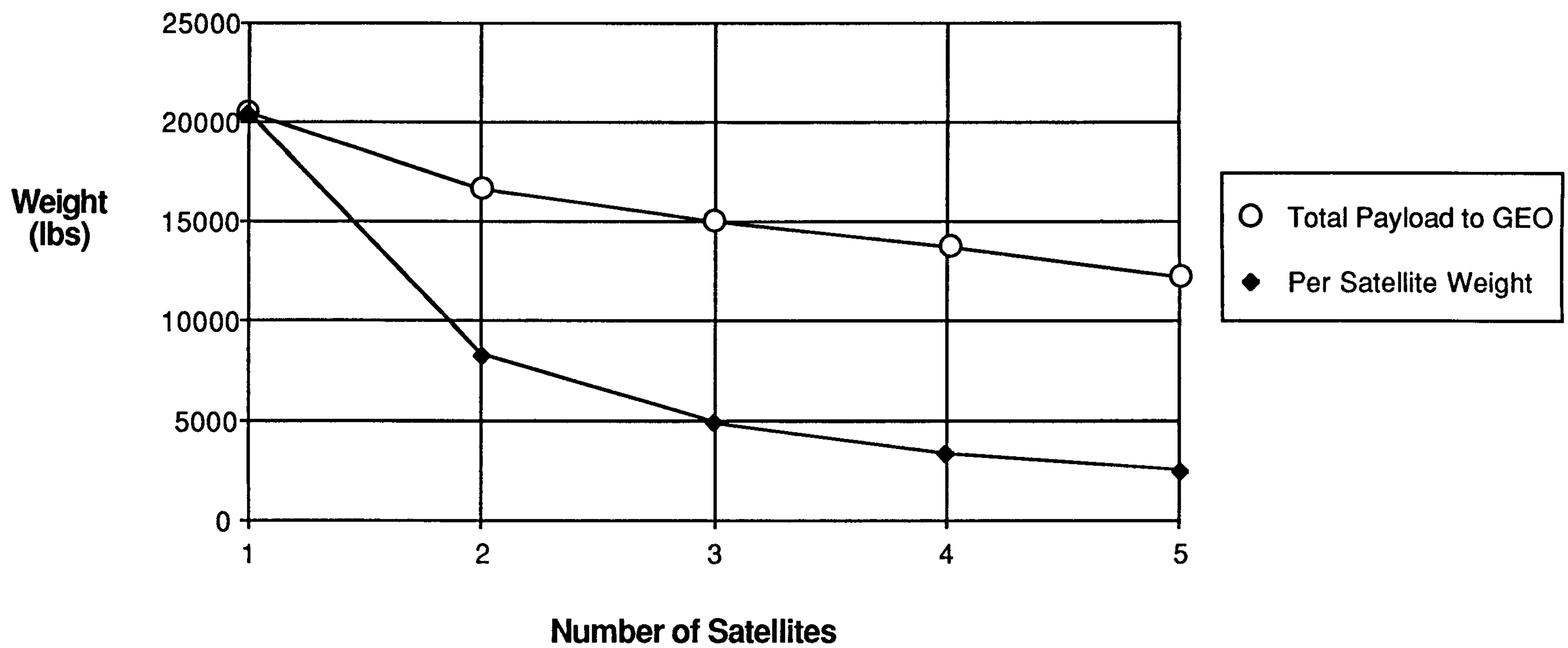
3.2.6. REMOTE MAINTENANCE, SERVICING, CHECKOUT, AND RETRIEVAL

Payloads, satellites and platforms remote from the manned element will be maintained, serviced and checked out via an unmanned Orbital Maneuvering Vehicle (OMV), Orbital Transfer Vehicle (OTV) or the NSTS. Servicing of a payload can be at its location, or the payload could be retrieved, serviced at the station and returned. The design will facilitate scheduled and unscheduled maintenance and servicing of modules, attached instruments, platforms and free-flyers. Servicing may be performed either in situ (unmanned by the OMV or manned by the NSTS) or at the station (manned). The tasks for unmanned in situ servicing will be limited by the capabilities of the OMV and by the design of the satellite or instruments to be serviced. The definition of these capabilities will be provided by NASA. The growth station will have the capability for in situ servicing at geosynchronous orbit.

3.2.7. PAYLOAD STAGING FOR EARTH RETURN

Payloads, experimental samples and captured samples requiring return to Earth will be demated, prepared and stored either pressurized or unpressurized until

MULTIPLE PAYLOAD GEO DELIVERY CAPABILITY



COST ESTIMATES FOR OTV TECHNOLOGY DEMONSTRATION MISSIONS AT SPACE STATION SIMILAR TO CSOD

TDM DESCRIPTION	ESTIMATED COST 87 M$
Berthing	48.2
Maintenance & Servicing	53.4
Payload Mating/Integration	8.0
Cryogenic Resupply	391.6
Delivery & ASE	102.2

Notes:

1) No cost assumed for STS service since these are NASA missions
2) Cryogenic Resupply may require platform not included in estimate
3) No test of multiple payload integration
4) No hangar cost included in Berthing TDM

APPENDIX H
TURNAROUND OPERATIONS ANALYSIS FOR OTV

TURNAROUND OPERATIONS ANALYSIS FOR OTV

FINAL REVIEW MEETING
AT NASA-MSFC
DECEMBER 9, 1987

NAS 8-36924
DR-3

GENERAL DYNAMICS
SPACE SYSTEMS DIVISION

COMPARISON OF OTV ON-ORBIT PROPELLANT DEPOT DEVELOPMENT OPTIONS - COST ($M)

	4/10 SCALE TEST		LTCSF				
OPTION #1	324	+	446	≅	770		
	1/10 SCALE TEST		GR TEST (LH2 ONLY)		LTCSF		
OPTION #2	256	+	61	+	446	≅	763
	1/10 SCALE TEST		TDM (LH2 ONLY)		REMAINING LTCSF DEV		
OPTION #3	256	+	356	+	228	≅	840

11/87

OTV PROPELLANT DEVELOPMENT OPTIONS
ROM COSTS

		1/10 SCALE TEST	4/10 SCALE TEST	PROTOFLIGHT GR TESTS (LH2 ONLY)	TDM* (LH2 ONLY)	DEVELOP OF* LTCSF AFTER TDM	LTCSF* (LH2 & LO2)
COST (87M$)	NR	159	181	61	296	190	371
	R	37	45	--	60	38	75
	ELV	60	98	--	--	--	--
	OPS	--	--	--	--	--	--
		256	324	61	356	228	446

*SHUTTLE LAUNCH COSTS EXCLUDED

TECHNOLOGY DEVELOPMENT PLAN FUNDING*

AREA \ FY	88	89	90	91	92	93	94	95	96	97	TOTAL $M
• AUTOMATED FAULT DETECTION/ISOLATION AND SYSTEM CHECKOUT (GR)	.2	1.0	1.0	3.0	4.0	.5	.3				10.0
• CRYOGENIC PROPELLANT TRANSFER	SEE PROPELLANT DEVELOPMENT OPTIONS COST CHART										
• MAINTENANCE FACILITIES/ AND EQUIPMENT (GR & SORTIE)	.5	1.0	6.0	5.7	2.0						15.2 ✓
• EVA OPERATIONS (GR & SORTIE)	.5	4.2	4.0	2.0							10.7 ✓
• MAINTENANCE/SERVICING OPERATIONS & SUPPORT EQUIPMENT (TDM)				2.0	12.0	15.5	14.0	5.0			48.5 x
• DOCKING & BERTHING (GR & SORTIE)	1.0	2.0	6.9	6.0	2.0						17.9 ✓
• OTV P/L MATING (GR & SORTIE)	.5	2.0	2.0	1.8	1.0						7.3 0

*DOESN'T INCLUDE LAUNCH OR OPERATIONS COSTS

11/87

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National Aeronautics and Space Administration

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16. Abstract

A study was conducted to expand on the results of an initial study reported in NASA CR-179593, "Centaur Operations at the Space Station". The previous study developed Technology Demonstration Missions (TDMs) that utilized the Centaur G-prime upper stage to advance OTV technologies required for accommodations and operations at the Space Station (SS). This study performs an initial evaluation of the cost to NASA for TDM implementation. Due to the potential for commercial communication satellite operation utilizing the TDM hardware, an evaluation of the Centaur's transportation potential was also performed.

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