OLYMPUS RECONFIGURATION FEASIBILITY STUDY

FINAL REPORT

REV 1

RML-009-86-102

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FINAL REPORT

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SPAR

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OLYMPUS RECONFIGURATION STUDY

FINAL REPORT

INTRODUCTION

This report summarizes the work performed under DOC Contract 36100-6-4274, which was a study of means of reconfiguring the basic Olympus spacecraft in order to increase its competitiveness with existing large communications spacecraft.

The contents of the report consist of the following sections:-

A. Olympus reconfiguration Study

B. Launch Vehicles for modified Olympus

C. Adaptation of Olympus for low-end payloads.

D. Solar Array for modified Olympus

E. Impact of proposal Olympus bus on current Olympus MGSE.

F. Viewgraph package of September 26, 1986 presentation to DOC at Spar Montreal

G. Input viewgraphs to PMC presentation at Stevenage, October 13, 1986.

H. Viewgraph package dated November 27, 1986 which was presented to DOC at DOC HQ, Ottawa.

J. Letter report addressing the DOC S.O.W. Annex A Rev 1 items.

The conclusions of this study are summarized in the final viewgraph of Part G. However, certain points should be emphasized. In its present configuration, the Olympus spacecraft is not competitive with such North American spacecraft as the RCA 5000. This study has shown that the Olympus could possibly be made competitive by modifications to the solar array and structure, which could reduce the bus mass by an amount up to 150 kg. Such modifications should be accompanied by an extensive subsystem by subsystem review to uncover additional mass savings.

Finally, it should be emphasized that these conclusions are preliminary, arising as they are from such a short study. What is now required is a more extensive study, involving other members of the Olympus team as well as Spar, in order to confirm and extend these findings.

OLYMPUS MODIFICATION STUDY

SECTION A: OLYMPUS RECONFIGURATION STUDY

Note de Service
Internal Correspondence

SPAR

À / To	K. Duffy	Service / Location		Date	August 27,1986
De / From	A. Kidd	Service / Location	Systems	Poste / Telephone	3644

Objet / Subject: OLYMPUS RECONFIGURATION STUDY

The goal of the Olympus Reconfiguration Study is to examine possible modifications for the Olympus spacecraft which would allow it to compete effectively with existing spacecraft like the RCA 5000. Such spacecraft are capable of supporting payloads at 300 - 400 Kg masses at powers at 2500 - 3500 watts, and are within the maximum dual launch capability of the Ariane IV (2500 Kg).

In order to focus this study, three payload models have been selected, which should be able to be carried on either the RCA 5000 or the modified Olympus. These payloads are the Aussat II spacecraft, carrying the proposed baseline payload with some of the suggested options; the Anik E spacecraft, carrying the current baseline with probable options, and a MSAT dual band spacecraft, carrying a UHF and L-band payload. Tables 1, 2 and 3 describe the payload parameters for each of these models.

Table 4 lists subsystem masses for the Olympus L-1 and the Anik E bus. The Anik E spacecraft has a mass of 2500 Kg in Transfer orbit and the L-1 spacecraft has a transfer orbit mass of 2430 Kg.

The payloads are also similar in mass (320 Kg vrs. 335 Kg). Despite these similar masses, the L-1 has only a 5 year lifetime and provides 2100 watts to the payload (only 850 watts in eclipse). One of the primary reasons for this shortfall is the extra mass in the Olympus bus, which weighs 125 Kg more than the RCA 5000 (5 years of stationkeeping fuel). A comparison with the RCA 500 spacecraft, as proposed for Anik E, shows the areas in which weight savings can be made.

The principal ~~att~~ offenders are the structure and solar array which together weigh 195 Kg more on Olympus than on the RCA 5000. (The power subsystem has less mass on the Olympus, but this is because the L-1 has only 1/3 the battery capability of the RCA 500).

The overall goals for the structure redesign concept during this study should be as follows:

1. Reduce structure mass by 100 Kg

2. Reduce solar array mass by 80 Kg

3. Allow for mounting four 50 amp H Nickel Hydrogen batteries and associated shunts.

4. Reduce the overall length of the spacecraft.

The implications of the above goals are essentially a complete structural/thermal redesign of the olympus, which is outside the scope of this study. However, this study will outline the major possibilities and implications of such a redesign.

L Morgan
for A Kidd

cc: For Info.
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L. Keyes
G. Lewis
H. Moody
G. Booth - DOC Hqtrs.

TABLE 1

AUSSAT II PAYLOAD MODEL

Ku Band Baseline payload 1 - 2.5 meter reflector	25 x 30 watts (32 for 25)	mass 300 Kg power 1890 watts
X-Band Military payload 1 - 80 cm reflector	1•20 watt	mass 20 kg power 60 watts
Mobile L-Band payload 1 - 3.3 meter reflector	1•50 watt	mass 45 kg power 200 watts
DBS payload replace 1 - 2.5 meter reflector with & 1 - 80 cm reflector	2 Ku 30 watt 2 Ku 100 watt	mass delta 40 kg power delta 300 watts
Total Payload	Mass Power Eclipse power	405 Kg 2450 Watts 2100 Watts
Lifetime	10 years plus 2 years in-orbit storage	
Assume 5 reflectors	1 2 2	3.3 meter (L-Band) 2.5 meter (Ku-Band) 80 cm (X-Band & Ku-Band)

TABLE 2

ANIK E PAYLOAD

Ku-Band	18 for 16 50 watt 1-80 inch reflector	Mass 163 Power 2011 watts
C-Band	30 for 24 12 watt SSPA's 1-80 inch reflector	Mass 145 Kg Power 943 watts
CONUS	option 1-24 inch reflector	Mass 25 power 11 watts

Total Payload	Mass Power	333 Kg 2965 Watts

Lifetime 10 years plus 2 years in-orbit storage

TABLE 3

MSAT DUAL BAND PAYLOAD MODE

UHF Payload	Mass	160
1 - 5 meter deployable reflector	Power	1150
L-Band Payload	Mass	160
1 - 5 meter deployable reflector	Power	1000
SHF Backhaul Payload	Mass	30
1 - 80 cm reflector	Power	350

Total Payload	Mass	350 Kg
	Power	2500 watts
	Eclipse Power	1250 watts

Lifetime	10 years

TABLE 4

COMPARISON AT SUBSYSTEM MASSES (KG)

	OLYMPUS L-1	RCA ANIK E	DELTA
TTC	39.5	34.7	+ 5
Power	151.0	165.0	- 14
Solar Array	168.0	81.0	+ 87
ACS	99.0	70.0	+ 29
CPS	118.4	143.7	- 25
Structure	243.0	135.0	+108
Thermal	63.0	43.0	+ 20
Mechanisms	-	40.5	
P/L Harness	13.5	-	
Other Harness	-	56.9	
	895.5	769.9	

SECTION B: LAUNCH VEHICLES FOR MODIFIED OLYMPUS

Note de Service
Internal Correspondence

SPAR

À / To	See Distribution	Service / Location		Date	22 September, 1986
De / From	A. Kidd	Service / Location	Systems	Poste / Telephone	3644

Objet / Subject: LAUNCH VEHICLES FOR MODIFIED OLYMPUS

INTRODUCTION

The Olympus L-1 satellite was designed to be launched as a dedicated payload on an Ariane B launch vehicle. Shuttle compatibility was also a design requirement, but not the major driver for the design.

As a result, the overall L-SAT configuration is tall and relatively narrow. Such a shape does not utilize either STS or Ariane IV resources as efficiently as possible.

For a STS launch, an upper stage is required. At this time the only suitable stage available is the Orbital Science Corporation TOS-S1 upper stage, currently under development for the Mars Orbiter Program. This stage was proposed for use with an Olympus Bus in the M-SAT phase B program. With an unmodified Olympus, the upper stage/spacecraft combination required 40 - 45% of the STS bay.

ELV's

The Challenger accident has renewed activity in the US toward development of commercial ELV's. However, the development of such vehicles has been proceeding in many other countries for a considerable period. As a result, while many ELV's may appear on the market within five years, at the present time Ariane is predominant, with only the Chinese Long March III offering competition at the lower end of the Ariane capability range.

In the 1991-1996 period, the Ariane IV class launcher can be expected to set the standard for launcher compatibility requirements. There may well be other launchers available in this period with similar capabilities, such as the Japanese HII, Martin Marietta Titan IV, the

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General Dynamics Atlas Super G and possibly the Hughes/Boeing Jarvis, it can be expected that these will offer an interface compatible with Ariane IV, so that their launchers would be in competition for existing spacecraft designs. It should be noted that some launchers, like the Titan IV, require the customer to supply an upper stage, similar to that required for an STS launch.

The above discussion gives the background for deciding to baseline Ariane IV as the primary launcher interface design requirement for the modified Olympus spacecraft. Meeting the Ariane IV dual launch interface requirements (3.6 meter diameter, 2500 kg in GT.O.) should insure compatability with all ELV's likely to be used in the future.

Figure 1 shows some envelopes of the Ariane IV and Titan launch vehicles.

STS

The situation in the US concerning the availability of the STS for commercial (and especially foreign commercial) launches is far from clear at the present time. The US administration has stated that the primary use at the STS system will be to support military/scientific payloads and the Space Station with some exceptions for commercial launches with existing shuttle contracts. However, it is not certain how many exceptions will be made, or indeed how consistantly this policy will be followed.

In any event, sizing the reconfigured Olympus spacecraft for a dual Ariane IV launch will also be consistent with an economical STS alunch configuration, since reducing the overall height of the Olympus platform tends to equalize the pay-by-length and pay-by-weight change factors.

The other major consideration for an STS launch is choice of an upper stage. Most North American spacecraft manufacturers have developed or are in the process of developing upper stages which are integral to their own spacecraft. That is to say these upper stages make use of the spacecrafts's Telemetry and Command and Guidance subsystems and therefore need consist of little more than a solid motor and structure. Such a concept gives a considerably more compact and cost effective upper stage than would be the case if the upper stage functioned autonomously from LEO to GTO.

As mentioned earlier, the upper stage most likely to be available to Olympus is the OSC TOS-S1. This is a general purpose upper stage and includes guidance and control. It therefore is not as compact as an integral upper stage but does have a large weight capability. Use of the TOS-S1 in the shuttle (or with the Titan IV) would tend to increase the length of the Olympus/upper stage stack and therefore make a reduction in spacecraft height very useful.

SUMMARY AND RECOMMENDATIONS

The Olympus spacecraft designed for a dedicated Ariane III launch, will find itself in on increasingly uncompetitive position as Ariane IV becomes the standard in sizing launch envelopes. Furthermore, even in a post shuttle era, the need for compatability with upper stages has not gone away, since some launchers, notably the Titan IV, will still require use of an upper stage for GEO missions. The Olympus must use the OSC TOS-S1 upper stage, so there is a considerable advantage in presenting as compact a size as possible, so as to fit into the titan launch envelope.

Even on the Ariane IV, it is very useful to keep spacecraft height to a minimum since payload requiring large fixed antennas will have to stow these antennas on top of the spacecraft, further reducing overhead clearance.

For all the above reasons, a design for Olympus which would make use of the Ariane IV envelope diameter while reducing overall spacecraft length to a minimum is recommended.

Alan M. Kidd

Distribution:

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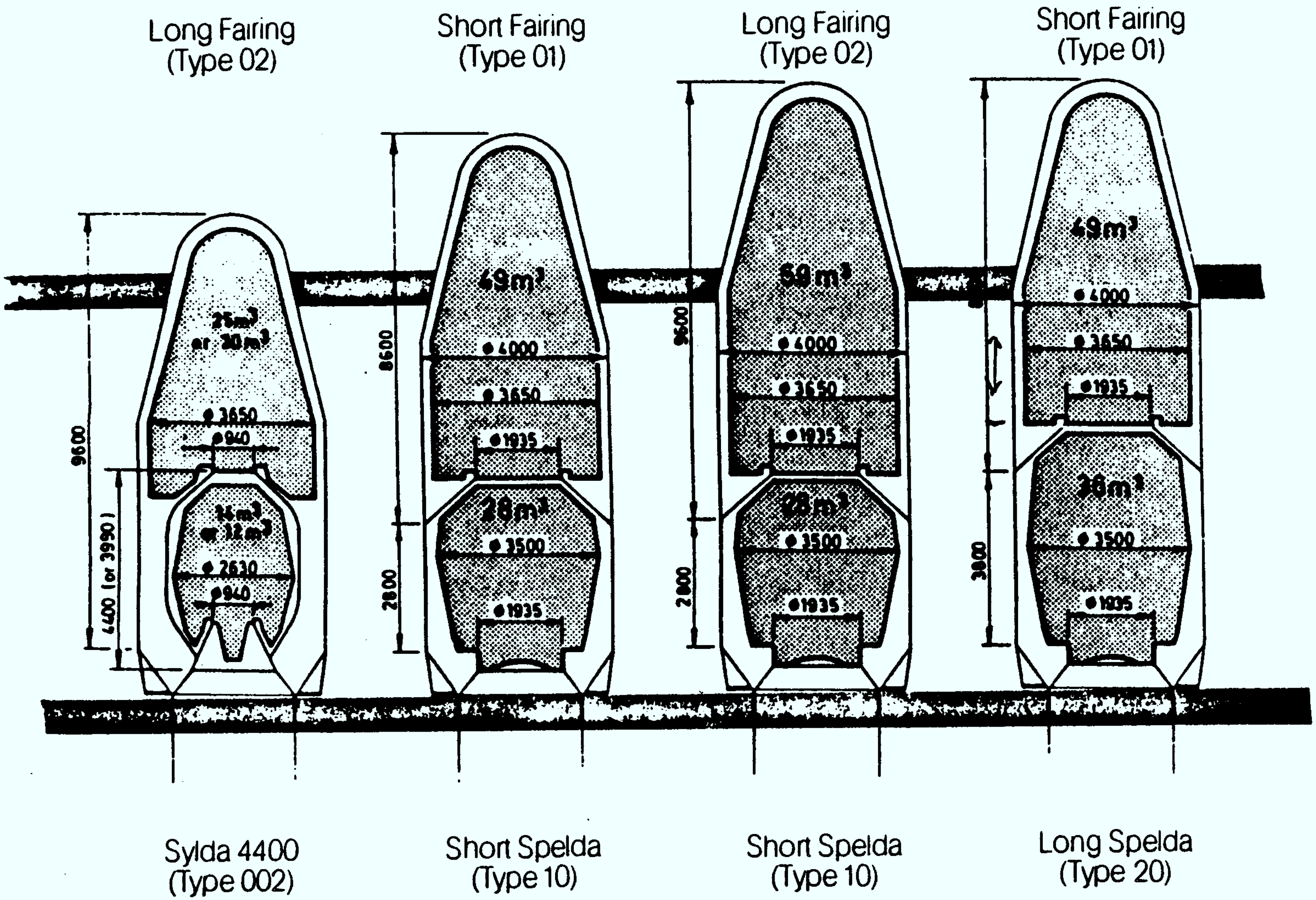
J.G. Lewis
C. Morgan
K. Duffy
L. Keyes

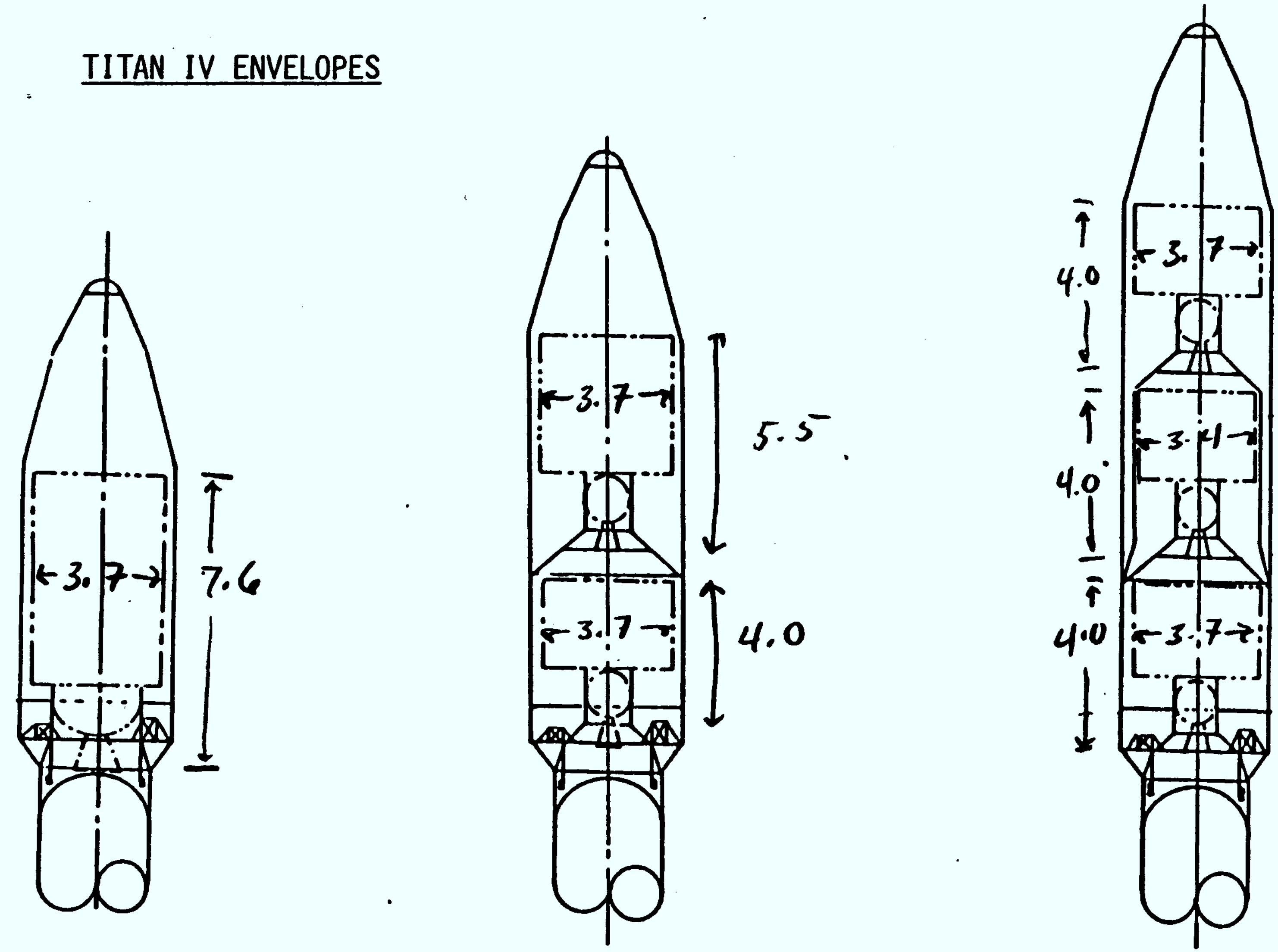
ca

ARIANE IV ENVELOPES

PAYLOAD COMPARTMENT CONFIGURATIONS
FOR DUAL LAUNCHES



TITAN IV ENVELOPES



ONE PAYLOAD TWO PAYLOADS THREE PAYLOADS

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SECTION C: ADAPTATION OF OLYMPUS FOR LOW-END PAYLOADS

Note de Service
Internal Correspondence

SPAR

À / To	A. Kidd	Service / Location	Date Oct.28/86
De / From	K. Duffy	Service / Location Eng.	Poste / Telephone 3175

Objet / Subject: Adaptation of Olympus for Low-End Payloads

Although the Olympus platform has been designed to accommodate payloads up to 600 kg and 7 kW, there is considerable interest in using it to support missions with more modest requirements. In particular, a large number of currently planned payloads fall within the range of 300-400 kg mass and 2500 - 3500 watts primary power. Most such missions are intended for launch on the Ariane IV system. There are several spacecraft buses which are capable of supporting these payloads, one of them being the RCA 5000. A direct comparison shows that this bus outperforms Olympus by nearly 200 kg in this range. The structure and power subsystems are by far the largest contributors to the overall mass difference. For the same payload, the Olympus structure is 243 kg versus 135 kg for the RCA bus. For direct competition, the Olympus mass must be reduced.

A closer look at the spacecraft reveals that there are several inherent aspects in the configuration which conspire to make the structure very heavy for payloads in the 300-400 kg range. The major ones are as follows:

i) Ariane III Envelope

Whereas most missions in the near future are planned for dual launch on the Ariane IV vehicle, Olympus has been sized for a dedicated Ariane III launch. Since the fairing envelope of the former is 25% wider, the Olympus spacecraft suffers the penalty of a longer and thinner structure. This results in much higher induced loads, and hence a much heavier structure.

ii) Payload Range

The Olympus structure has been designed to support payloads up to 600 kg and 7 kW. This makes the platform much larger and heavier than necessary for payloads in the F1 range.

iii) Structural Modularity

The Olympus structure consists of three functionally independent assemblies, the Propulsion, Communications, and Service Modules. Each of these carries a significant portion of the spacecraft primary loads. This design results in cumbersome transitions (eg. struts/shear walls) and requires a large number of heavy bolted interfaces.

It was initially concluded that: a) a drastic reduction in structural mass is necessary for Olympus to be competitive for payloads in its "low-end" range; and b) a reconfiguration of the platform would result in very substantial mass savings; the benefits of such an activity might well outweigh the costs.

Reconfiguration Study

The concept of reconfiguring Olympus was subsequently explored in more depth. The following ground rules were used for the redesign.

i) Ariane IV dual launch, upper position.

ii) Payload of 300-400 kg, requiring 2500-3500 watts.

iii) Retain existing Olympus subsystems and technology.

Based on these parameters, a new configuration for Olympus was developed, as shown in Figure 3. The goal of the new design was to arrive at a structural mass of 100 kg less than Olympus. To that end, the size and shape of the platform were modified to take full advantage of the Ariane IV envelope. The external dimensions were changed from 2.1 x 1.75 x 3.54 meters to 2.9 x 1.9 x 2.2 meters. Essentially, Olympus was "truncated" at what was the solar array drive level, while the lateral (X,Y) dimensions have been increased to fill out the available envelope. The result is a spacecraft which is roughly half the height of Olympus, but which provides only 15% less mounting area on the north/south walls. The efficiency of the structure has been immensely improved, mostly because of the reduction in height above the separation plane. Although it constitutes a radical change to Olympus, this reshaping of the platform is essential for achieving large mass savings.

The reduced height of the spacecraft means that the upper fuel tank can no longer be accommodated in the thrust tube. Therefore, the single spherical tank has been replaced by two smaller, cylindrical tanks embedded in the East/West shear walls. The thrust tube has become a continuous structural member extending from the separation ring to the earth-facing panel. Its construction should be changed from corrugated sheet to honeycomb, in order to simplify and lighten the various structural interfaces. It is supported by four radial shear walls which run the full length of the spacecraft. As an assembly, the thrust tube and shear walls form the propulsion module structure, and also serve as the primary load paths for the spacecraft.

Since it no longer supports the spacecraft loads, the Service Module now merely consists of four north/south equipment panels plus a horizontal floor for AOCS units. The strutwork and thrust tube interfaces have been eliminated, since the panels are fastened directly to the central structure (Propulsion Module). A lower floor has been added for structural support and equipment mounting. Also, in order to conserve mounting area and reduce mass, it is recommended that the Nickel-Hydrogen batteries be mounted on an integral radiator in the north/south bus panels.

The Communications Module is still comprised of the north/south payload panels plus the nadir panel. It is possible to have either single or dual panels on the north or south faces. The panels have also been extended well above the earth-facing panel to provide additional equipment mounting and heat rejection areas.

The RMSD rigid panel solar array has been designed as an efficient alternative to the flexible array for low-to-medium power payloads on Olympus. Up to 6 panels of this array can be accommodated on each of the north and south faces of the reconfigured platform. The solar array drive, as well as the array tie-downs, are fastened directly to the central structure; the shear walls and floors will prevent excessive load transfer from the spacecraft. The solar array tie-down scheme is shown in Figure 4.

A very preliminary mass budget was formulated for the reconfigured Olympus, and is presented in Table 1. A total mass savings of 84kg is projected. Further mass savings could be realized by the use of lightweight structural materials in the design (e.g. GFEC beryllium). In Table 2 is presented a comparison of the structural mass of the RCA 5000, Olympus, and the reconfigured Olympus. It is clear that although the reconfigured Olympus has been slimmed considerably from its forebear, its mass budget is still quite generous in comparison with the RCA bus.

Finally, in order to explore the implications of its modified shape, an example payload was used to perform an accommodation exercize for the reconfigured Olympus. The externally-mounted equipment for the Aussat II payload is as follows:

- One 2.0 m dual-polarized Ku-band reflector.
- One 1.5 m dual-polarized Ku-band reflector.
- One 0.8 m X-band reflector.
- One or two 3.0 m solid L-Band reflectors

  or

  Two 5.0 m deployable mesh UHF reflectors.

The results of the work are presented in Figures 5 through 10. It was found that the reshaped platform was excellent for stowing the large groups of reflectors, primarily because of its good "headroom". The low height of the spacecraft leaves a large volume under the Ariane IV fairing which is available for stowing stackes of large-diameter solid reflectors, or the bulky packages of the deployable mesh-type reflectors.

Summary and Conclusions

The study concluded that in order to make Olympus competitive at the lower end of its payload range, severe mass reductions are required. It was found that if a modified spacecraft configuration were adapted for the smaller missions, a great deal of structural mass could be saved. In addition, the new design would be easier to analyze, assemble, and test. The adapted platform would be a derivative of the Olympus bus, but would not supercede it; rather it would be complementary, forming part of an Olympus product line. If the costs of the rework can be minimized (by limiting testing, for example), a reconfigured Olympus could prove profitable.

K.Duffy

cc: V. Jha
M. Donato

Table 1

STRUCTURE MODIFICATION

K.Duffy

RECONFIGURED OLYMPUS:

STRUCTURAL MASS ESTIMATES

THRUST TUBE & SEPARATION RING	35	
E/W SHEAR WALLS	10	
N/S SHEAR WALLS	5	
N/S PAYLOAD PANELS	30	
N/S BUS PANELS	15	
UPPER FLOOR (CM)	12	
MIDDLE FLOOR (SM)	10	
LOWER FLOOR (SM)	8	
CPS MOTOR FLOOR	8	
ACCESS PANELS	6	
MISCELLANEOUS STRUCTURAL EQUIPMENT	20	
TOTAL	159	KG
PREVIOUS OLYMPUS STRUCTURE MASS	243	KG
SAVINGS	84	KG

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Table 2

STRUCTURE MODIFICATION

K.Duffy

STRUCTURAL MASS COMPARISON

RCA 5000		OLYMPUS		RECONFIGURED OLYMPUS	
CENTER STRUCTURE + BULKHEADS + TANK SUPPORT	43.1	CYLINDERS + MOTOR FLOOR + E/W STRUCTURE	95.8	THRUST TUBE + SEPARATION RING + SHEAR WALLS	58
PAYLOAD PANELS	34.8	PAYLOAD PANELS	57.3	PAYLOAD PANELS	42
BUS PANELS	20.8	SERVICE MODULE	57.9	BUS PANELS	33
OTHER	36.6	OTHER	32.0	OTHER	26
ACCESS PANELS REFLECTOR MUX SHELVES INSERTS & BRACKETS ENGINE & THRUSTER SUPPORTS		ACCESS PANELS CLIPS & SUPPORTS BRACKETS (SAD, THRUSTERS, ETC) BALANCE RAILS FASTENERS & CONNECTIONS		ACCESS PANELS ETC.	
TOTAL	135.3		243.0		159

OLYMPUS RECONFIGURATION
DOC PRESENTATION
SEPTEMBER 26, 1986

<u>Figure 1</u>

STRUCTURE MODIFICATION

K.Duffy

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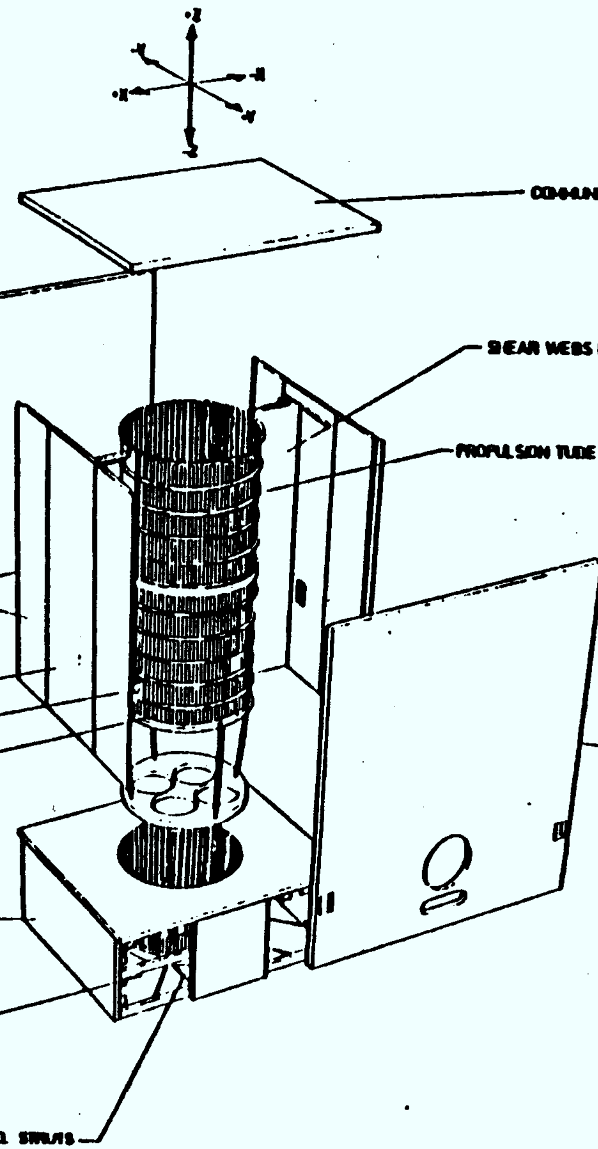
BASIC OLYMPUS STRUCTURE



SASD

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* Figure 2

STRUCTURE MODIFICATION

K.Duffy

BASIC OLYMPUS STRUCTURE

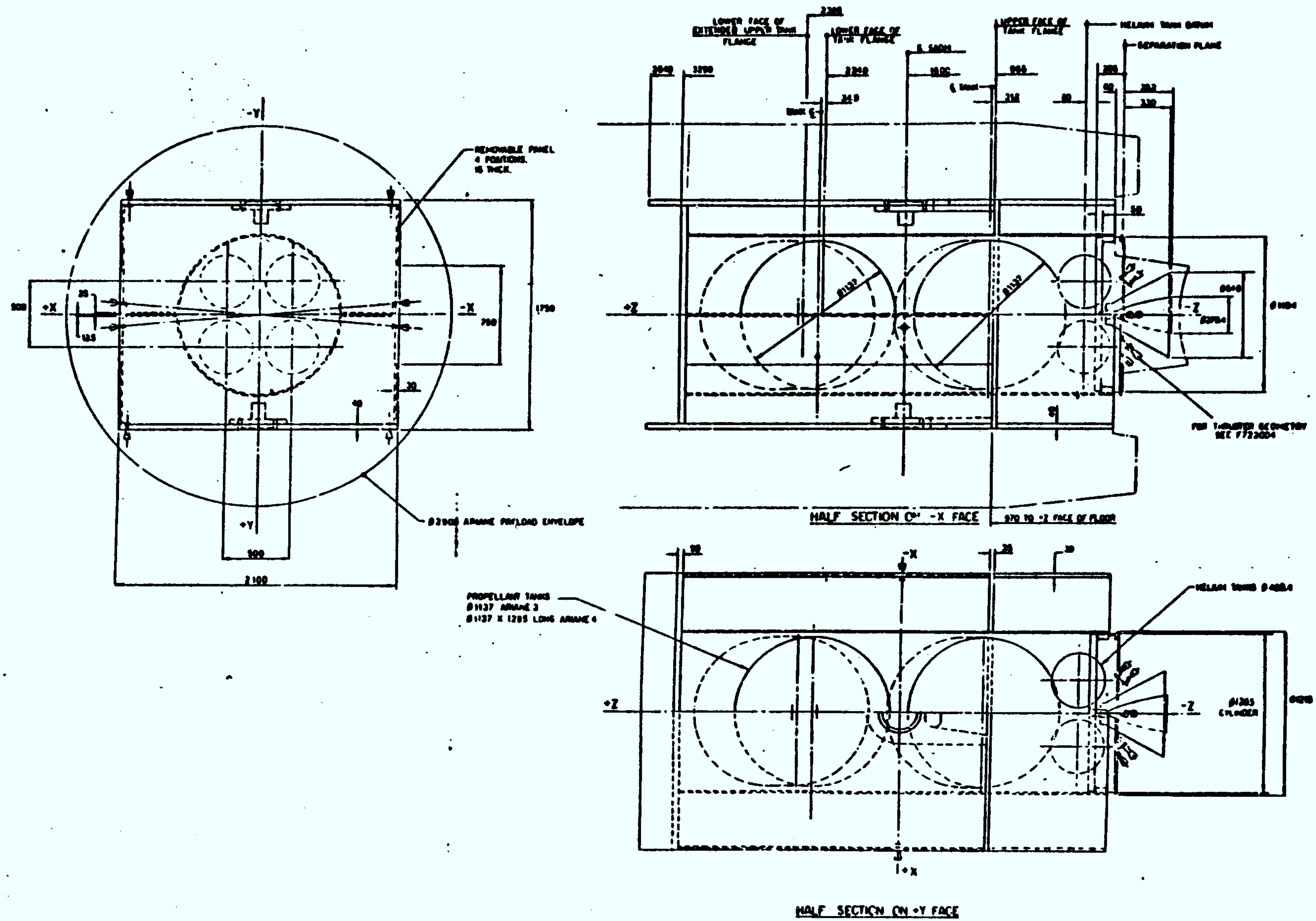

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Figure 3

STRUCTURE MODIFICATION

K.Duffy

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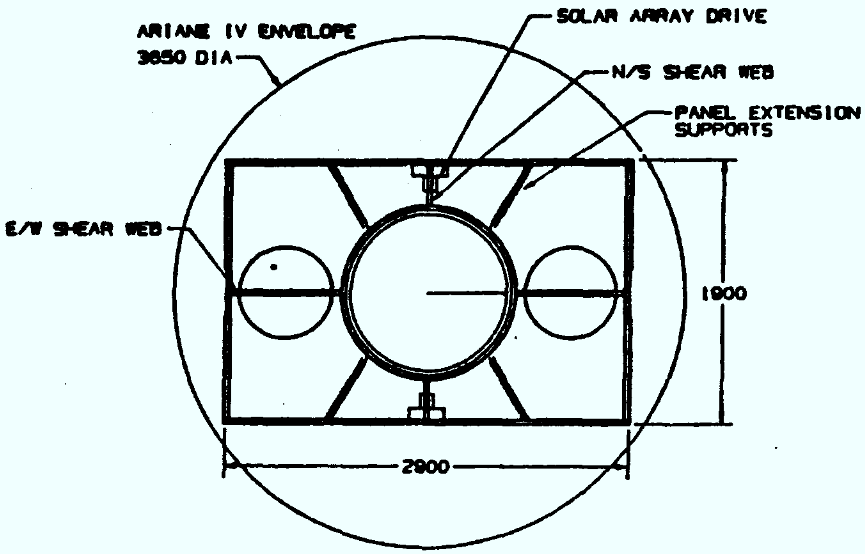
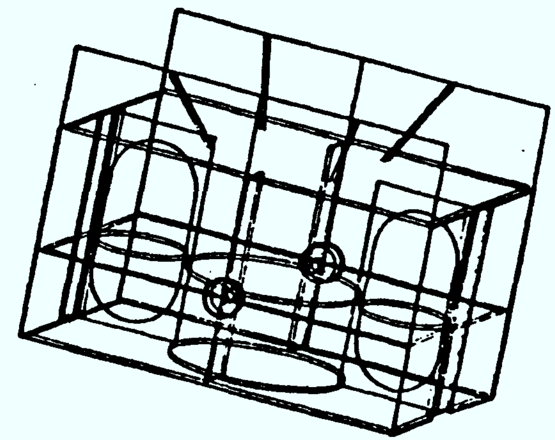
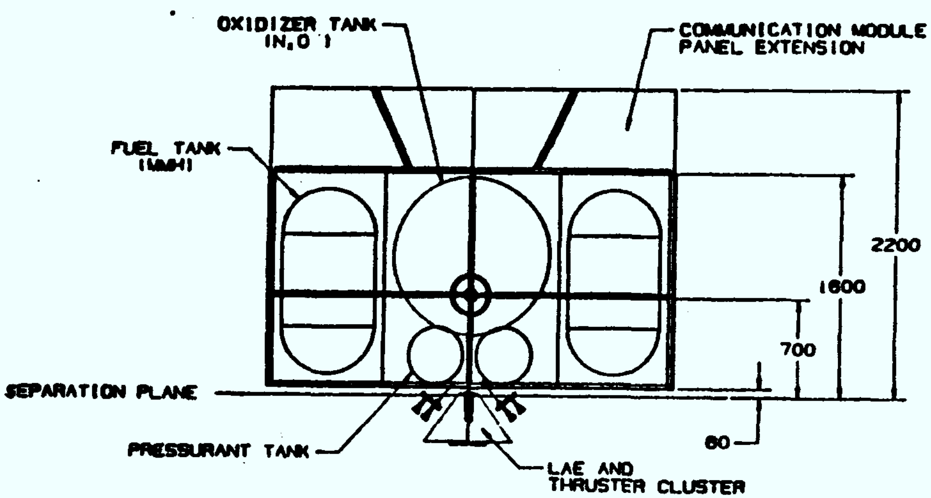
PLATFORM GEOMETRY : RECONFIGURED OLYMPUS



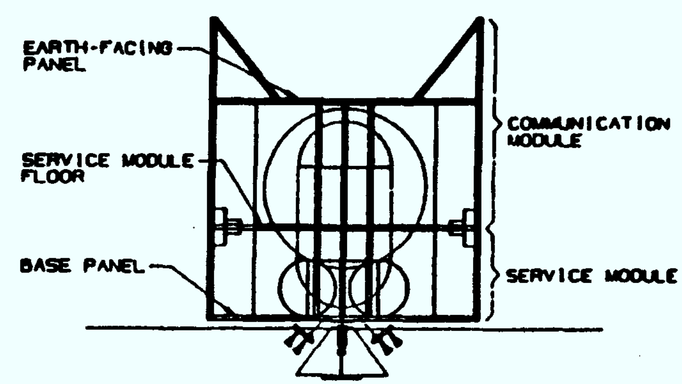
EARTH-FACING VIEW



ISOMETRIC VIEW
(PROPULSION MODULE REMOVED)



NORTH/SOUTH VIEW



EAST/WEST VIEW

OLYMPUS RECONFIGURATION
DOC PRESENTATION
SEPTEMBER 26, 1986

Figure 4

STRUCTURE MODIFICATION

K.Duffy

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SOLAR ARRAY TIEDOWN SCHEME

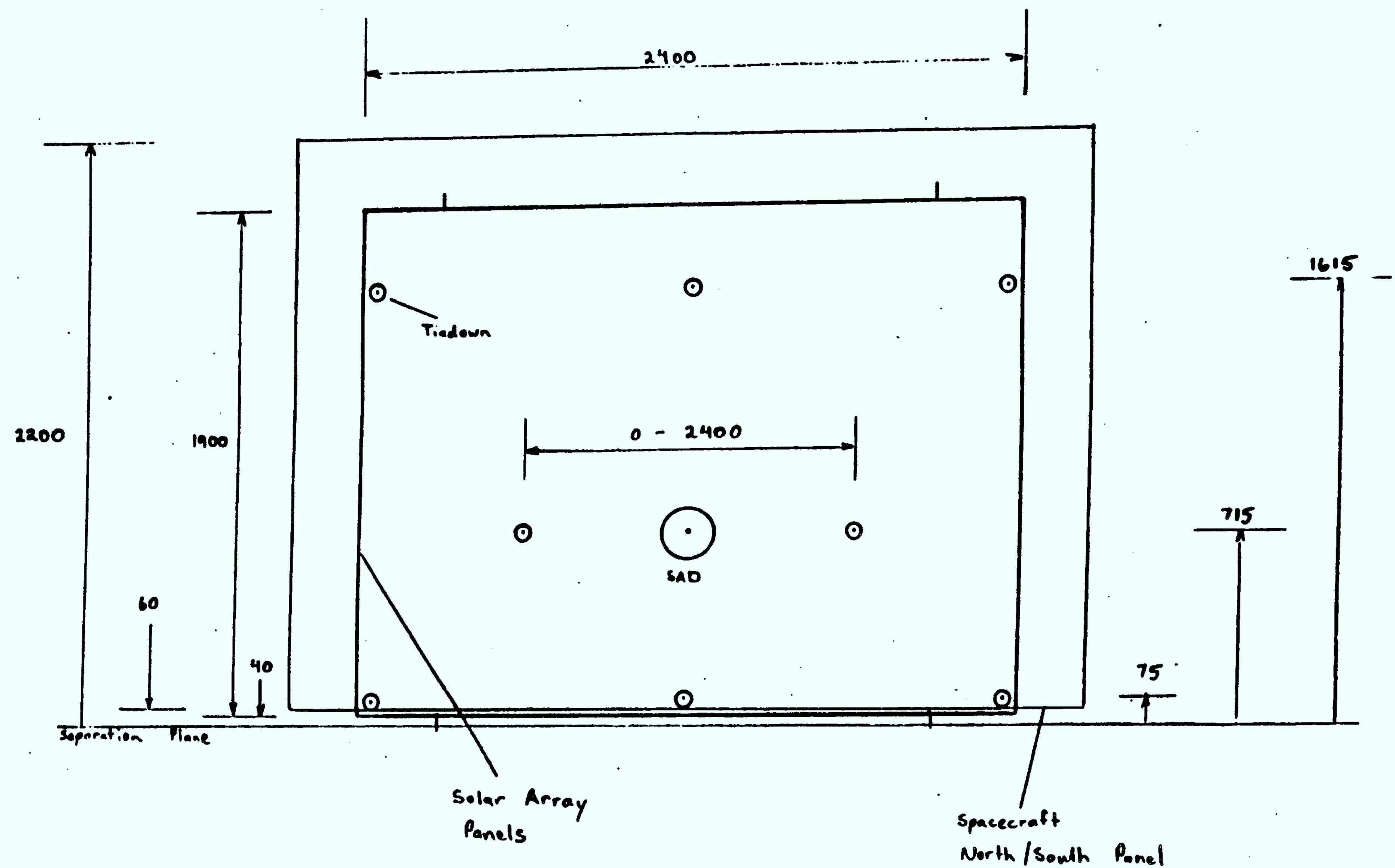

Figure 5

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AUSSAT II SINGLE 3-METER SOLID CONFIGURATION (STOWED)

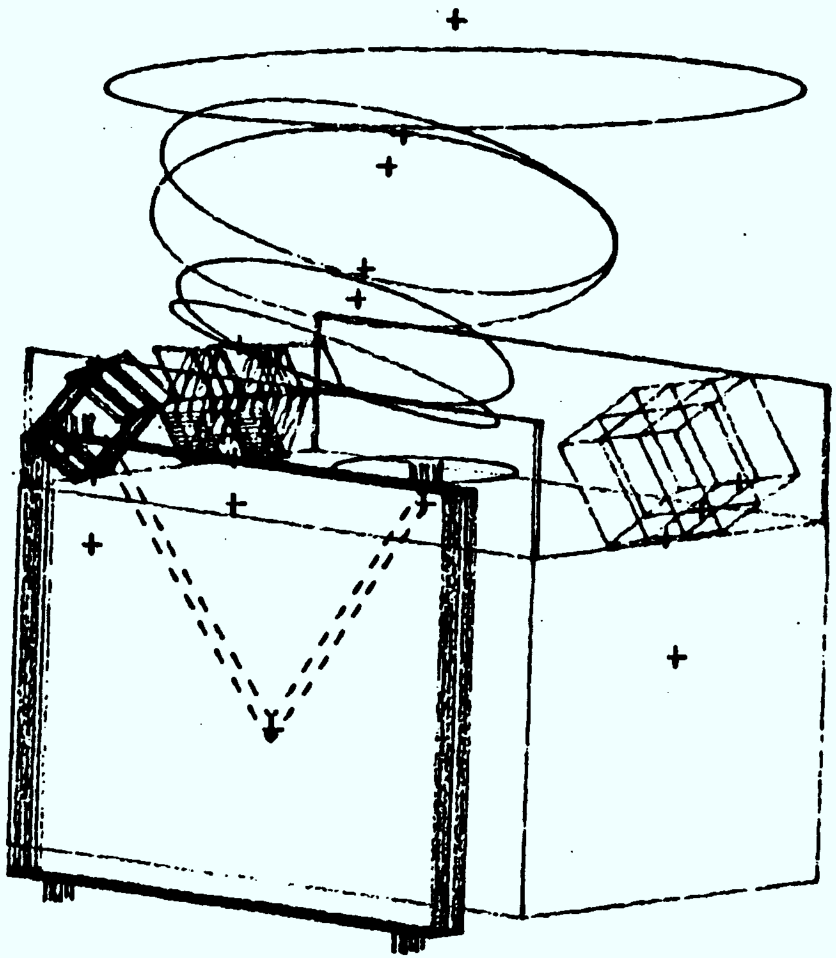

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Figure 6

AUSSAT II SINGLE 3-METER SOLID CONFIGURATION (DEPLOYED)

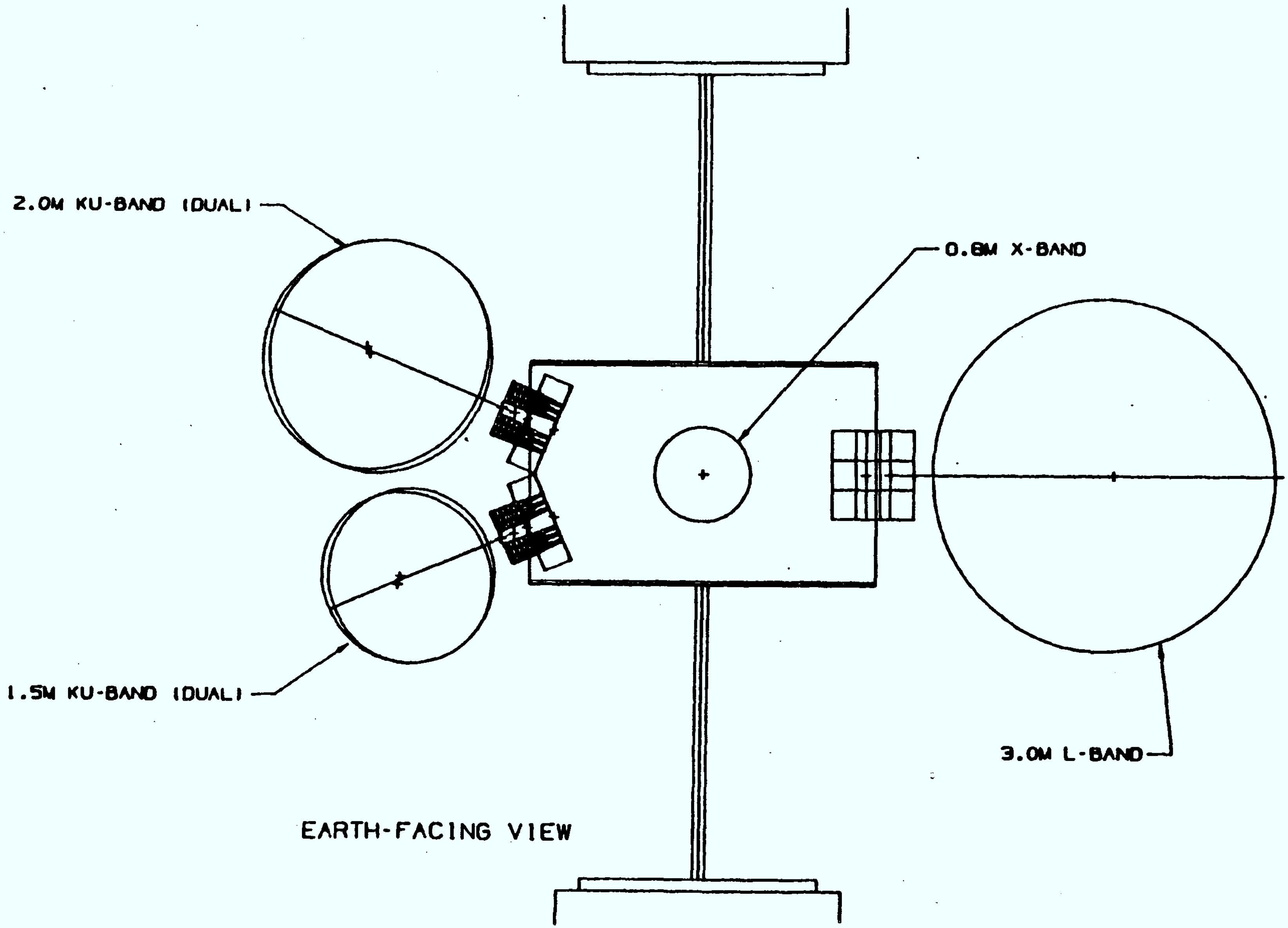

Figure 7

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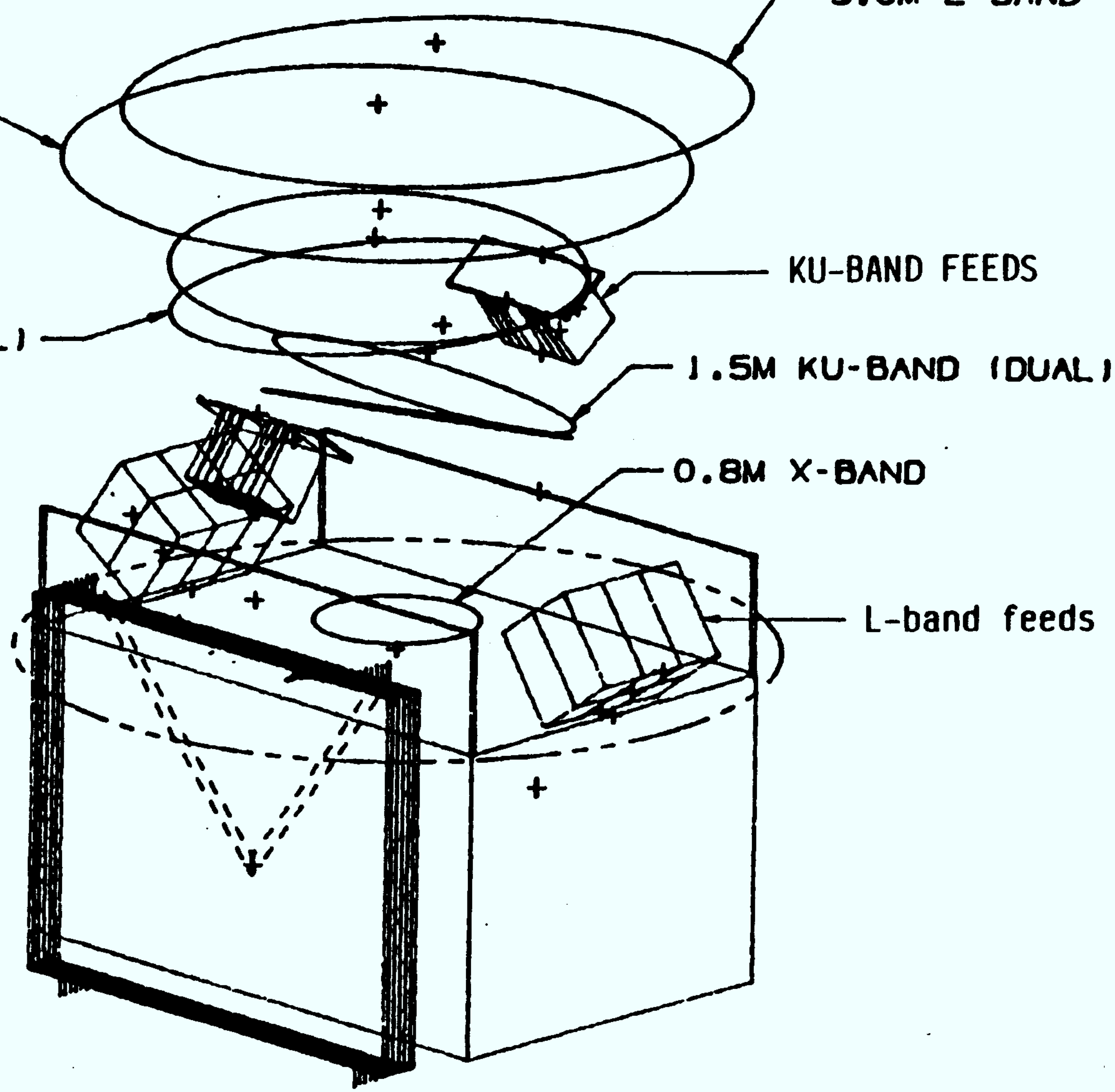
AUSSAT II DUAL 3-METER SOLID CONFIGURATION (STOWED)



C14

Figure 8

SPAR

AUSSAT II DUAL 3-METER SOLID CONFIGURATION (DEPLOYED)

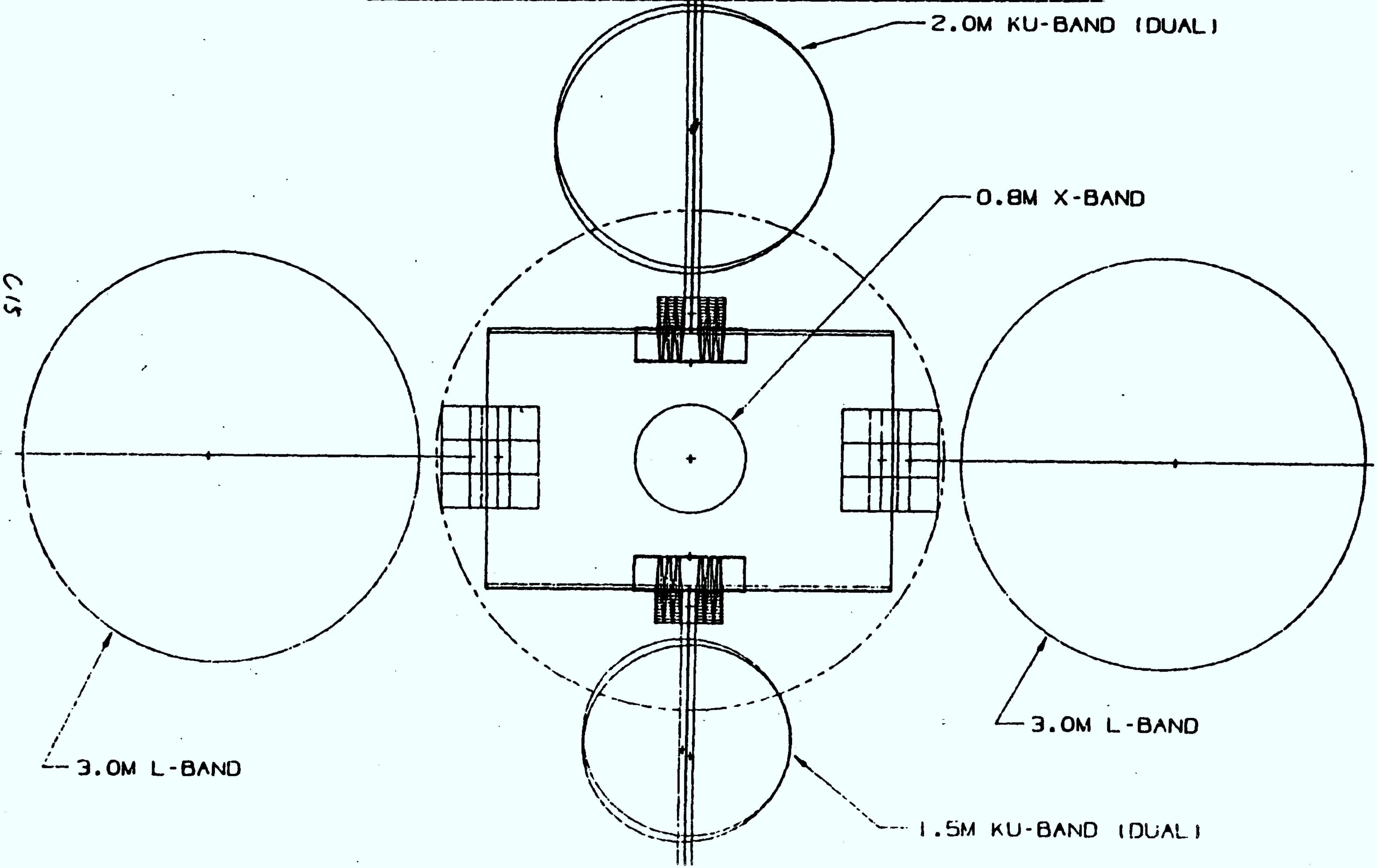


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Figure 9

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AUSSAT II DUAL 5-METER UNFURLABLE CONFIGURATION (STOWED)

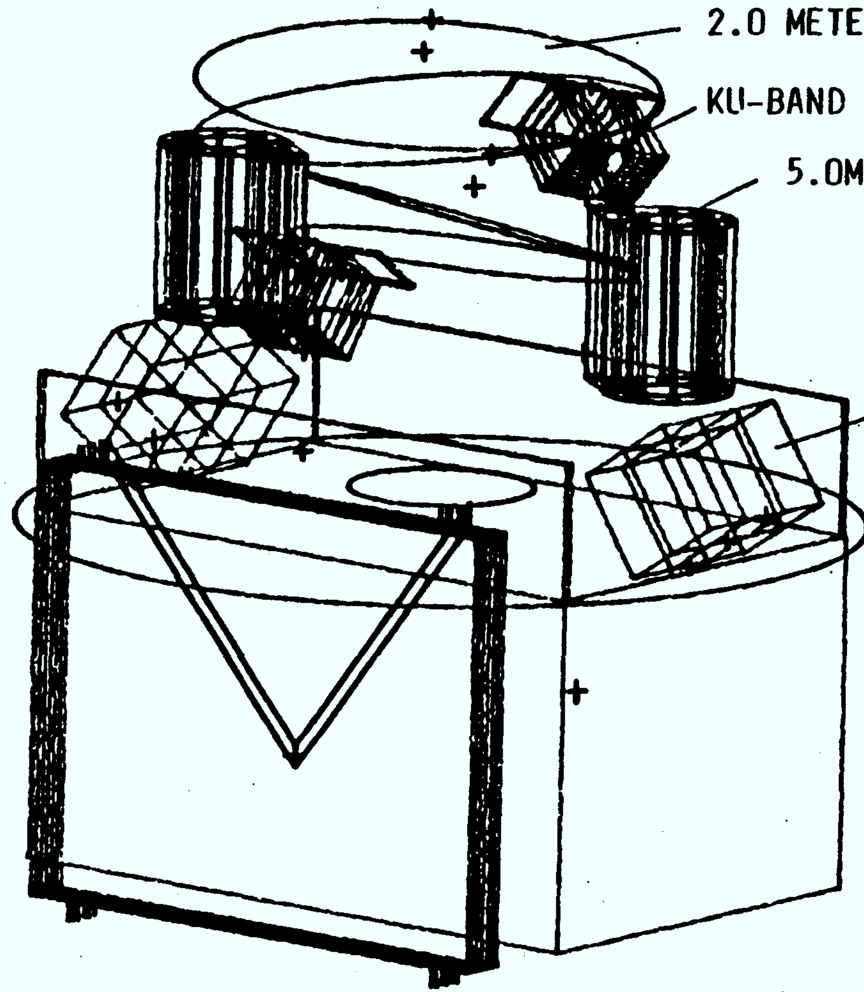

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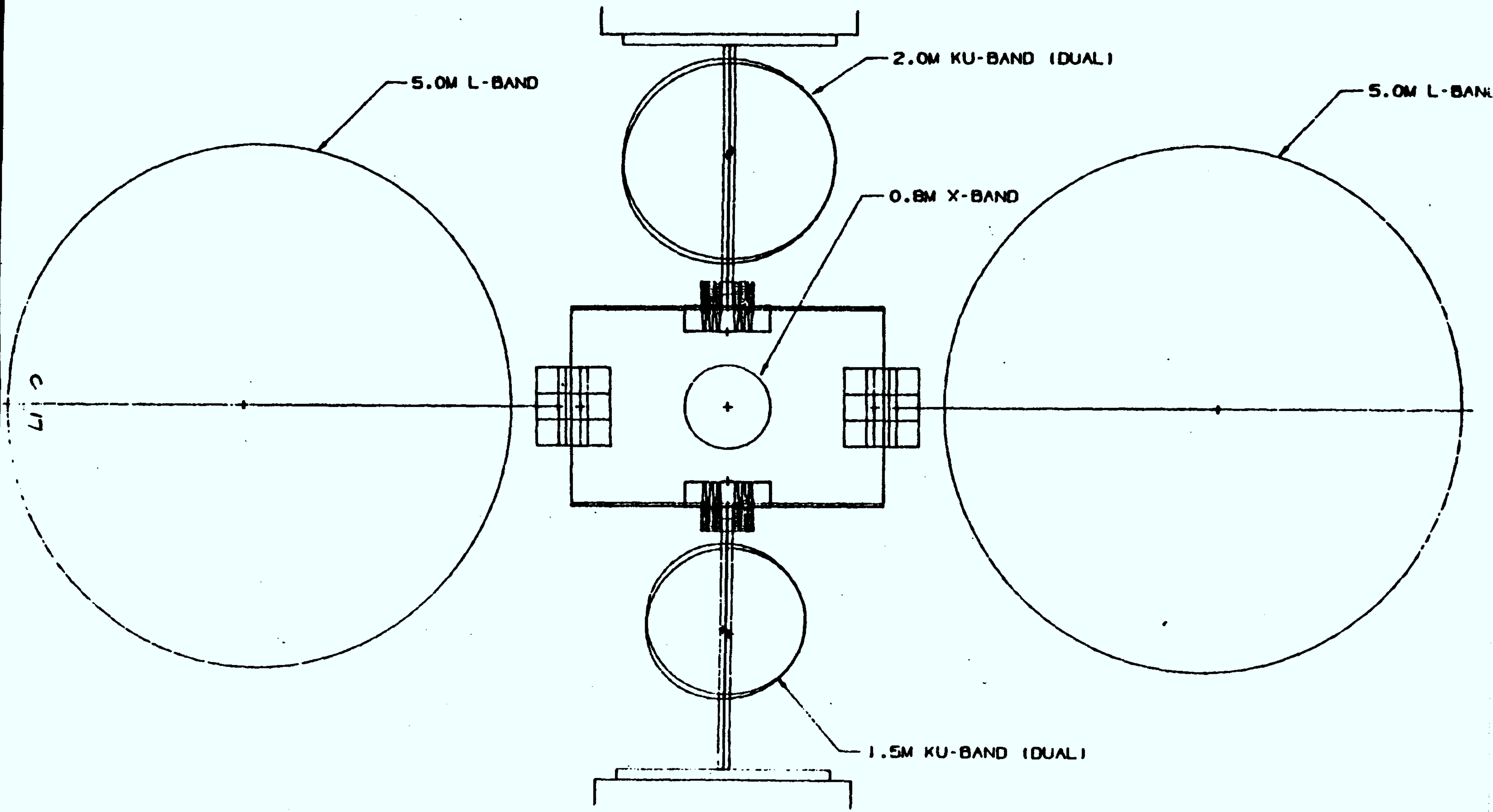
Figure 10

SPAR

AUSSAT II DUAL 5-METER UNFURLABLE CONFIGURATION (DEPLOYED)



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SECTION D: SOLAR ARRAY FOR MODIFIED OLYMPUS

"Reduced Olympus" Solar Array

This report describes a rigid panel solar array suitable for the reduced Olympus bus.

The heritage of the design is as follows. Spar RMSD created a rigid panel array design for the DBS spacecraft in 1984. Some breadboard work was done on this array in that year. This design was modified for the Olympus Rigid Panel Solar Array study which had as an aim, to consider a rigid panel solar array for the original Olympus bus. Also evolved from this design is our proposed concept for the Radarsat spacecraft. The array design described below is based on those produced for these studies.

An overall description of the array is provided in the attached Appendix. The panel hinges, SAD hinge, tie down system, deployment coordination system, and damper have all been sized in the previous studies and these should be compatible with the requirements of the reduced Olympus bus array as well. The basic yoke design is also standard although, of course, the detailed dimensions of the yoke depend on the exact spacecraft requirements. The panel facesheet material and thickness, and the core thickness are definitely spacecraft dependent. Allowing these panel parameters to vary ensures a mass optimum design while still keeping the basic panel construction compatible with that we have already studied.

The panel plan which was considered is that described in your FAX of September 23, 1986. The panel dimensions are 1900 mm by 2400 mm. There are eight tie downs. Although you stated that all tiedowns could be rigidly fixed we have considered that some must be released in-plane in order to minimize loads caused by differential thermal strains between the array and the spacecraft. We have also incorporated moment releases into the tie downs at the level of the lowest panel to lower the moments applied to the spacecraft. These features lower the stowed frequency somewhat, and as such, cause the array to be heavier to meet the stowed frequency requirement. We feel, however, that these force and moment releases are likely to be required, so we have included them in the design.

We considered the following design requirements:

- array life of 10 years in geosynchronous orbit.

- power requirement of approximately 4 kW EOL per spacecraft. This implies 6 panels per wing.

- stowed natural frequency at least 50 Hz, and as an alternate, 30 Hz.

- deployed natural frequency at least 0.05 Hz.

- launch loads on the order of 20 g quasi-static at qual.

- on-orbit loads due to apogee engine firing on the order of 0.03 to 0.04 g.

Of these requirements, the stowed natural frequency dominated in all cases described below.

We considered facesheets of carbon composite and Kevlar. In general it is found that the available material thicknesses of carbon composite have a significant impact on the mass of the array by being thicker than would be desired for an optimum design. Although thinner and higher modulus materials are available, we have opted in this design for 0.007 inch thick woven carbon composite fabric with modulus of 10E6 psi. This should provide a panel sufficiently robust for easy ground handling, and also should minimize program risk because the panel manufacturing procedures which would have to be developed would be simpler. Note, however, that mass reductions of the carbon composite arrays of 10% to 20% from the values reported below would be possible by going with slightly more exotic materials (with a correspondingly greater development risk) if this is deemed desireable. The Kevlar used was 0.002 inch thick woven fabric with a modulus of 4.5E6 psi. A Kapton insulating layer is necessary with carbon fiber, none with Kevlar.

The honeycomb core for the panels was the lightest available, 1 lb/cu.ft., in all cases.

The solar cells considered were 180 micron silicon cells, similar to those on Olympus 1, as well as 150 micron, 100 micron, and 50 micron cells. It was found that there was surprisingly little difference in power to mass ratio using these various cell types. This is due to the long life of Olympus which causes the higher radiation degradation of the thin cells to reduce or even overcome any initial mass advantage they may have. Of the designs described below, only the 30 Hz Kevlar design was optimum (in terms of power to mass) with the 50 micron cells, while the others were optimum with the 180 micron cells.

The power analysis accounted for radiation degradation, and included typical allowances for breakage during ground handling and for contamination due to thrusters on orbit. Also considered was the effect of operating temperature on the solar cell efficiency. Thicker honeycomb panels lead to warmer cells, and therefore less power per unit area. Therefore the arrays described below vary slightly in their power output, although all achieve at least 4 kW per spacecraft.

Panel designs which meet the 50 Hz stowed natural frequency requirement within the material limitations described above, and the mass of the complete array with these designs, are:

- 0.007 inch carbon fiber facesheet on 1.22 inch core, providing 2.10 kw per wing EOL equinox power for 79.1 kg per wing, giving 26.5 W/kg.

- 0.006 inch Kevlar facesheet on a 1.82 inch core, providing 2.06 kW per wing EOL equinox power for 79.2 kg per wing, giving 26.0 W/kg.

It can be seen that carbon fiber is preferred for the 50 Hz design.

Panel designs which meet the 30 Hz stowed natural frequency requirement are:

- 0.007 inch carbon fiber facesheet on a 0.68 inch core, providing 2.14 kW per wing EOL equinox power for 73.2 kg per wing, giving 29.2 W/kg.

- 0.004 inch Kevlar facesheet on a 1.18 inch core, providing 2.10 kW per wing EOL equinox power for 67.5 kg per wing, giving 31.1 W/kg.

It can be seen that Kevlar is preferred for the 30 Hz design, although as stated above, going with somewhat more exotic carbon fiber technology could improve the carbon fiber power to mass ratio to the 35 to 40 W/kg range, providing the more difficult development effort and ground handling were deemed acceptable.

APPENDIX - ARRAY DESCRIPTION

General Description

Two solar arrays are used on the   spacecraft. The arrays are rigid panel type arrays, in which honeycomb panels are used for the cell substrates. Each array consists of a yoke and six panels, which are connected in series using a pair of hinges between the yoke and the inboard panel and a pair of hinges between each panel and the next. A base-hinge is fitted on the inboard end of the yoke which is used to attach the array to the spacecraft .

These hinges permit the array to be folded up concertina fashion so that the spacecraft will fit within the launch vehicle envelope. The folded array (or "pack") is held against the spacecraft by eight tie-downs.

After launch the tie-downs are released and the array (driven by springs fitted in the hinges) unfolds and deploys. The deployment is controlled by a system of co-ordination cables and a damper.

Array Design

2.1 Panel Design

2.1.1 Mechaincal Design

The panel design is described in the body of the report. Eight holes are cut through the panel in which cup and cone fittings are bonded. These fittings are used with the tie-downs to support the stowed array.

Two aluminum inserts for mounting the hinges are fitted into each long edge of the panel.

2.1.2 Electrical Design

Four by six centimetre cells are used. They are connected in six strings in parallel, each of 156 cells in series, which are subdivided into 12 submodules. Each submodule is shunted by a bypass diode. Three strings are wired to form an electrical section. There are two electrical sections on each panel.

The cells and their interconnecting wiring are bonded onto the frontside of the panel and the bypass diodes are bonded onto the rearside. Through panel wiring is used to connect the diodes to the frontside wiring and to connect the electrical sections to the main cable harness which is bonded onto the rearside of the panel. An insulation layer of Tedlar is bonded onto each side of the panel prior to bonding the cells and the diodes.

2.2 Yoke Design

The yoke is a jointed design consisting of a short link arm and a double "V" shaped structure linked by the intermediate hinge. This design is used to provide the required array-spacecraft separation within the constraints imposed by the S.A.D. and tie-down locations.

The yoke is assembled from graphite fabric-epoxy tubes and aluminum end-fittings. The fittings on the inboard end of the link arm are attached to the base hinge and those on the outboard end of the yoke are attached to the inboard panel hinges.

2.3 Hinges

2.3.1 Base Hinge

The base hinge attaches the yoke to the spacecraft. It also supports the bracket for the cable harness interface connectors and supports the damper.

The hinge consists of a base bracket and a yoke fitting which are joined by a hollow hinge pin. The hinge pin is fixed in the yoke fitting and rotates in bearings mounted in the base bracket.

A helical torsion spring is fitted on one side of the hinge which opens the hinge during array deployment. (It is connected to the base bracket at one end and to the hinge pin at the other).

A pulley for the coordination system is fitted on the other side of the hinge.

The damper is mounted on the rear of the base bracket. It is driven via a pinion by a gear fixed to the yoke fitting. Eddy current and hydraulic dampers are being evaluated, although at this time we feel that an eddy-current damper is most suitable.

The hinge rotates through 90 degrees during deployment,at the end of which a latch engages to lock the hinge in position.

2.3.2 Intermediate Hinge

The intermediate hinge joins the link arm to the "V" frame. It consists of a hollow hinge pin fitted into the end of the link arm upon which the two fittings attaching the "V" frame rotate.

A torsion spring is fitted to drive the hinge open during deployment.

Pulleys for the coordination system are fitted on each side.

The hinge rotates through 180 degrees during deployment,at the end of which a latch engages to lock the hinge in position.

2.3.3 Panel Hinge

Two panel hinges are used to connect one panel to the next. They are attached to the edges of the panels by screws which allows the panels to be of modular design and facilitates quick replacement of a panel.

Each panel hinge consists of a fork bracket and a cam bracket joined together by a hinge pin. The hinge pin passes through holes in the fork bracket and through a bearing mounted in the cam bracket.

A self-aligning ball bearing is used to give low friction torques during deploynment and to allow for any misalignments caused by thermal distortion of the panels.

The hinge rotates through 180 degrees during deployment, at the end of which a latch, fitted on the fork bracket, engages in a hole, in the cam bracket, to lock the array in positon.

Co-ordination cable pulleys and guide pulleys are also fitted on the hinge.

2.4 Co-ordination System

The panel rotations during deployment are coordinated by a cable and pulley system. The cables run behind the panels from the pulley fitted on one hinge to the pulley fitted on the next outboard hinge.

Spring tensioners will be fitted to keep the cables taut during deployment and to allow for differential thermal expansion between the cables and the panels.

2.5 Tie-down System

The tie downs hold the folded panels together in a pack and hold the pack to the spacecraft. Each tie-down consists of a bracket, tie-together components (clamp plates and a tie-rod) and a release mechanism for the tie-rod.

The panels are held together under preload between the outer clamp plate (fixed to the outboard panel) and the inner clamp plate (attached to the top of the bracket). The preload is generated by the tie-rods which pass through the centres of the tie-downs. Their inner ends are held by the release mechanisms (mounted in the brackets) and nuts on their outer ends are tightened up against belleville washers contacting the outer clamp plate.

The release mechanism consists of a latch and lever system. The tie-rod is retained by a latch, which is retained by a lever, which in turn is retained by a roller cam. The mechanism is operated by a system of spring tensioned cables which rotates the roller cam thus releasing in turn the lever, latch and the tie-rod. Two cable systems will be used, each one operating the release mechanisms in three tie-downs.

The cables are prevented from operating the system while the array is stowed by the pin of pyrotechnic pin-puller. The release of the system is initiated by firing the pyro. The pyro has dual initiators for redundancy.

2.6 Cable Harness

The power from the panels is fed down to the spacecraft through flat conducter cables which are bonded to the backs of the panels. These cables are brought down to terminal boards fitted on the cross bar of the yoke.

From there the power is fed through 16-gauge wires running down the yoke to connectors mounted on the base hinge connector bracket.

2.7 Thermal Design

Thermal control of the array will be by passive means only, ie. black paint on the back of the panels and thermal blankets on the yoke.

SECTION E: IMPACT OF PROPOSED OLYMPUS BUS ON CURRENT OLYMPUS MGSE

IMPACT OF PROPOSED OLYMPUS BUS ON CURRENT OLYMPUS M.G.S.E.

A comparison of existing and proposed bus dimensions in millimeters is presented below.

	XX-AXIS (mm)	YY-AXIS (mm)	ZZ-AXIS* (mm)
PRESENT BUS	1750	2100	3640
PROPOSED BUS	1600	2900	2200

* dimension includes Y face thermal skirt.

The significant changes in dimension in the YY and ZZ axes impact MGSE items in four ways - depending whether it is cost effective to modify each piece or not. They are:

CATEGORY I — Item cannot be used without extensive and costly modifications. Therefore item deemed unusable and must be redesigned and replaced.

CATEGORY II — Item can be used with modifications which is cost effective and enables it unlimited use.

CATEGORY III — Item can be used without modifications but with certain limitations placed on its use so as not to adversely affect the safety of the spacecraft.

CATEGORY IV — Item can be used freely without modification.

This study has been conducted by going through the BAe-produced MGSE user's guide and to classify each piece of MGSE in one of the categories above.

1. SPACECRAFT TRANSPORT CONTAINER (G600-210-000-00)

Used to transport the present integrated spacecraft with the spacecraft mounted on the spacecraft handling trolley with its ZZ-axis horizontal as shown in Figure I below. The spacecraft transport trolley has been placed in Category III and can be used by limiting its use in transporting only the spacecraft bus less any antennae or appendages. The spacecraft would have to be mounted on the simple stand with its ZZ-axis vertical as shown in Figure 2a & 2b.

A new integrated spacecraft transport container will have to be designed and manufactured.

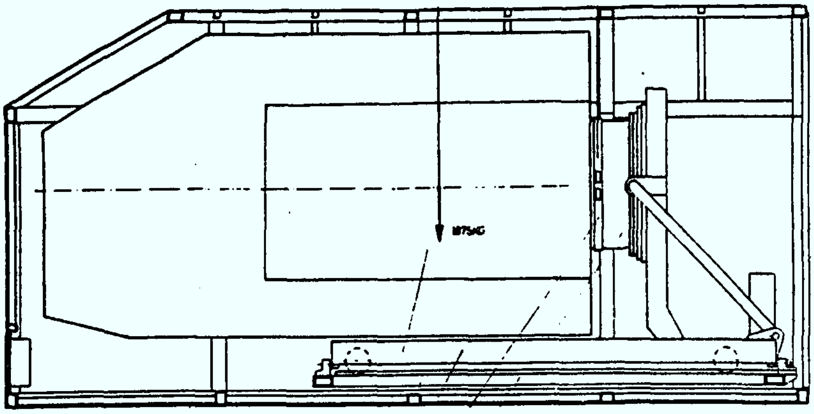
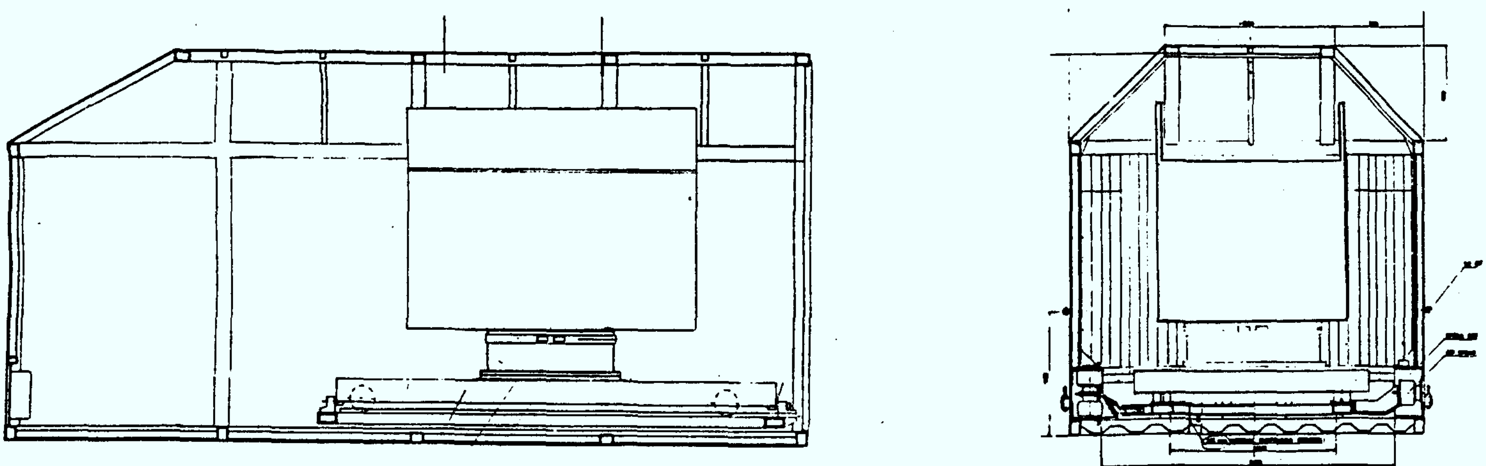


FIG. 1



(a) (b)

FIG. 2

2. UNIVERSAL CONTAINER (G600-110-000-00)

This item is presently used to transport the basic bus without antennae or appendages on the spacecraft transport trolley in the ZZ-axis horizontal position.

The proposed bus dimensions put the universal container in Category I, rendering it unusable. The present spacecraft transport container could take the universal container's function.

3. SPACECRAFT HANDLING TROLLEY (G600-220-000-00)

Is presently used in the spacecraft transport container and on its own as a "work bench" for operations on the spacecraft. It can hold the spacecraft in the vertical position or in the horizontal position with either the YY-axis vertical or the XX-axis vertical as shown in Figure 3.

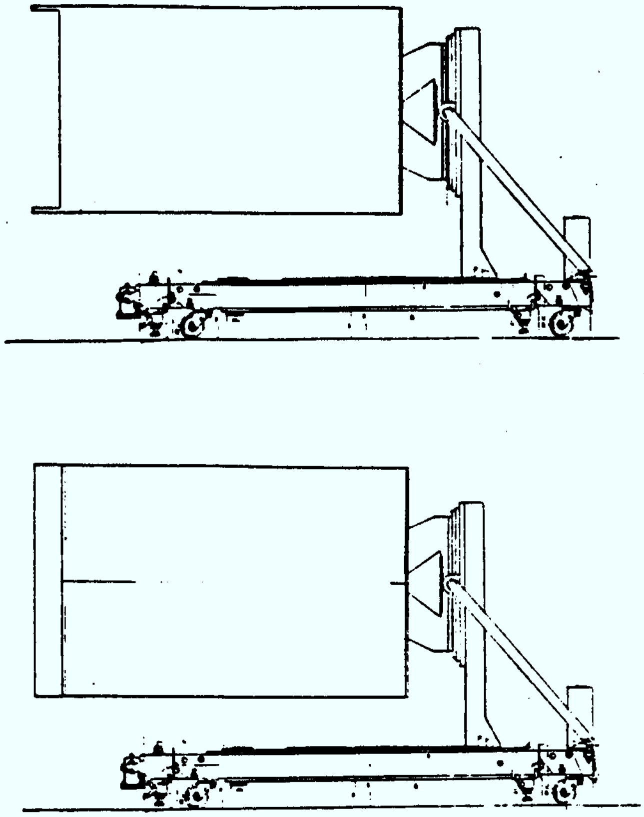


FIG.3

3. SPACECRAFT HANDLING TROLLEY (G600-220-000-00) (cont)

The proposed bus dimensions place this item in Category III except that a minor electrical modification is required to maintain spacecraft safety. As shown in Figure 4(b), when the spacecraft is in the YY-axis horizontal position the clearance is small between trolley and spacecraft. Diagonally the spacecraft would interfere with the trolley and thus the turntable must be prevented from rotating while the spacecraft is in the ZZ horizontal position. A simple electrical lock would be required. Thus the operation to reposition the spacecraft from Figure 4a to 4b would require the spacecraft to be put in the vertical position, rotated and then moved back to the horizontal position.

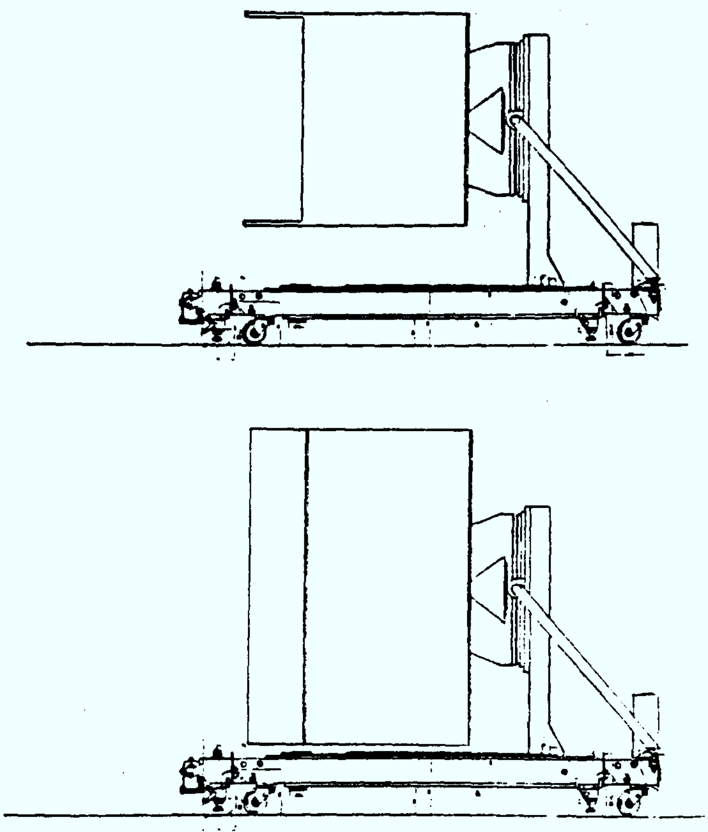


FIG. 4

4. SPACECRAFT TRANSPORT TROLLEY (G600-130-000-00)

Is used to transport the present basic bus in the universal container. For this purpose the proposed dimensions put this item in Category I.

In an emergency, however, the trolley could be used to transport a bus with its YY-axis horizontal. This practice is not strongly recommended.

5. MODULE HANDLING TROLLEY (G600-120-000-00)

Used to support several pieces of flight hardware at different times during AI&T program. Classified in Category II, item could be reused with various adaptors manufactured to enable correct interface with the various pieces of hardware.

6. SERVICE MODULE ADAPTOR FRAME (G600-15-000-00)

Used to support the Service Module on the Module Handling Trolley, this item is classified Category I and would probably have to be redesigned.

7. NORTH/SOUTH PANEL ADAPTOR (G600-140-000-00)

Used to support North or South panels on the Module Handling Trolley, classified as Category I and would have to be redesigned.

8. SPACECRAFT SIMPLE STAND (G600-230-000-00)

Supports any configuration of spacecraft with an adaptor required to interface the two. This item is Category IV and can be used as is.

9. PROPULSION MODULE INTEGRATION STAND (G600-240-000-00)

Used during Propulsion Module Assembly, this item is Category IV and can be reused because the propulsion system is unchanged.

10-16 Items 10 through 16 all pertain to the propulsion Module Assembly and are classified as Category IV to be reused as they are.

17 SPACECRAFT VERTICAL SLING (G600-311-000-00)

Used to lift the all-up spacecraft with its ZZ-axis vertical in conjunction with the lifting bracket (item 19). Classified as Category IV the sling can be reused providing modifications are done on the lifting bracket. See item 19.

18 SPACECRAFT HORIZONTAL SLING (G600-320-000-00)

This sling is used to lift the spacecraft with the ZZ-axis horizontal. A spacecraft adaptor or spacecraft lifting adaptor are required to connect the sling to the spacecraft carried at the separation plane end. Because the proposed ZZ-axis dimension is 2200mm (original 3640) this item is classified II and can be modified as shown in Figure 5.

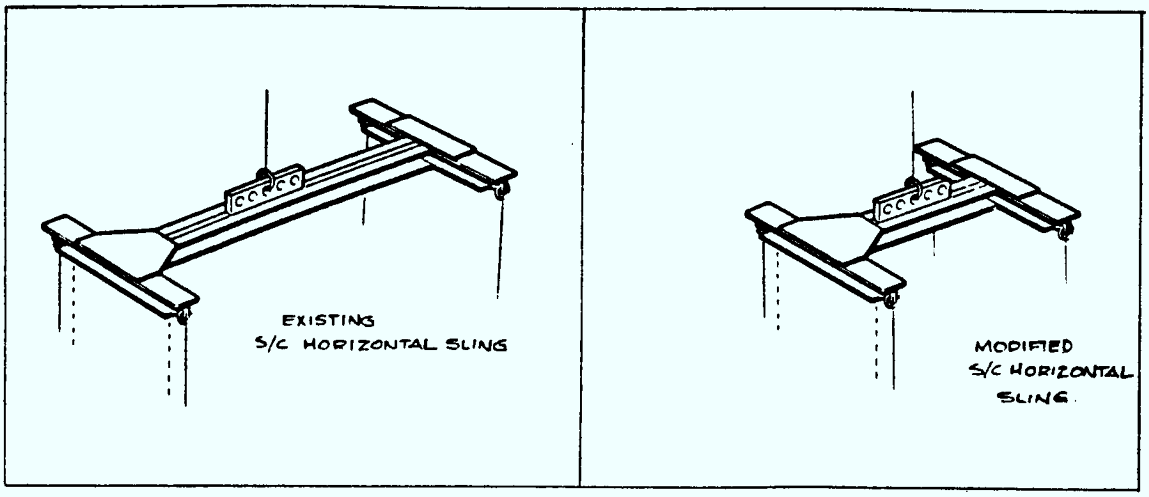


FIG. 5

19 LIFTING BACKET A (G600-330-000-00)

Used with Item 17 as shown in Figure 6a this item is classified as Category II and requires that the four lifting links be repositioned to the outboard side of the lifting brackets and lengthened to accomodate the spacecraft lift points as shown in Figure 6b.

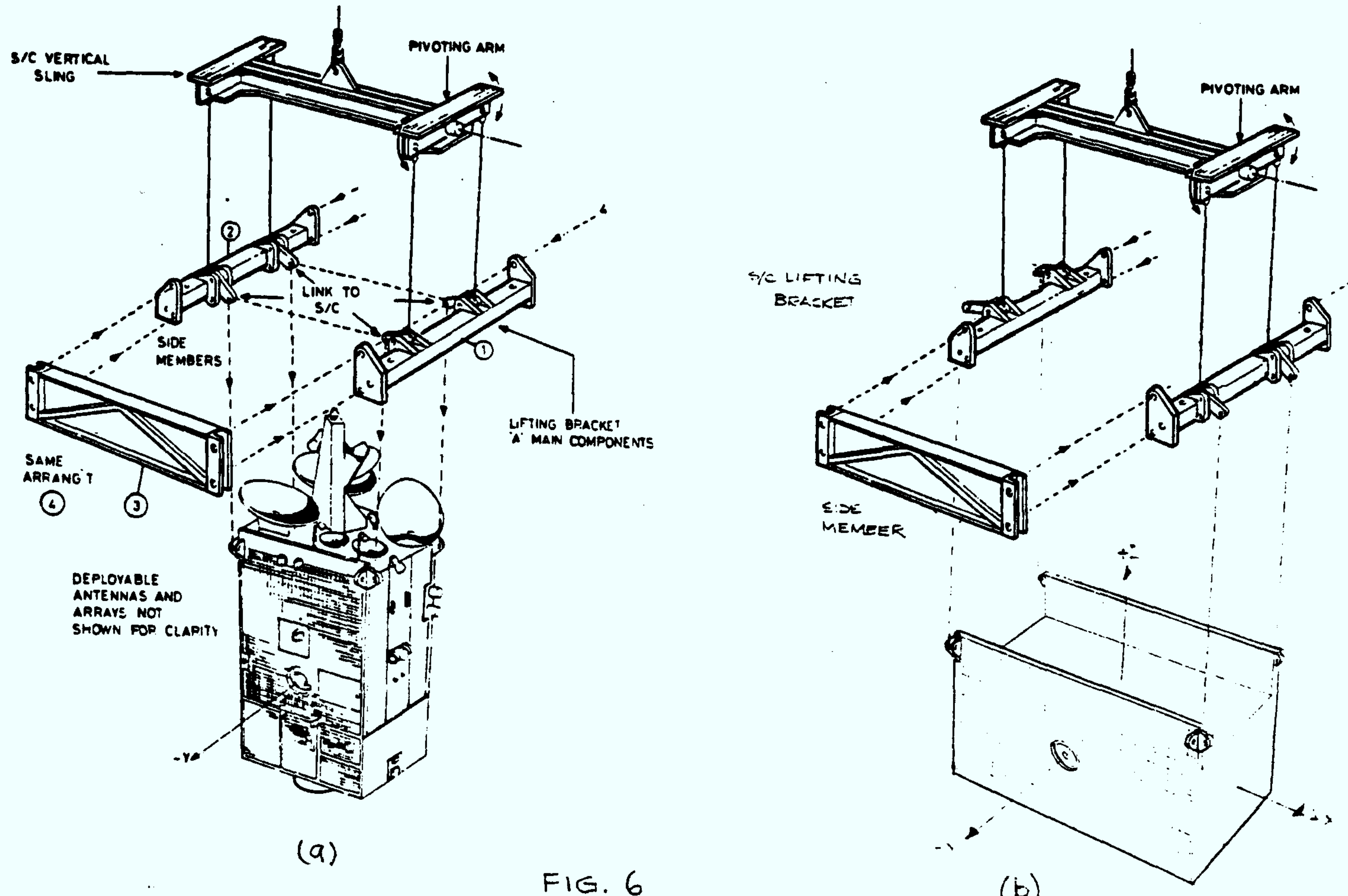


FIG. 6

20 NORTH/SOUTH PANEL SLING (G600-350-000-00)

This item is used to lift N and S panels from the Module Handling Trolley onto the spacecraft. Classified as Category I this item would require too many modifications to effectively adapt it to proposed bus dimensions. A new design is required.

21-22 BATTERY GROUND SUPPORT EQUIPMENT

These items are used during battery conditioning and integration and are classified Category IV as battery mods. are not anticipated.

23 SPACECRAFT LIFTING ADAPTOR (G600-362-000-00)

Used throughout the AI&T program, this item is classified Category IV.

24 ANTENNA FLOOR ADAPTOR (F640768)

Used to integrate the +Z floor to the spacecraft this item would have to be redesigned to the new dimensions. Category I.

25-27 VARIOUS G.S.E. ITEMS
All items are classified Category IV and can be reused.

28 N/A

29 WORKSTANDS (F640772, 773 & 774)

Scaffolding is modular and easily adapts to any size. Reusable.

30 THERMAL CONTROL EQUIPMENT

Used to control the temperature of the spacecraft. During operations testing this equipment is classified Category IV and can be reused.

31 DEPLOYABLE ANTENNA MGSE

Payload related.

32-37 VARIOUS SERVICE & TEST ITEMS

All can be reused. Category IV.

38 COMMUNICATIONS MODULE STRUCTURE (F640292)

Consists of a framework which supports N, S and top panels in positions which represent Flight Model as closely as possible. Classified Category I this item will require redesign.

39 SERVICE MODULE STRUCTURE (F640291)

Purpose is similar to that of item 38. Category I. Redesign required.

40 STAND EIM (F640293)

Purpose is similar to that of items 38 & 39. Category I. Redesign required.

41-43 PAYLOAD RELATED GSE

44-51 EIM RELATED GSE

Category I. Redesign required for all items.

52 ACOUSTIC TEST ADAPTOR ASSEMBLY (E2565160)

Part of test equipment, the acoustic test adaptor assembly interfaces with the spacecraft via the thrust tube and therefore is classified as Category II and can be reused with minor modifications.

53 VIBRATION ADAPTOR (K4046-02-101)

Part of test equipment, item supports the spacecraft on the DFL 40-K vibrator. Spacecraft is attached to the adaptor using a Heavy Duty Clamp Band (Item 26). Classified Category IV depending on test requirements. Should be able to be reused.

54 I.R. THERMAL TEST ADAPTOR

This item is test equipment used at DFL. Significant redesign is necessary to use the I.R. rig for a different bus. Classified Category II, most parts of the rig can be reused.

55 SOLAR ARRAY GSE (71297E)

This item is used to deploy the unfurlable solar array in air and in vacuum. Because the proposal is to use a rigid solar array design this item is classified as Category I and must be redesigned to suit a different deployment.

56-59 N/A

60-61 THERMAL TEST ADAPTORS A & B

If solar simulation testing is required, these items will be used at JPL and would require some redesign to be reusable, especially A, which interfaces with the spacecraft via the S/C lifting/interfaces brackets. Category II.

63 SPACECRAFT ALIGNMENTS ADAPTOR (G600-363-000-00)

Used during alignment, this item interfaces the spacecraft via the thrust tube and is classified Category IV. Reusable.

SECTION F: VIEWGRAPH PACKAGE OF SEPTEMBER 26, 1986

PRESENTATION TO DOC AT SPAR MONTREAL

OLYMPUS RECONFIGURATION STUDY

DOC PRESENTATION

SEPTEMBER 26, 1986

F2

AGENDA

10:00	INTRODUCTION	C. MORGAN
10:05	STUDY GOALS AND DRIVERS	A. KIDD
10:20	RMSD RIGID ARRAY STUDY	A. KIDD
10:30	STRUCTURE MODIFICATION	K. DUFFY
11:30	IMPACT TO OLYMPUS GSE	A. KIDD
11:45	SUMMARY AND CONCLUSIONS	A. KIDD

STUDY GOALS AND DRIVERS

O THE BASIC PURPOSE OF THIS STUDY IS TO EXAMINE METHODS OF INCREASING THE COMPETITIVENESS OF THE OLYMPUS SPACECRAFT FOR PAYLOADS IN THE MEDIUM RANGE (MASS: 300 - 400KG; POWER 2500 - 3500W).

THIS IS THE RANGE MOST PAYLOADS ARE EXPECTED TO BE WITHIN THE NEXT DECADE.

F4

STUDY GOALS AND DRIVERS (CONT'D)

- THE NEED FOR SUCH AN EFFORT CAN BE SEEN WHEN THE OLYMPUS L-1 MISSION IS COMPARED TO THE ANIK E ON THE RCA 5000.

		OLYMPUS C-1	ANIK E (RCA 5000)
PAYLOAD:	MASS	335KG	337KG
	POWER	2850 WATTS	2100 WATTS
	ECLIPSE POWER	1110 WATTS	2100 WATTS
LAUNCH VEHICLE		DEDICATED ARIANE III (2420KG GTO)	DUAL LAUNCH ARIANE IV (2500KG GTO)
MISSION LIFE		5 YEARS	10 YEARS + 2 YEARS IN ORBIT STORAGE

THE RCA SPACECRAFT CLEARLY OFFERS GREATER ECLIPSE POWER AND LIFE WHILE USING ALMOST IDENTICAL LAUNCH RESOURCES.

STUDY GOALS AND DRIVERS (CONT'D)

THE REASON FOR THE SUPERIOR PERFORMANCE OF THE RCA 5000 IS REVEALED BY A COMPARISON OF SUBSYSTEM MASSES BETWEEN THE TWO SPACECRAFT.

SUBSYSTEM	OLYMPUS L-1	ANIK E (RCA 5000)	DELTA
TT&C	40	35	+ 5
POWER *	150	165	- 15
SOLAR ARRAY	170	80	+ 90
ACS	100	70	+ 30
CPS	120	145	+ ~~110~~ -25
STRUCTURE	245	135	+110
THERMAL	60	40	+ 20
MECHANISMS	-	40	- 40
P/L HARNESS	10	-	+ 10
OTHER HARNESS	-	60	- 60
TOTAL PLATFORM	895	770	125

* OLYMPUS L-1 PROVIDES MUCH LESS BATTERY CAPABILITY THAN ANIK E

F6

STUDY GOALS AND DRIVERS (CONT'D)

- O THE PREVIOUS CHART SHOWS THAT THE SUBSYSTEMS IN WHICH OLYMPUS IS SIGNIFICANTLY HEAVIER THAN NORTH AMERICAN SPACECRAFT ARE THE STRUCTURE AND SOLAR ARRAY.

- O THE REASON FOR THIS DISCREPANCY ARE BASED IN THE OLYMPUS DESIGN PHILOSOPHY. IN ORDER TO ACHIEVE GROWTH CAPABILITY FOR PAYLOADS UP TO 600KG AND POWERS OF UP TO 7 KILOWATTS, THE SPACECRAFT STRUCTURE WAS DESIGNED TO ACCOMMODATE THESE WEIGHTS, THUS RESULTING IN OVER DESIGN AT LOWER PAYLOAD WEIGHTS. LIKEWISE, THE FLEXIBLE SOLAR ARRAY, WHILE EFFICIENT AT HIGH POWER LEVELS, IS NOT COMPETITIVE WITH RIGID PANELS OF LOWER POWERS.

GOALS AND DRIVERS (CONT'D)

THREE TARGET PAYLOADS WERE IDENTIFIED AS TYPICAL FOR SIZING THE MODIFIED OLYMPUS. THESE WERE:-

- O ANIK E
- O AUSSAT II
- O NORTH AMERICAN M-SAT

F8

GOALS AND DRIVERS (CONT'D)

ALL OF THE ABOVE PAYLOADS CAN BE EXPECTED TO REQUIRE A DUAL LAUNCH ON AN ARIANE IV. THIS LEADS TO THE FOLLOWING BUS REQUIREMENT.

- o LAUNCH CAPACITY OF ARIANE IV DUAL LAUNCH IS 2500 KG INTO GTO.

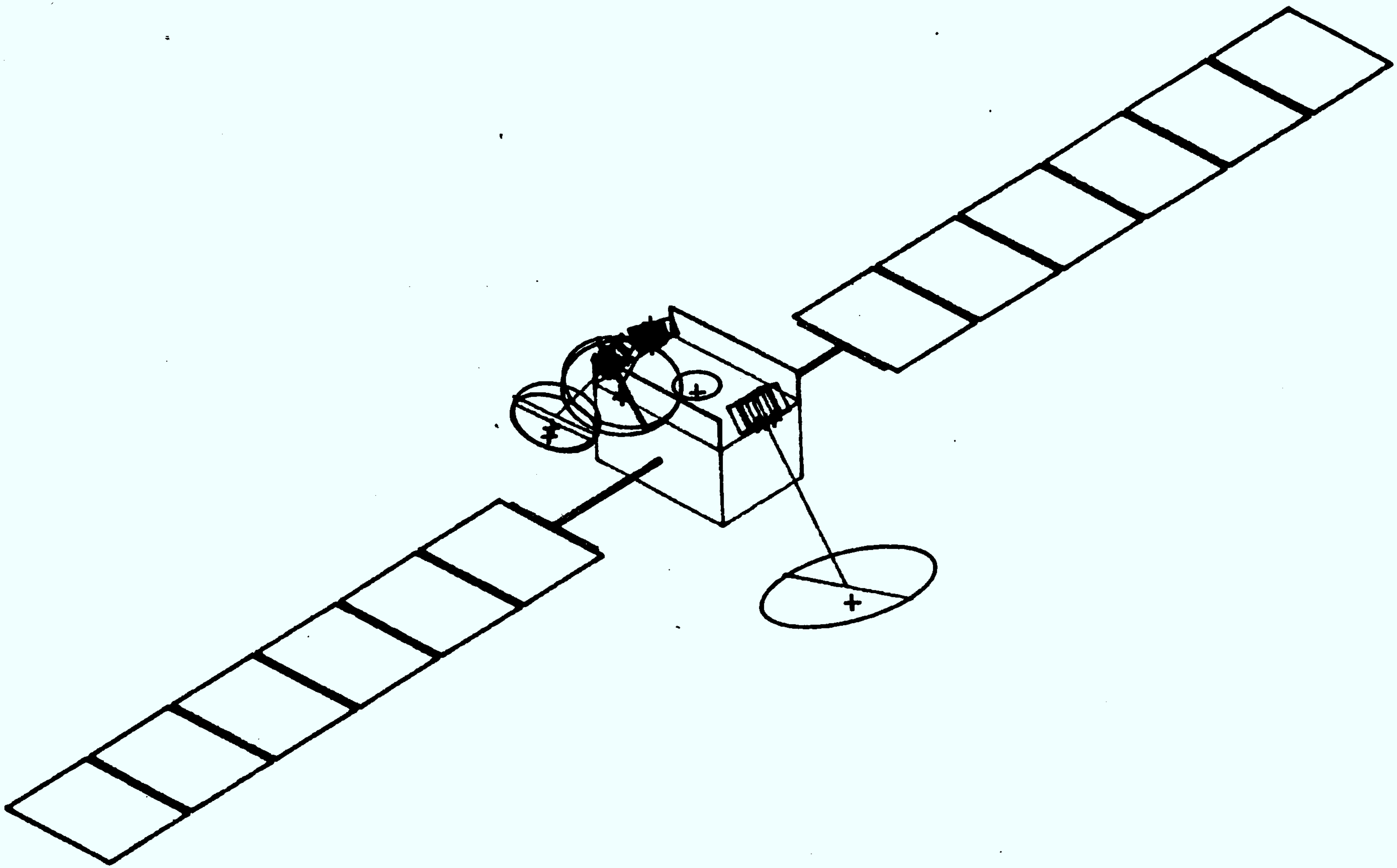
- O AVERAGE PAYLOAD REQUIREMENTS ARE 400 KG AND 2500 WATTS

THE ABOVE DATA COMBINED WITH ASSUMPTION OF LIQUID APOGEE ENGINE, BIPROP CPS, AND 10 YEAR LIFE LEAVE A MAXIMUM ALLOWANCE OF 796 KG FOR THE PLATFORM WITH A 44 KG MARGIN. THE SPACECRAFT POWER MUST BE SUPPLIED USING THIS MASS.

F9

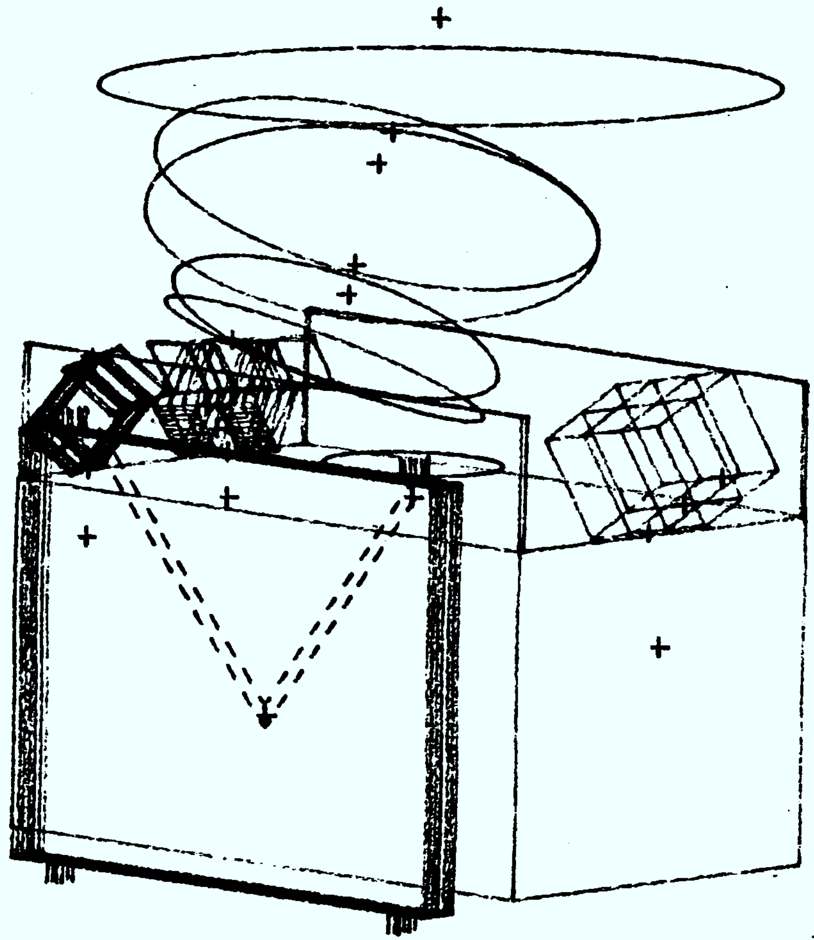
SPAR

AUSSAT II SINGLE 3-METER SOLID CONFIGURATION (DEPLOYED)



F10

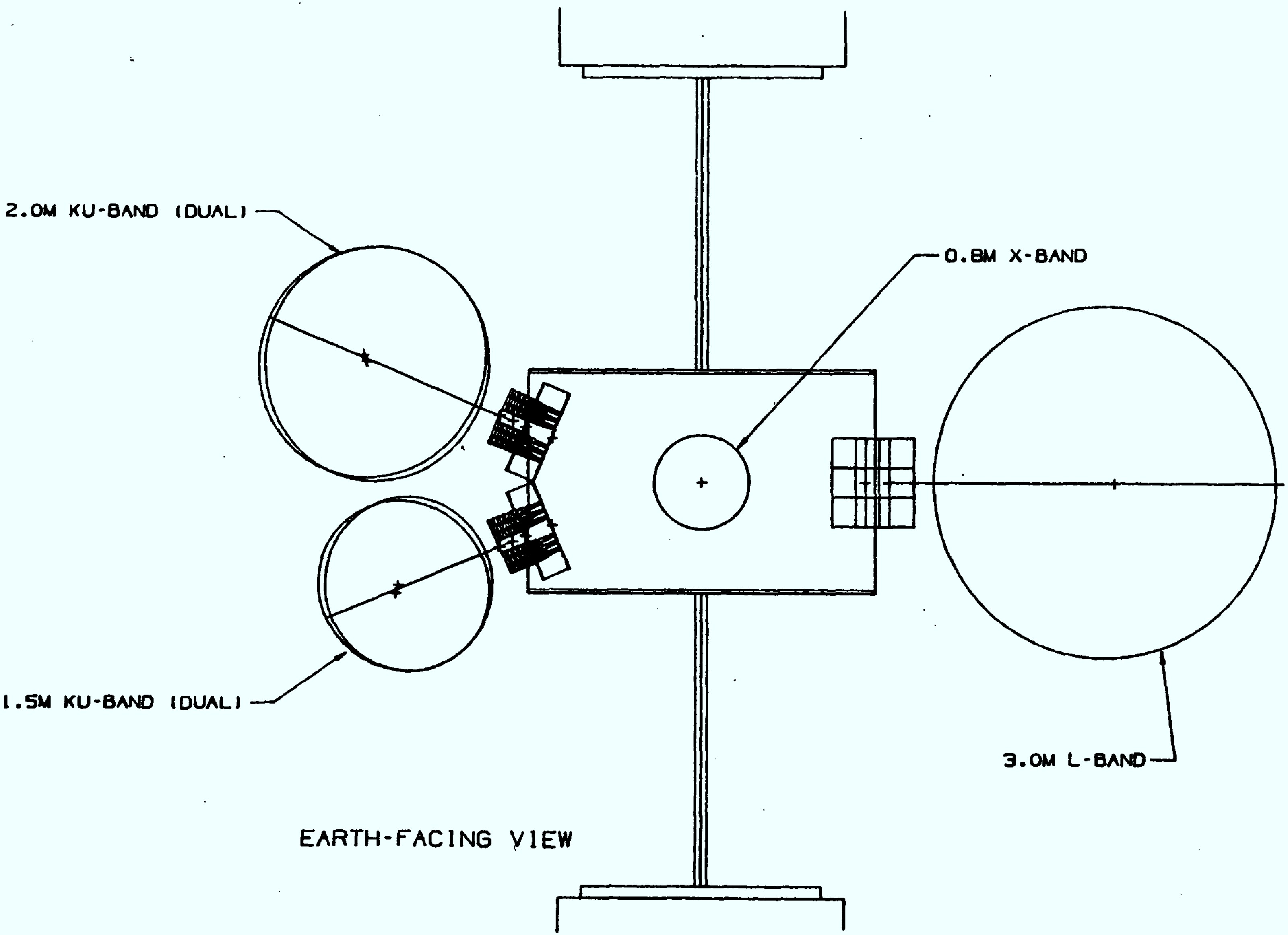
AUSSAT II SINGLE 3-METER SOLID CONFIGURATION (STOWED)



F11

SPAR

AUSSAT II SINGLE 3-METER SOLID CONFIGURATION (DEPLOYED)

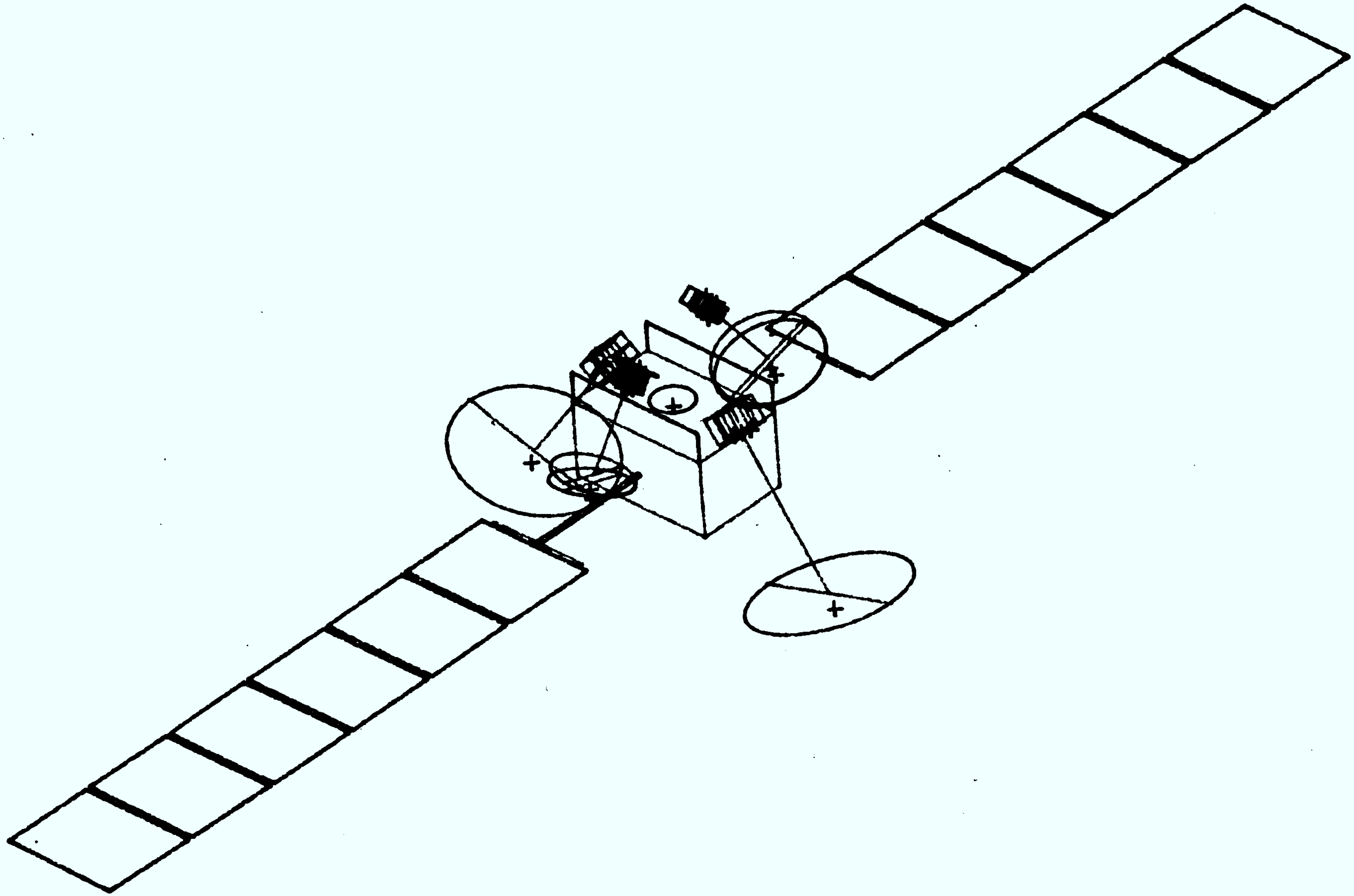


F12

SPAR

AUSSAT II DUAL 3-METER SOLID CONFIGURATION (DEPLOYED)

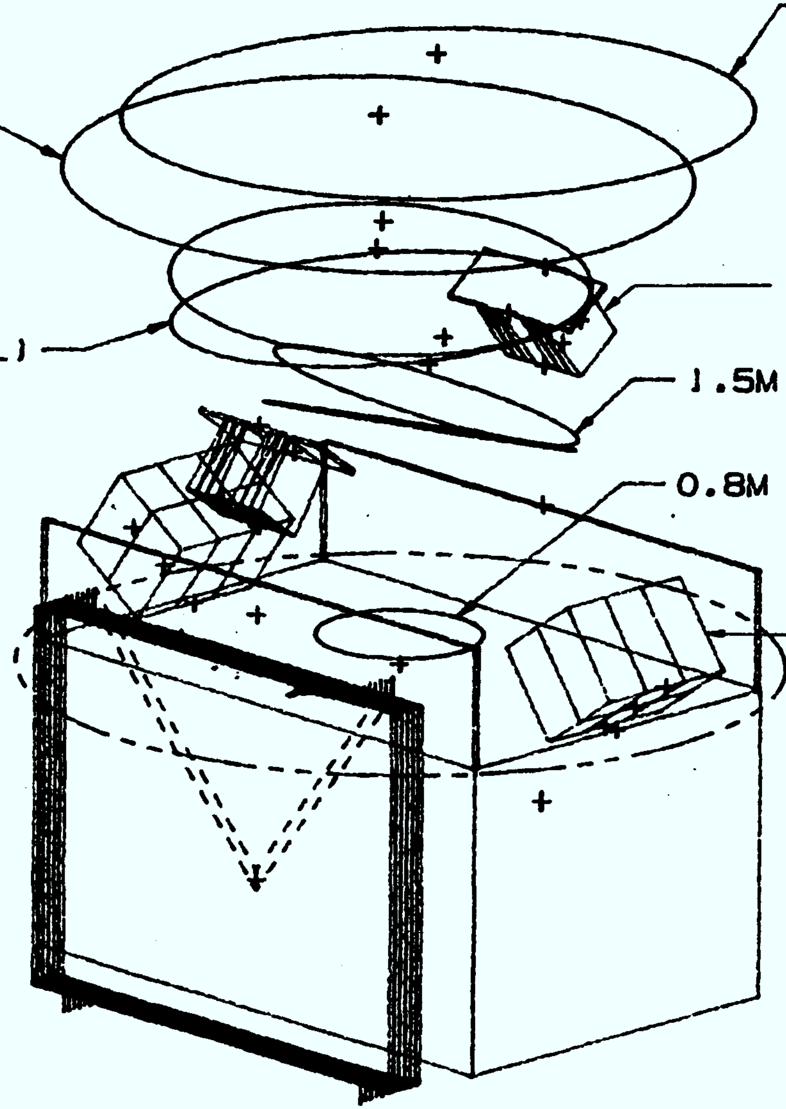
F13



SPAR

13

AUSSAT II DUAL 3-METER SOLID CONFIGURATION (STOWED)

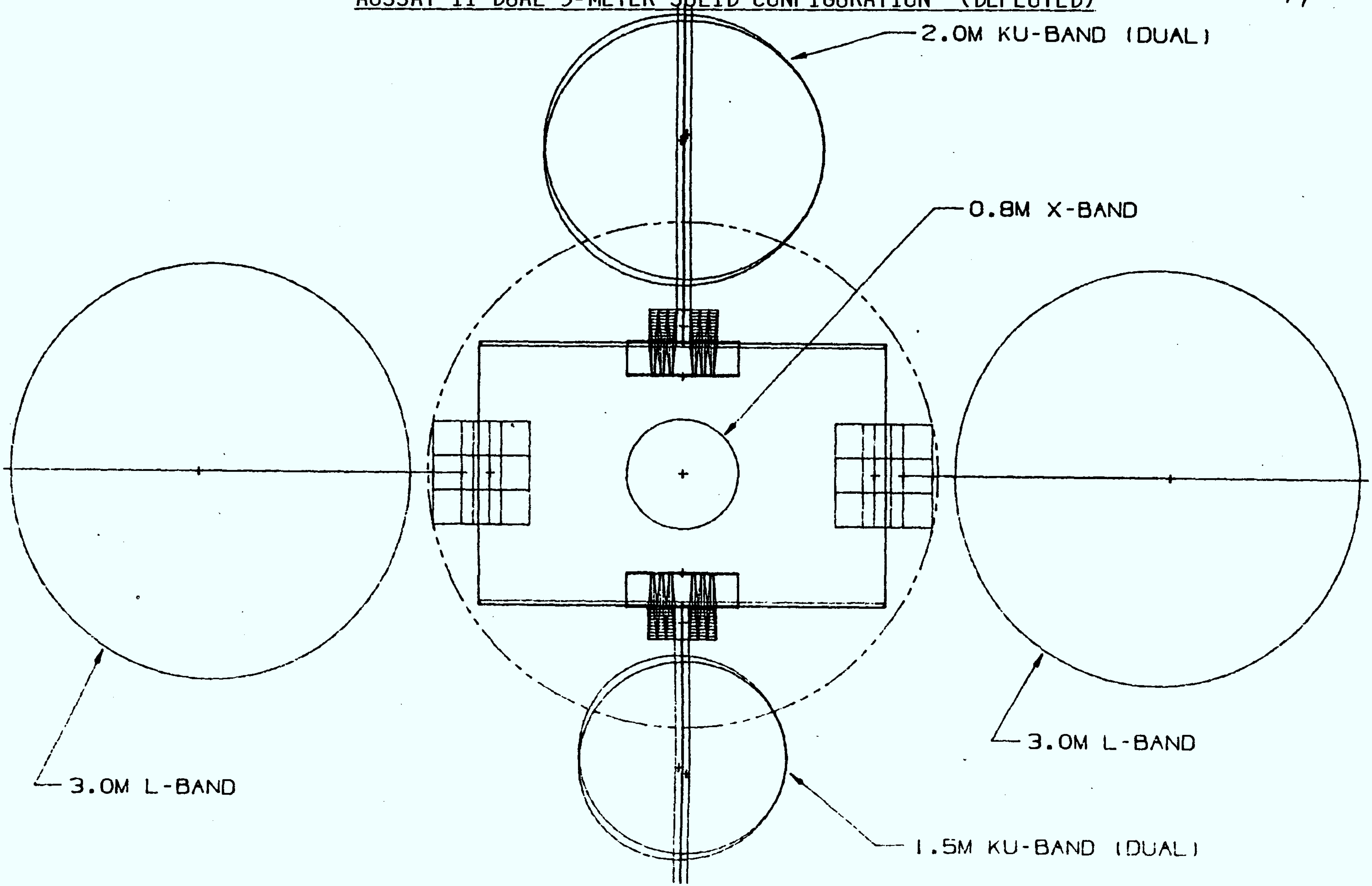


F14

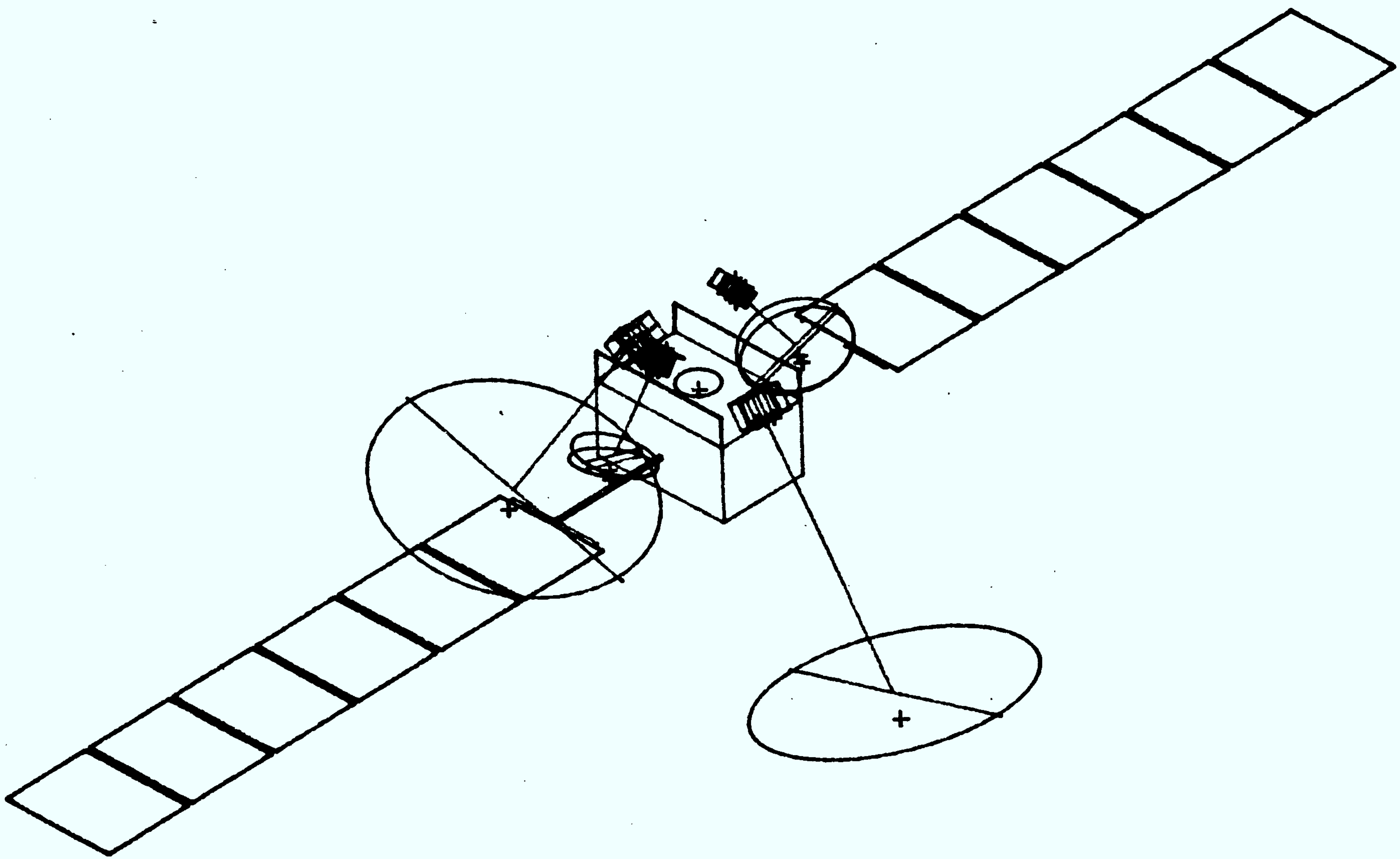
SPAR

14

AUSSAT II DUAL 3-METER SOLID CONFIGURATION (DEPLOYED)



AUSSAT II DUAL 5-METER UNFURLABLE CONFIGURATION (DEPLOYED)

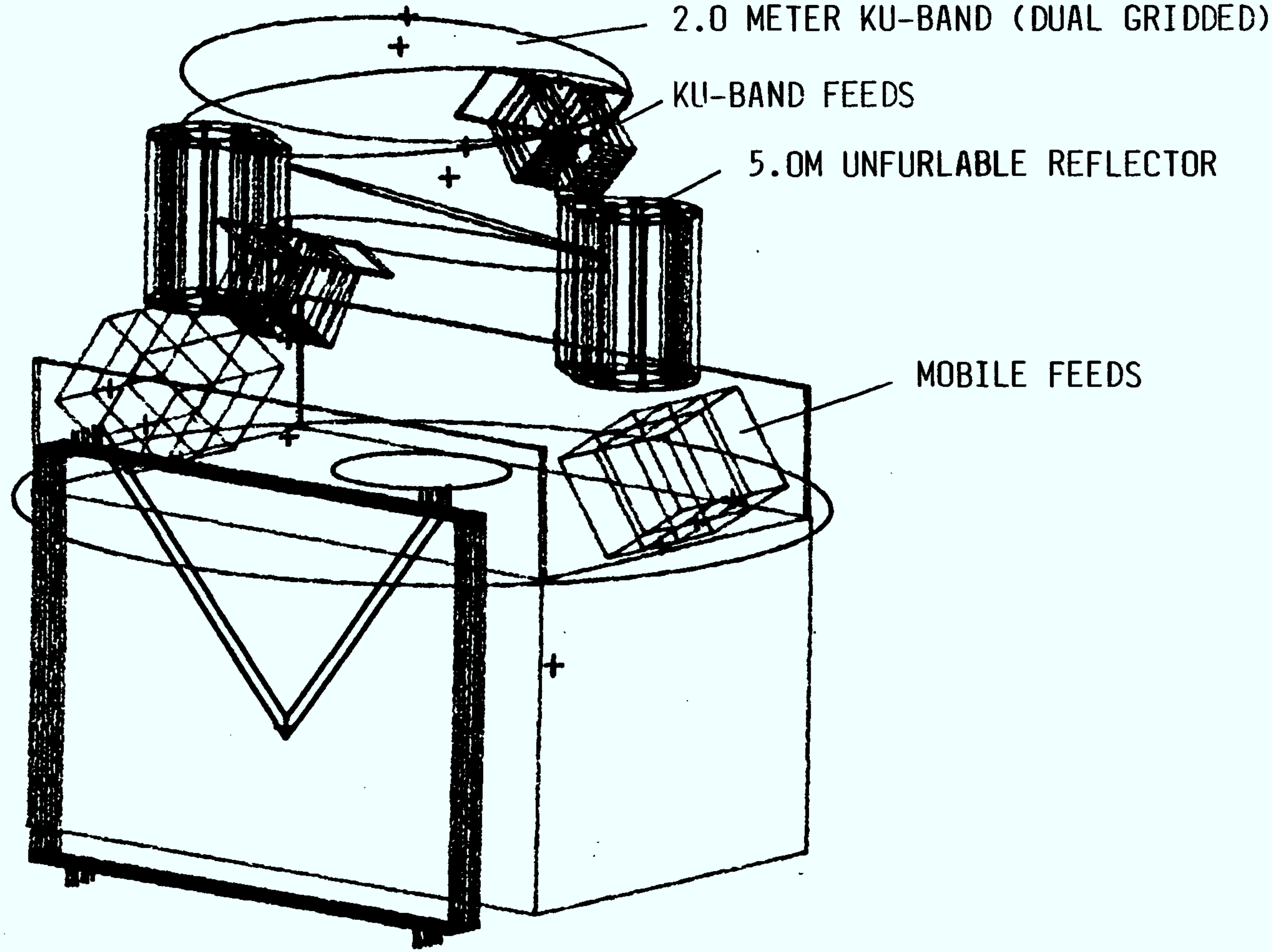


F16

SPAR

16

AUSSAT II DUAL 5-METER UNFURLABLE CONFIGURATION (STOWED)

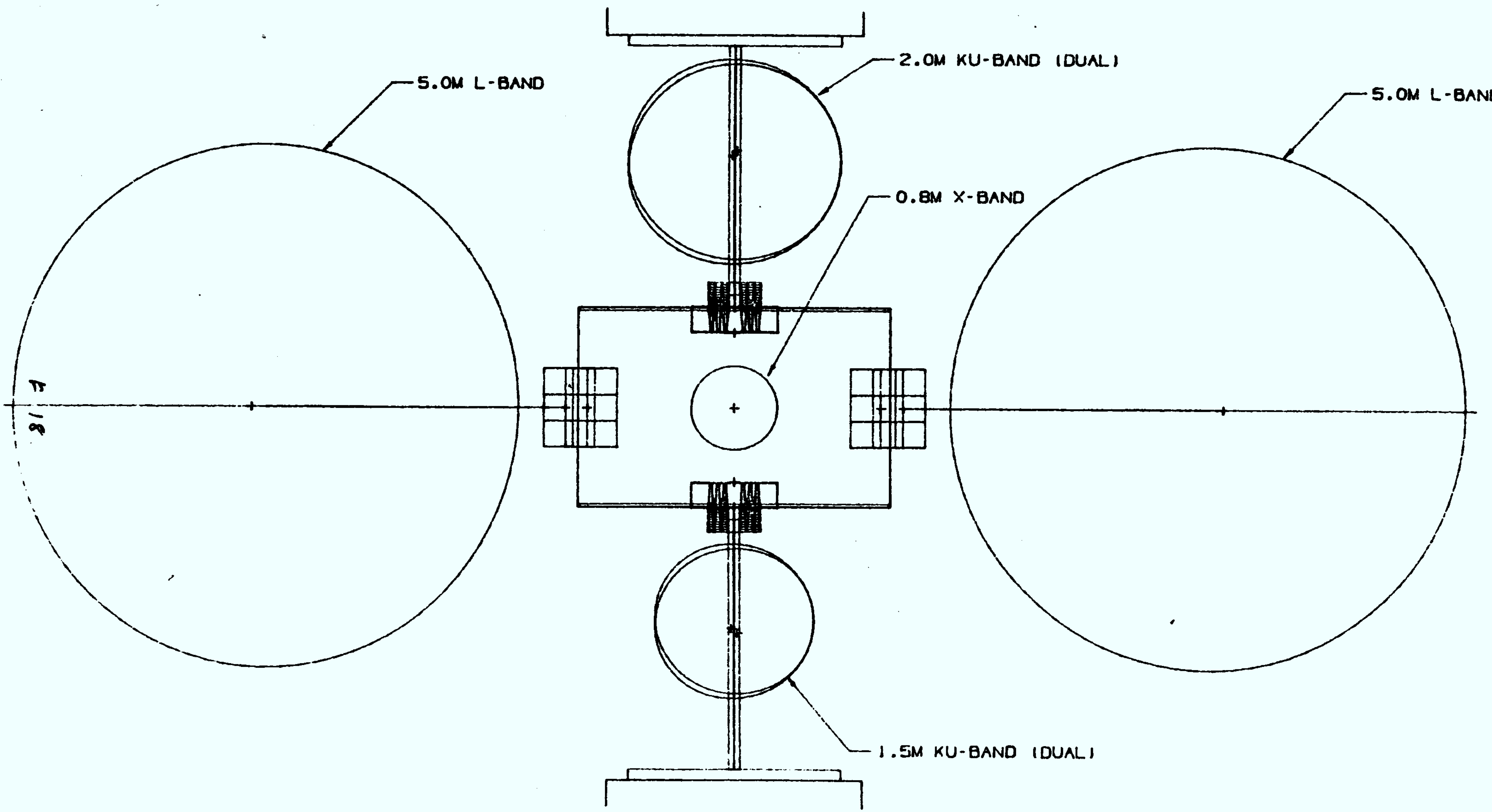


F17

SPAR

17

AUSSAT II DUAL 5-METER UNFURLABLE CONFIGURATION (DEPLOYED)



GOALS FOR RECONFIGURATION STUDY

1. ADOPT RIGID ARRAY AS PER RMSD STUDY
2. RECONFIGURE STRUCTURE TO SAVE ~100 KG FROM EXISTING OLYMPUS
3. DESIGN FOR ARIANE IV, DUAL LAUNCH COMPATIBILITY
4. RETAIN EXISTING OLYMPUS DESIGN FEATURES TO THE GREATEST EXTENT POSSIBLE
5. MAXIMIZE REUSE OF EXISTING OLYMPUS GSE

F19

LAUNCH VEHICLE COMPATIBILITY

- IN LIGHT OF THE STS ACCIDENT AND ARIANESPACE'S DEVELOPMENT LEAD, IT WOULD APPEAR THAT ARIANE IV CAN BE EXPECTED TO SET THE STANDARD FOR LAUNCH VEHICLE ENVELOPES.

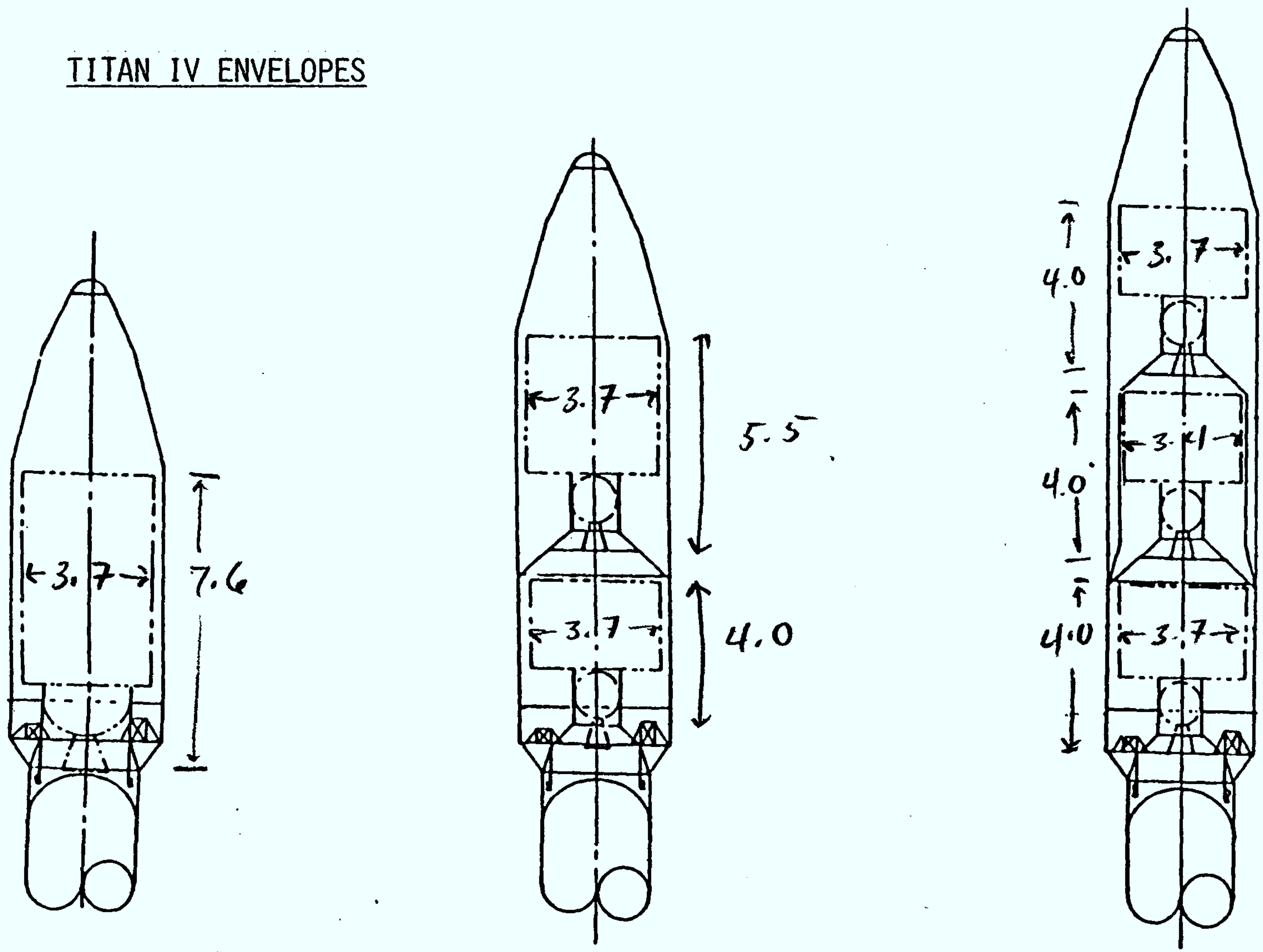
- USING THE FULL ARIANE IV DIAMETER (3.65M) RESULTS IN A MORE COMPACT PLATFORM DESIGN WHICH ALLOWS MORE HEADROOM TO STOW THE LARGE REFLECTORS REQUIRED FOR MANY PAYLOADS.

- SOME EXISTING ELV'S (SUCH AS THE TITAN IV) REQUIRE USE OF AN UPPER STAGE. A COMPACT DESIGN GREATLY EASES STOWAGE CONSTRAINTS IN SUCH CASES.

F20

F.21

TITAN IV ENVELOPES



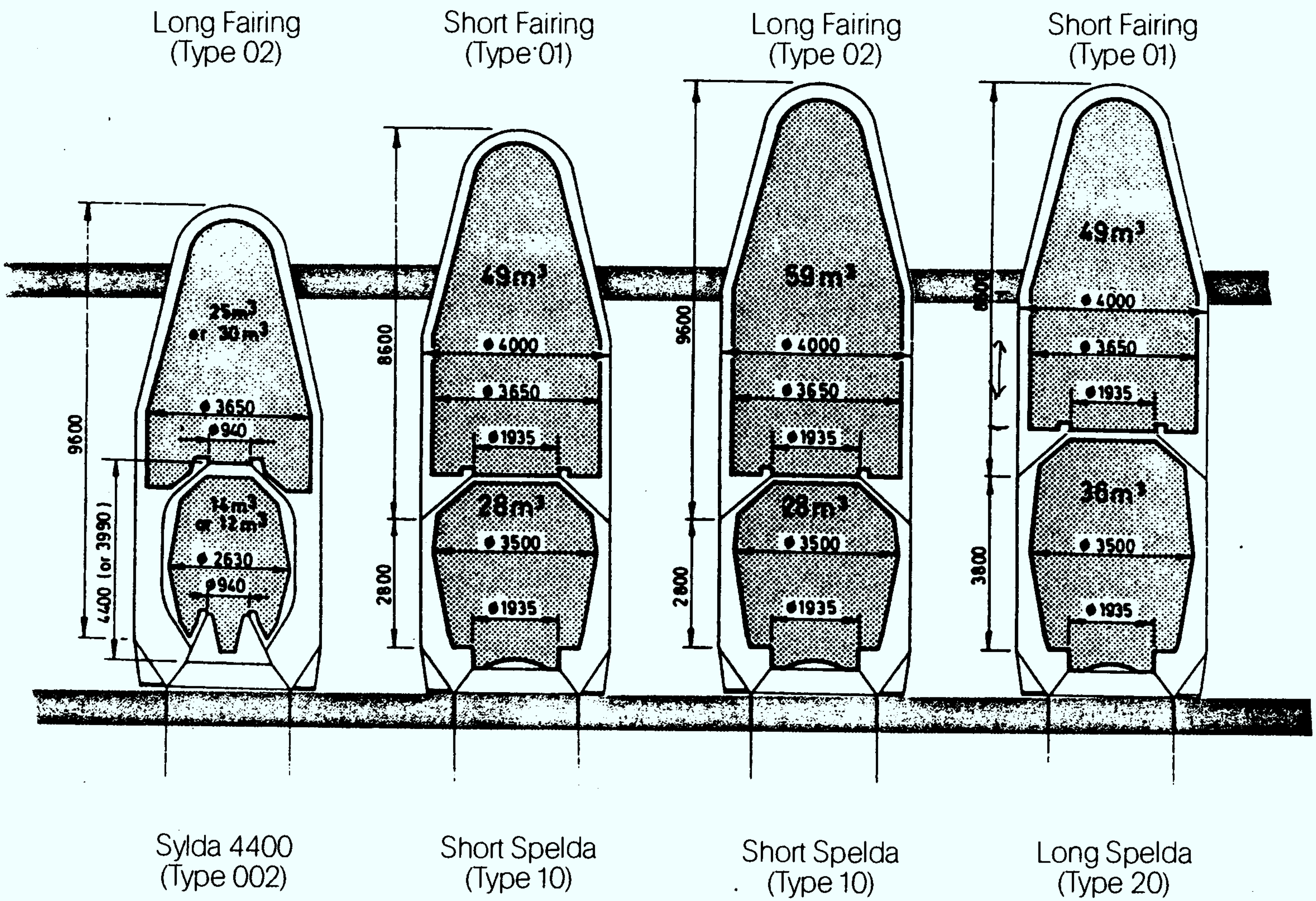
ONE PAYLOAD

TWO PAYLOADS

THREE PAYLOADS

ARIANE IV ENVELOPES

PAYLOAD COMPARTMENT CONFIGURATIONS
FOR DUAL LAUNCHES



F22

RMSD RIGID ARRAY STUDY

- SIGNIFICANT WEIGHT SAVINGS POSSIBLE
- DESIGN COMPATIBLE WITH BOTH L-1 AND MODIFIED OLYMPUS
- SOME CONSTRAINTS IMPOSED BY ESA AND BAE RESULTS IN INCREASED WEIGHT. THESE ARE: -
    - STOWED FREQUENCY
    - USE OF AEG SOLAR CELLS
    - LIMITED PYROS, RESULTING IN MORE COMPLICATED RELEASE MECHANISM

F23

RMSD ARRAY STUDY

MASS PROPERTIES:

NUMBER OF PANELS/WING	WING POWER LEVEL (EOL 10 YR)	WING MASS (KG)
1	0.349	20
2	0.698	31
3	1.048	41
4	1.397	52
5	1.746	62
6	2.095	72
7	2.445	82

F24

OLYMPUS RECONFIGURATION STUDY

OBJECTIVE OF STUDY :

- EXAMINE POSSIBLE MODIFICATIONS TO OLYMPUS WHICH WOULD ALLOW IT TO COMPETE EFFECTIVELY WITH EXISTING SPACECRAFT (EG. RCA 5000)
- RETAIN THE OLYMPUS DESIGN PHILOSOPHY (GROWTH AND MODULARITY).
- RETAIN IMPORTANT TECHNOLOGY AND SUBSYSTEMS DEVELOPED FOR OLYMPUS

F25

F26

DESIGN TARGETS

PAYLOADS :

- MANY CURRENT PAYLOADS ARE IN THE RANGE

  300 - 400 KG
  2500 - 3500 WATTS
  ARIANE IV LAUNCH

- FOR THESE PAYLOADS, THE OLYMPUS BUS IS ROUGHLY 150 - 200 KG. HEAVIER THAN THE RCA 5000.

- ONE OF THE PRINCIPAL COMPONENTS OF THIS VARIANCE IS THE STRUCTURE SUBSYSTEM WHICH IS 108 KG, OR 80%, GREATER FOR OLYMPUS

WHY THIS DIFFERENCE ?

F27

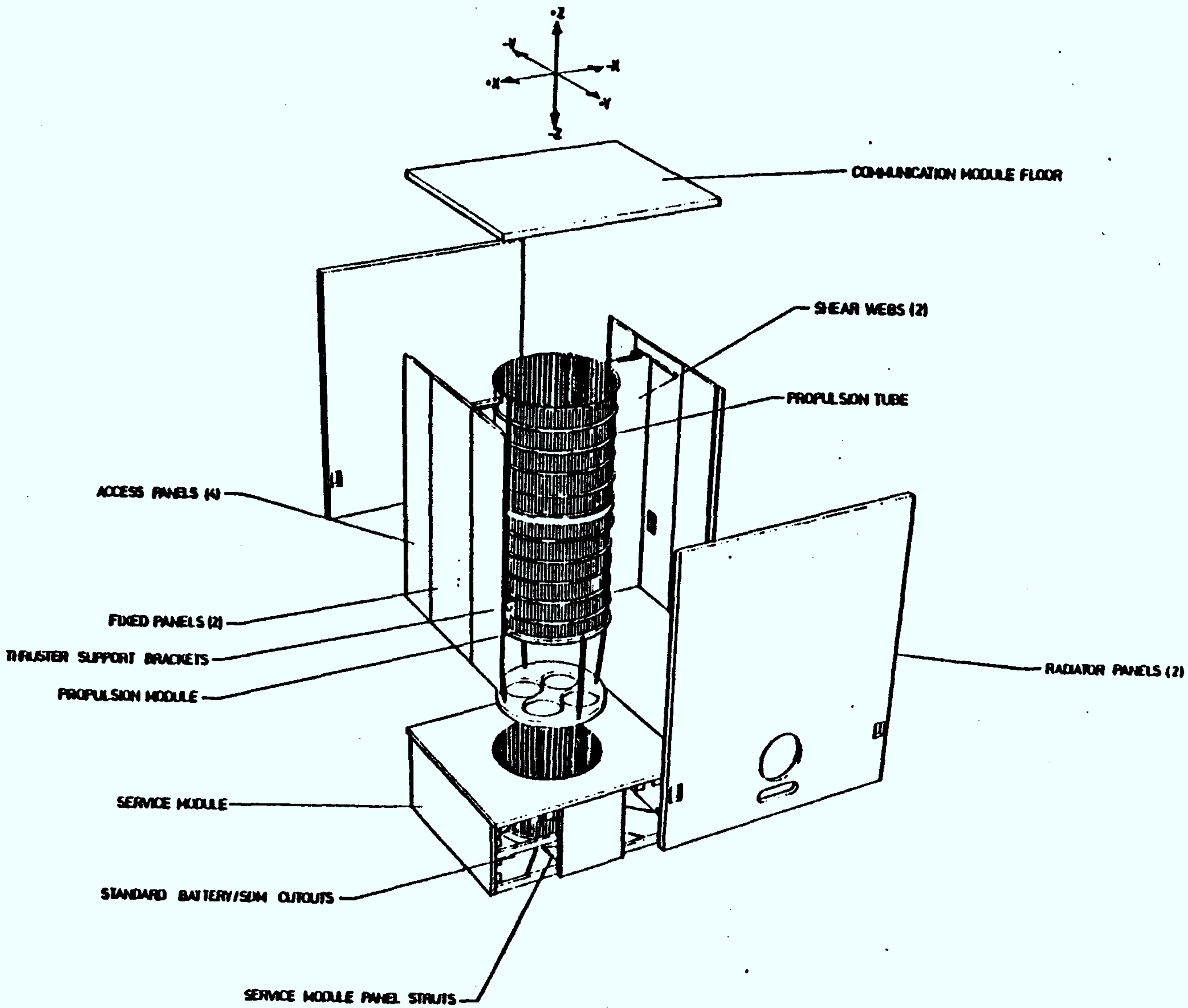
STRUCTURE / CONFIGURATION

THE CURRENT OLYMPUS LAYOUT HAS SOME SERIOUS PROBLEMS WHICH CONSPIRE TO MAKE THE SPACECRAFT VERY HEAVY FOR SMALLER PAYLOADS:

1. SIZED FOR ARIANE III LAUNCH
   - RESULTS IN A LONGER, THINNER, AND HEAVIER STRUCTURE
2. DIMENSIONED FOR MAXIMUM PAYLOAD
   - RESULTS IN HEAVIER THERMAL, ELECTRICAL DISTRIBUTION
   - SOME STRUCTURE (NOT ALL) SIZED FOR MAXIMUM P/L
3. INEFFICIENT LOAD PATHS
   - SHEAR WALLS AND THRUST TUBE TRANSITIONS
   - MUCH OF THE LOAD IS CARRIED BY PAYLOAD PANELS AND SOLAR ARRAYS.

F28

BASIC OLYMPUS STRUCTURE



OLYMPUS RECONFIGURATION
DOC PRESENTATION
SEPTEMBER 26, 1986

SASD

SPAR

STRUCTURE MODIFICATION

K.Duffy

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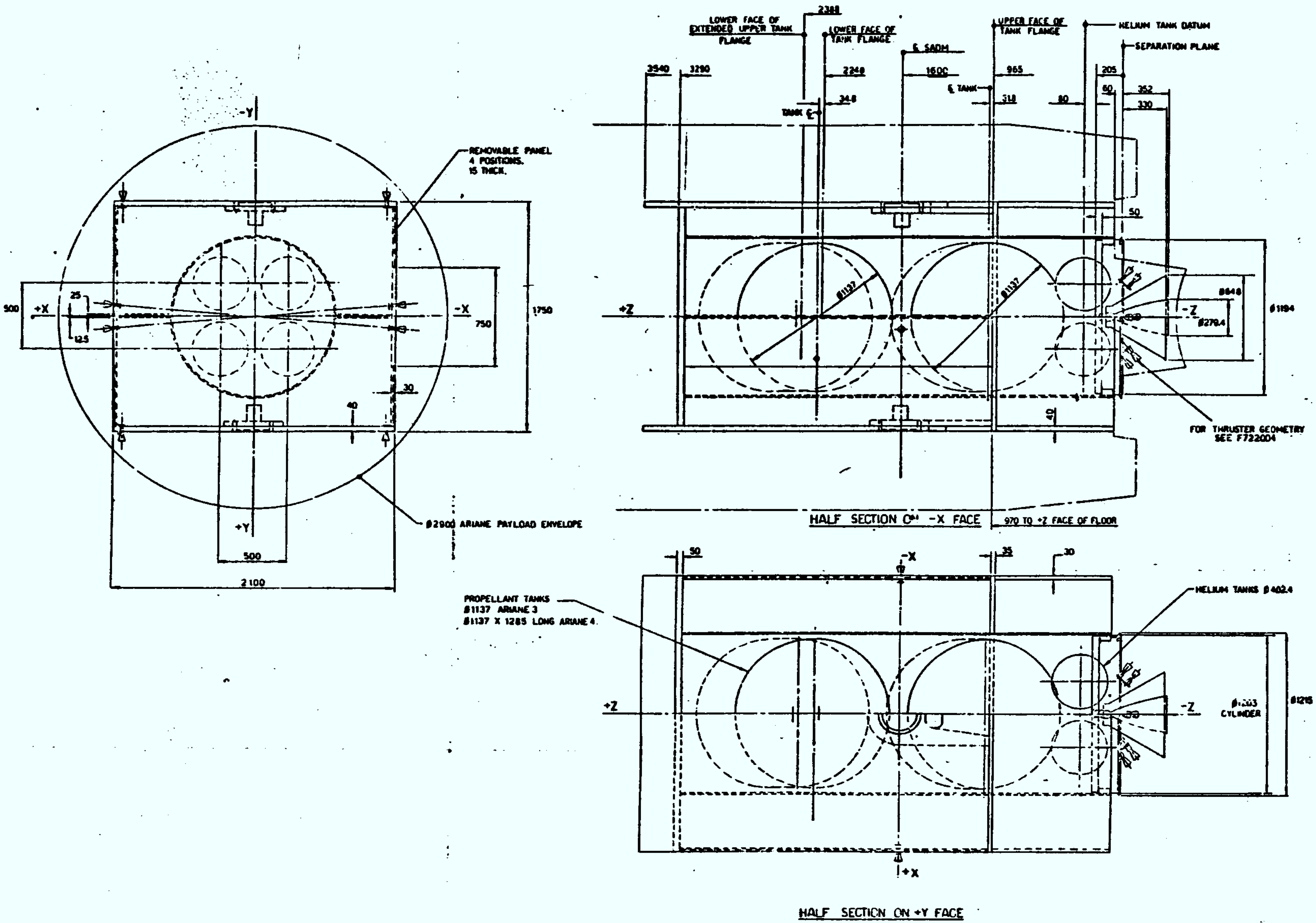
BASIC OLYMPUS STRUCTURE



F29

F30

INITIAL CONCLUSION :

- MASS DISCREPANCY CANNOT BE MADE UP BY SIMPLE TRIMMING OF THE STRUCTURE.
- STRUCTURE IS CHEAP BUT HEAVY - SPACECRAFT CANNOT AFFORD INEFFICIENT DESIGNS

⇛ BASIC RECONFIGURATION IS REQUIRED

APPROACH :

- DESIGN FOR ARIANE IV DUAL LAUNCH, UPPER POSITION
- DESIGN THE STRUCTURE TO ACCOMMODATE THE SMALLER PAYLOADS, AND A MORE NARROW RANGE OF SIZES.
- SPACECRAFT GROWTH IS ACHIEVED BY COMMONALITY OF DESIGN APPROACH AND STRUCTURAL TECHNIQUES.

OLYMPUS RECONFIGURATION

- DIMENSIONS :

  OLD - 2.1 X 1.75 X 3.54 M

  NEW - 2.9 X 1.9 X 2.2 M

- SPACECRAFT HAS BEEN "TRUNCATED" AT WHAT WAS PREVIOUSLY THE SOLAR ARRAY DRIVE LEVEL, 1.6 M. ABOVE THE SEPARATION PLANE

- X/Y DIMENSIONS HAVE BEEN INCREASED TO FILL THE ARIANE IV ENVELOPE

  ⇒ RESULTS IN A SHORTER, SQUATTER S/C WHICH IS NOT MUCH SMALLER THAN BEFORE.

- UPPER FUEL TANK REMOVED ; HYDRAZINE CARRIED IN TWO CYLINDRICAL TANKS OF EQUAL VOLUME LOCATED ADJACENT TO THE THRUST TUBE

  ⇒ GREATLY REDUCES THE BENDING MOMENTS ON THE STRUCTURE

F31

F32

DESIGN MODIFICATIONS

- THRUST TUBE
  - CHANGED TO A CONTINUOUS MEMBER STRETCHING FROM SEPARATION RING TO EARTH FACING PANEL (OLD DESIGN: BOLTED I/F AT SM FLOOR)
  - SHOULD BE CONSTRUCTED FROM HONEYCOMB (INSTEAD OF CORRUGATED SHEET) TO SIMPLIFY STRUCTURAL I/F'S
- RADIAL SHEAR WALLS
  - TOTAL OF 4 IN E/W AND N/S PLANES
  - EXTEND THE FULL LENGTH OF THE S/C
  - SUPPORT THRUST TUBE, FORMS PART OF THE CPS CENTRAL STRUCTURE

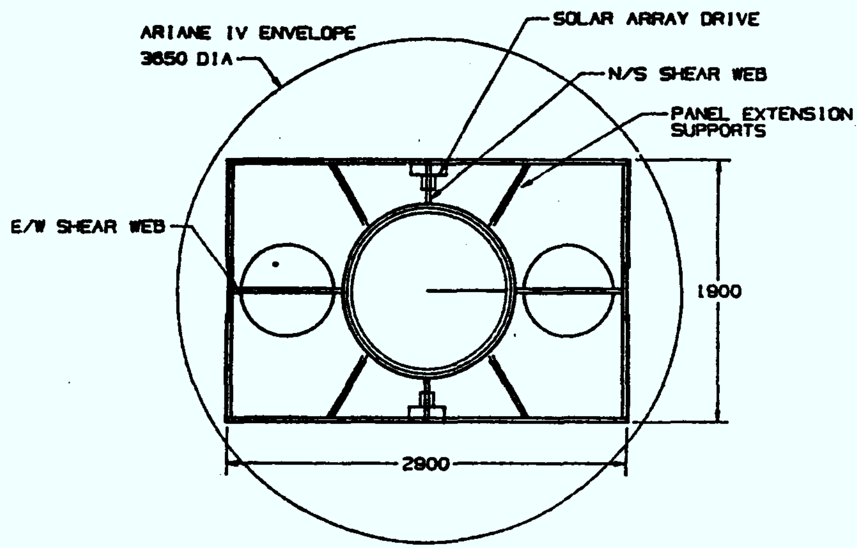
    ➡ BECOMES PRIMARY LOAD PATH
- LOWER PANEL
  - SERVICE MODULE COMPLETELY ENCLOSED
- SERVICE MODULE
  - STRUTWORK REMOVED
  - ONLY SUPPORTS BUS EQUIPMENT ; DOES NOT FROM PART OF CENTRAL STRUCTURE

F33

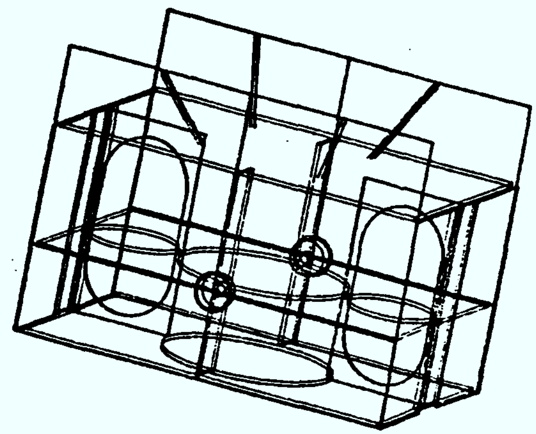
DESIGN MODIFICATIONS (CONT'D)

- HYDRAZINE TANKS
  - 1.324 X 0.662 M EACH
  - EMBEDDED IN E/W SHEAR WALLS
- PAYLOAD PANELS
  - ONE OR TWO PANELS PER SIDE (OPTIONAL)
  - VARIABLE EXTENSION ABOVE NADIR PANEL TO ALLOW GROWTH
- BUS PANELS
  - SM FLOOR RETAINED TO SUPPORT AOCS EQUIPMENT
  - 4 N/S BUS PANELS
- BATTERIES
  - MOUNT DIRECTLY TO BUS PANELS (INTEGRAL RADIATOR)
  - SHOULD NOT BE THERMALLY ISOLATED FROM S/C INTERIOR (REDUCE RADIATOR REQUIREMENTS)
- MATERIALS
  - IT IS RECOMMENDED THAT MATERIALS SUCH AS BERYLLIUM AND GFEC BE USED MORE EXTENSIVELY IN THE OLYMPUS DESIGN, IN ORDER TO SAVE MASS.

PLATFORM GEOMETRY : RECONFIGURED OLYMPUS

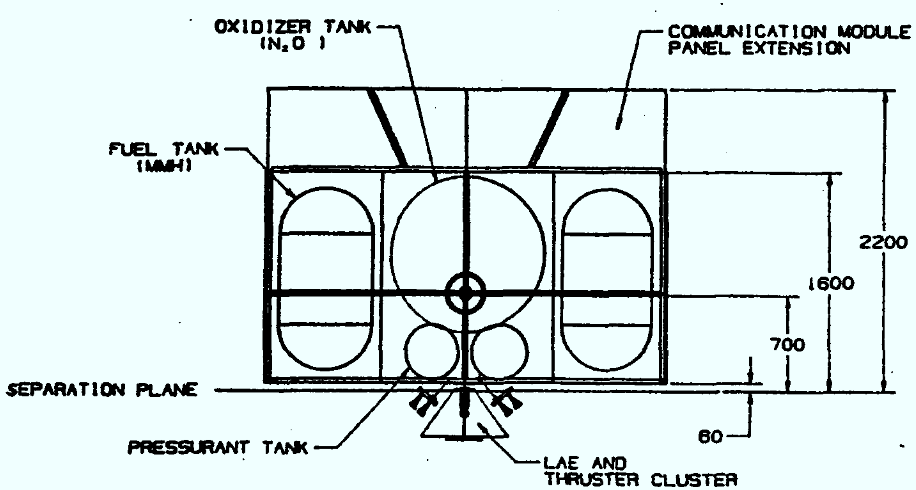


EARTH-FACING VIEW

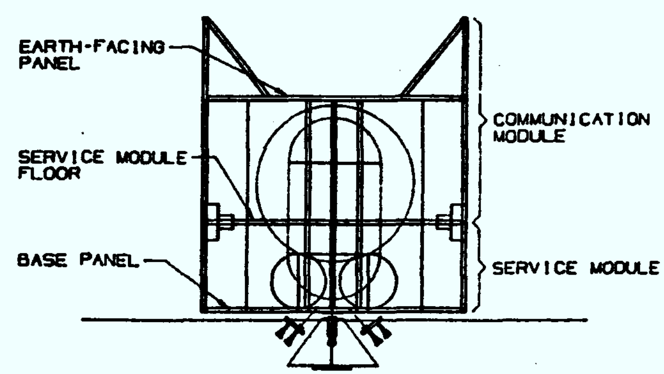


ISOMETRIC VIEW
(PROPULSION MODULE REMOVED)

F34



NORTH/SOUTH VIEW



EAST/WEST VIEW

SOLAR ARRAY ACCOMMODATION

- EXISTING RMSD RIGID PANEL SOLAR ARRAY HAS BEEN ASSUMED (2.4 X 1.9 M. PANELS)

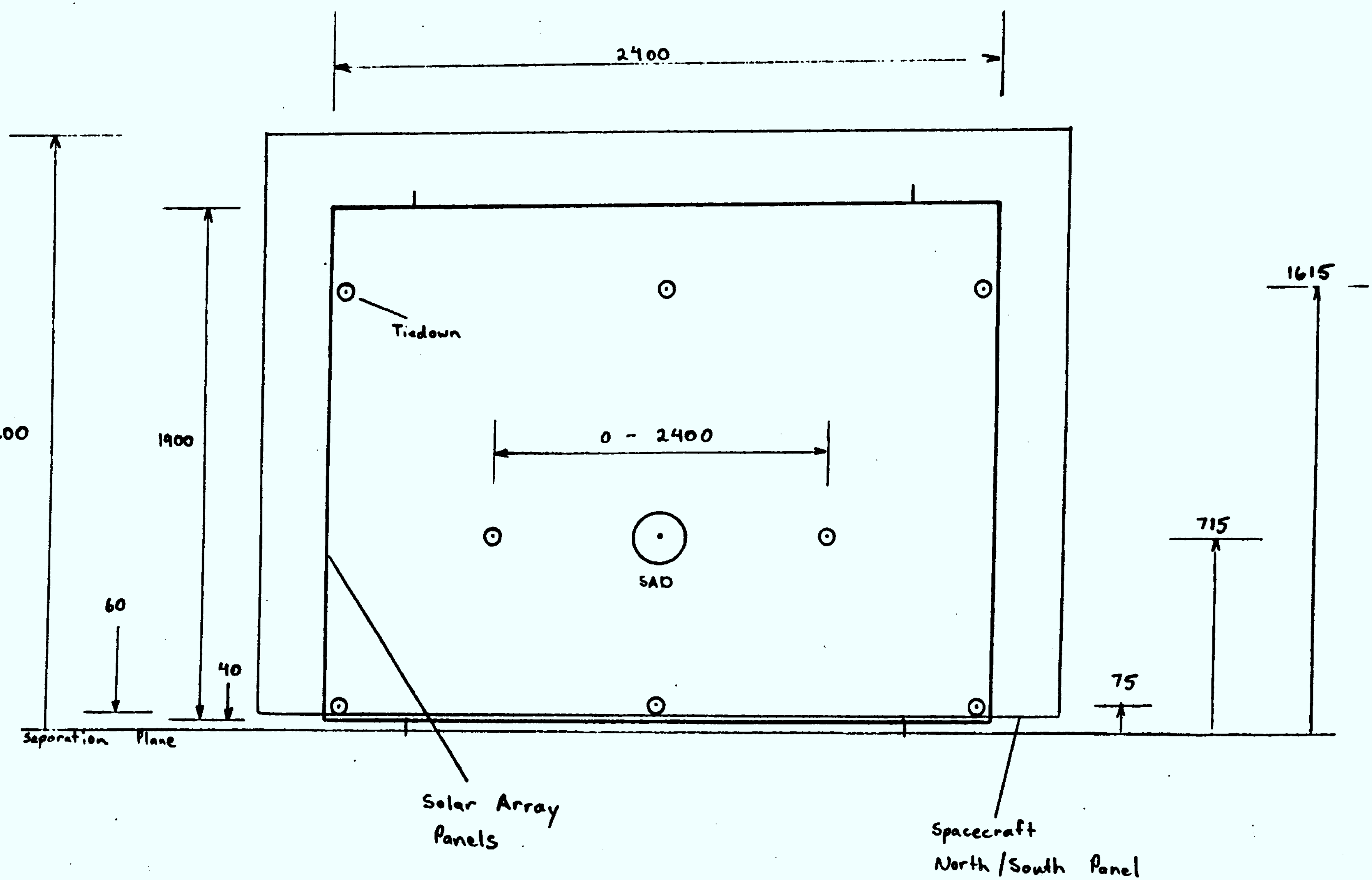
- UP TO 6 SA. PANELS CAN BE STOWED ON THE N/S FACES

- SOLAR ARRAY DRIVE IS MOUNTED DIRECTLY TO THE CENTRAL STRUCTURE (NOT THE PAYLOAD PANELS)

- TIEDOWNS ARE MADE DIRECTLY TO THE FIXED STRUCTURE (SHEAR WALLS AND FLOORS) ; THE SCHEME CAN BE MADE STATICALLY INDETERMINATE WITHOUT EXCESSIVE LOAD TRANSFER TO THE ARRAYS.

F35

SOLAR ARRAY TIEDOWN SCHEME



F36

RECONFIGURED OLYMPUS:

STRUCTURAL MASS ESTIMATES

THRUST TUBE & SEPARATION RING	35	
E/W SHEAR WALLS	10	
N/S SHEAR WALLS	5	
N/S PAYLOAD PANELS	30	
N/S BUS PANELS	15	
UPPER FLOOR (CM)	12	
MIDDLE FLOOR (SM)	10	
LOWER FLOOR (SM)	8	
CPS MOTOR FLOOR	8	
ACCESS PANELS	6	
MISCELLANEOUS STRUCTURAL EQUIPMENT	20	
TOTAL	159	KG
PREVIOUS OLYMPUS STRUCTURE MASS	243	KG
SAVINGS	84	KG

F37

F38

STRUCTURAL MASS COMPARISON

RCA 5000		OLYMPUS		RECONFIGURED OLYMPUS	
CENTER STRUCTURE + BULKHEADS + TANK SUPPORT	43.1	CYLINDERS + MOTOR FLOOR + E/W STRUCTURE	95.8	THRUST TUBE + SEPARATION RING + SHEAR WALLS	58
PAYLOAD PANELS	34.8	PAYLOAD PANELS	57.3	PAYLOAD PANELS	42
BUS PANELS	20.8	SERVICE MODULE	57.9	BUS PANELS	33
OTHER	36.6	OTHER	32.0	OTHER	26
- ACCESS PANELS REFLECTOR MUX SHELVES INSERTS & BRACKETS ENGINE & THRUSTER SUPPORTS		- ACCESS PANELS CLIPS & SUPPORTS BRACKETS (SAD, THRUSTERS, ETC) BALANCE RAILS FASTERNERS & CONNECTIONS		- ACCESS PANELS ETC.	
TOTAL	135.3		243.0		159

F39

PAYLOAD MODELS

VARIOUS OPTIONS FOR THE AUSSAT II PAYLOAD WERE USED IN AN ACCOMMODATION EXERCIZE FOR THE RECONFIGURED OLYMPUS

KU-BAND BASELINE

- ONE 2.0 M. DUAL POLARIZED REFLECTOR
- ONE 1.5 M DUAL POLARIZED REFLECTOR

X-BAND MILITARY

- ONE 0.8 M REFLECTOR

MOBILE PAYLOAD

- ONE OR TWO 3.0 M SOLID REFLECTORS (L-BAND)

  OR
- TWO 5.0 M DEPLOYABLE MESH REFLECTORS (UHF)

SUMMARY OF NEW DESIGN

- o MORE COMPACT
  - MORE EFFICIENT STRUCTURE, THERMAL CONTROL, ELECTRICAL AND RF DISTRIBUTION
  - EASIER TO HANDLE AND TEST.
  - LEAVES MUCH MORE VOLUME AVAILABLE FOR STOWING ANTENNAS WITHIN LAUNCHER ENVELOPE (ESPECIALLY ABOVE EARTH FACING PANEL)
- o MORE EFFICIENT STRUCTURE DESIGN
  - EASIER TO ASSEMBLY AND ANALYZE
  - FEWER INTERFACES
  - LOWER MASS
- o ALLOWS FOR GROWTH
  - EXTENDIBLE RADIATOR PANELS
  - SIMILAR DESIGN APPROACH COULD BE USED FOR MUCH LARGER STRUCTURES.
- o RETAINS EXISTING OLYMPUS TECHNOLOGY
  - STRUCTURE CONCEPT IS DERIVATIVE
  - MINIMUM MODIFICATIONS TO VIRTUALLY ALL OTHER SYSTEMS.

F40

GSE COMPATIBILITY

- ALL MAJOR OLYMPUS GSE WILL BE APPLICABLE TO THE MODIFIED OLYMPUS WITHOUT MODIFICATION OR CONSTRAINTS EXCEPT FOR THE SPACECRAFT CONTAINER AND THE SPACECRAFT HANDLING TROLLEY.

- THE HANDLING TROLLEY CAN BE USED WITH THE SPACECRAFT IS ROTATED WITH THE Z-AXIS VERTICAL.

F41

GSE COMPATIBILITY (CONT'D)

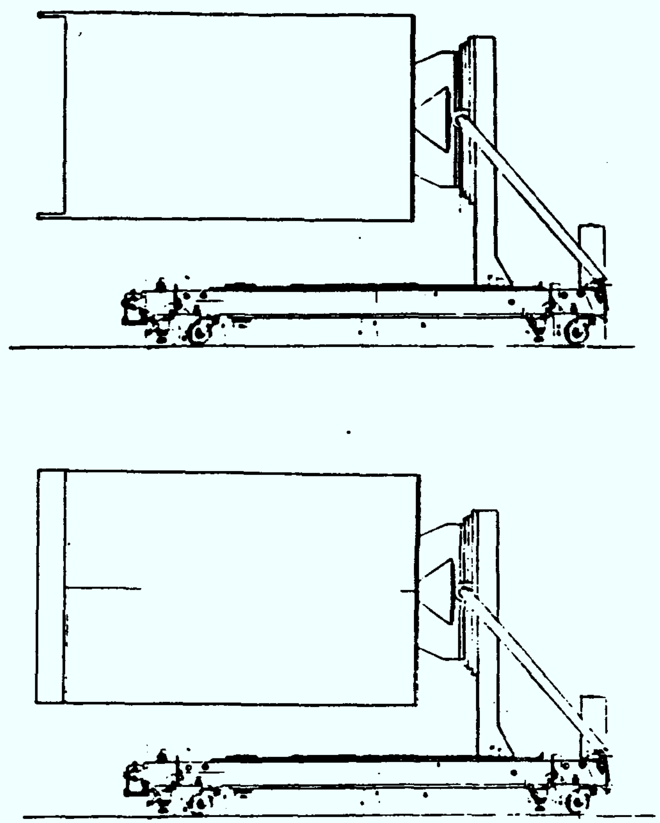
O SEVERAL MINOR ADAPTORS, BRACKETS, SLINGS WHICH ARE SPECIFIC TO OLYMPUS MODULE DIMENSIONS WOULD HAVE TO BE MODIFIED OR REDESIGNED. THIS INCLUDES SUCH ITEMS AS:

- SERVICE MODULE ADAPTOR FRAME
- N/S PANEL ADAPTOR
- S/C HORIZONTAL SLING
- N/S PANEL SLING
- ANTENNA FLOOR ADAPTOR
- EIM RELATED GSE
- SOLAR ARRAY GSE

F42

GSE COMPATIBILITY (CONT'D)

SPACECRAFT HANDLING TROLLY WITH MODIFIED OLYMPUS



F43

GSE COMPATIBILITY (CONT'D)

THE OVERALL COST IMPACT OF THESE CHANGES ON A SPECIFIC PROGRAM SHOULD BE MINOR.

IT SHOULD BE NOTED THAT MUCH OF THE GSE FOR OLYMPUS WOULD HAVE TO BE DUPLICATED OR REPLACED IF MULTIPLE S/C WERE INTEGRATED AT THE SAME TIME IN ANY EVENT. ALSO SPECIFIC PAYLOAD REQUIREMENTS DICTATE THE DESIGN OF SUCH GSE AS THE SPACECRAFT CONTAINER AND LIFTING SLINGS AND BRACKETS IN ANY EVENT.

F44

F45

CONCLUSIONS

- WITH STRUCTURAL AND SOLAR ARRAY MODIFICATIONS AS DESCRIBED IN THIS STUDY, THE OLYMPUS PLATFORM HAS THE CAPABILITY OF COMPETING DIRECTLY AGAINST ALL EXISTING NORTH AMERICAN AND EUROPEAN PLATFORMS.

  SUCH MODIFICATIONS CAN BE CARRIED OUT IN A WAY TO RETAIN THE ESSENTIAL DESIGN CHARACTERISTICS OF THE OLYMPUS PLATFORM AND PRESERVE COMPATIBILITY WITH EXISTING GSE.

- ANY STRUCTURE MODIFICATION SHOULD BE VERIFIED BY A COMBINATION OF ANALYSIS AND PROTOFLIGHT TESTING/ NO QUALIFICATION STRUCTURE SHOULD BE REQUIRED SINCE THE EXTENSIVE OLYMPUS TEST PROGRAM HAS PROVEN ANALYSIS TECHNIQUES.

- IF THE ABOVE PROCEDURE IS ADOPTED, THE COST AND SCHEDULE IMPACT OF THE STRUCTURAL MODIFICATION CAN BE MINIMIZED.

SPAR

SECTION G: INPUT VIEWGRAPHS TO PMC PRESENTATION AT STEVENAGE, OCTOBER 13, 1986

STUDY GOALS AND DRIVERS

- THE BASIC PURPOSE OF THIS STUDY IS TO EXAMINE METHODS OF INCREASING THE COMPETITIVENESS OF THE OLYMPUS SPACECRAFT FOR PAYLOADS IN THE MEDIUM RANGE (MASS: 300 - 400 KG; POWER 2500 - 3500W).

  THIS IS THE RANGE MOST PAYLOADS ARE EXPECTED TO BE WITHIN THE NEXT DECADE. (REF: BAE STUDY)

- THE NEED FOR SUCH AN EFFORT CAN BE SEEN WHEN THE OLYMPUS L-1 MISSION IS COMPARED TO THE ANIK E ON THE RCA 5000.

		OLYMPUS L-1	NORTH AMERICAN PLATFORM (ANIK E)
PAYLOAD:	MASS	335 KG	337 KG
	POWER	2850 WATTS	2100 WATTS
	ECLIPSE POWER	1110 WATTS	2100 WATTS
LAUNCH VEHICLE		DEDICATED ARIANE III (2420 KG GTO)	DUAL LAUNCH ARIANE IV (2500 KG GTO)
MISSION LIFE		5 YEARS	10 YEARS + 2 YEARS IN ORBIT STORAGE

THE NORTH AMERICAN PLATFORM CLEARLY OFFERS GREATER ECLIPSE POWER AND LIFE WHILE USING ALMOST IDENTICAL LAUNCH RESOURCES.

GOALS AND DRIVERS (CONTINUED)

- WHEN THE OLYMPUS SPACECRAFT IS COMPARED TO NORTH AMERICAN SPACECRAFT OF THE 300 - 400 KG/2500 - 3500 WATT RANGE, TWO SUBSYSTEMS STAND OUT AS VERY HEAVY IN COMPARISON. THESE ARE STRUCTURE AND POWER. THE OLYMPUS STRUCTURE DELTA IS ABOUT 100 KG AND THE OLYMPUS SOLAR ARRAY ALONE HAS A 90 KG DELTA. THE REMAINING OLYMPUS POWER SUBSYSTEM IS ABOUT 60 KG HEAVIER THAN AN EQUIVALENT NORTH AMERICAN SYSTEM

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OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

GOALS AND DRIVERS (CONTINUED)

- THREE TARGET PAYLOADS USED IN STUDY:

  ANIK E
  AUSSAT II
  NORTH AMERICAN MSAT

- ALL OF THE ABOVE PAYLOADS CAN BE EXPECTED TO REQUIRE A DUAL LAUNCH ON AN ARIANE IV. THIS LEADS TO THE FOLLOWING BUS REQUIREMENT.

  - LAUNCH CAPACITY OF ARIANE IV DUAL LAUNCH IS 2500 KG INTO GTO.

  - AVERAGE PAYLOAD REQUIREMENTS ARE 400 KG AND 2500 WATTS.

- THE ABOVE DATA, COMBINED WITH ASSUMPTION OF LIQUID APOGEE ENGINE, BIPROP CPS, AND 10 YEAR LIFE, LEAVE A MAXIMUM ALLOWANCE OF 796 KG FOR THE PLATFORM WITH A 44 KG MARGIN. THE SPACECRAFT POWER MUST BE SUPPLIED USING THIS MASS.

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OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

POWER SUBSYSTEM

- STUDIES MADE DURING THE RADARSAT PROGRAM AT SPAR HAVE SHOWN THAT SUBSTANTIAL MASS SAVINGS (ABOUT 30 KG) WOULD BE POSSIBLE BY ALLOWING THE PAYLOAD UNITS THE OPTION AT UNREGULATED POWER, WHILE CONTINUING TO SUPPLY 50 VDC REGULATED POWER TO THE PLATFORM UNITS

  THE ISSUE OF WHETHER THE BATTERIES NEED TO HAVE DEDICATED RADIATOR PANELS SHOULD ALSO BE ADDRESSED

65

LAUNCH VEHICLE COMPATIBILITY

- IN LIGHT OF THE STS ACCIDENT AND ARIANCESPACE'S DEVELOPMENT LEAD, IT WOULD APPEAR THAT ARIANE IV CAN BE EXPECTED TO SET THE STANDARD FOR LAUNCH VEHICLE ENVELOPES.
- USING THE FULL ARIANE IV DIAMETER (3.65M) RESULTS IN A MORE COMPACT PLATFORM DESIGN WHICH ALLOWS MORE HEADROOM TO STOW THE LARGE REFLECTORS REQUIRED FOR MANY PAYLOADS.
- SOME EXISTING ELV'S (SUCH AS THE TITAN IV) REQUIRE USE OF AN UPPER STAGE. A COMPACT DESIGN GREATLY EASES STOWAGE CONSTRAINTS IN SUCH CASES.

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STRUCTURE/CONFIGURATION

THE CURRENT OLYMPUS ARCHITECTURE MAKES THE SPACECRAFT HEAVIER FOR SMALLER PAYLOADS:

1. SIZED FOR ARIANE III LAUNCH
   - RESULTS IN A LONGER, THINNER, AND HEAVIER STRUCTURE
2. DIMENSIONED FOR MAXIMUM PAYLOAD
   - RESULTS IN HEAVIER THERMAL, ELECTRICAL DISTRIBUTION
   - SOME STRUCTURE (NOT ALL) SIZED FOR MAXIMUM P/L
3. DIFFICULT LOAD PATHS
   - COMMUNICATION/SERVICE MODULE TRANSITIONS
   - MUCH OF THE BENDING LOAD IS IMPARTED TO PAYLOAD PANELS AND SOLAR ARRAYS.

INITIAL CONCLUSIONS:

- TWO POSSIBLE WEIGHT REDUCTION SCENARIOS CAN BE ENVISAGED
  - SUBSYSTEM WEIGHT SAVING CAMPAIGN BY USE OF LIGHTWEIGHT MATERIALS, DESIGN OPTIMIZATION, ETC.
  - RECONFIGURATION OF STRUCTURE SUBSYSTEM
- IT IS UNLIKELY THAT SUBSYSTEM WEIGHT SAVING CAMPAIGN COULD PROVIDE THE DESIRED 180 KG REDUCTION, THEREFORE, THE SPAR STUDY FOCUSED IN ON THE RECONFIGURATION SCENARIO.

APPROACH FOR THE RECONFIGURATION STUDY

- DESIGN FOR ARIANE IV DUAL LAUNCH, UPPER POSITION
- DESIGN THE STRUCTURE TO ACCOMMODATE THE SMALLER PAYLOADS, AND A MORE NARROW RANGE OF SIZES.
- SPACECRAFT GROWTH IS ACHIEVED BY COMMONALITY OF DESIGN APPROACH AND STRUCTURAL TECHNIQUES.

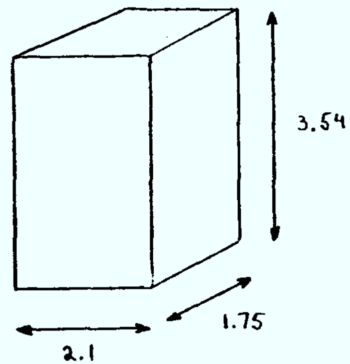
OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

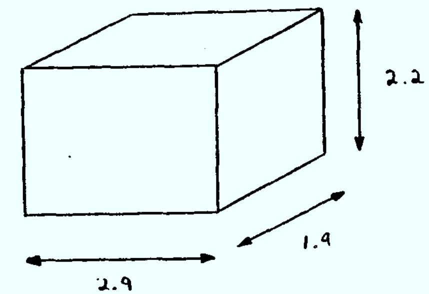
OLYMPUS RECONFIGURATION

- DIMENSIONS:

OLYMPUS -



PROPOSED -



- SPACECRAFT HAS BEEN "TRUNCATED" AT WHAT WAS PREVIOUSLY THE SOLAR ARRAY DRIVE LEVEL, 1.6M. ABOVE THE SEPARATION PLANE

- X/Y DIMENSIONS INCREASED TO FILL THE ARIANE IV ENVELOPE

- UPPER FUEL TANK REMOVED ; HYDRAZINE CARRIED IN TWO CYLINDRICAL TANKS OF EQUAL VOLUME LOCATED ADJACENT TO THE THRUST TUBE.

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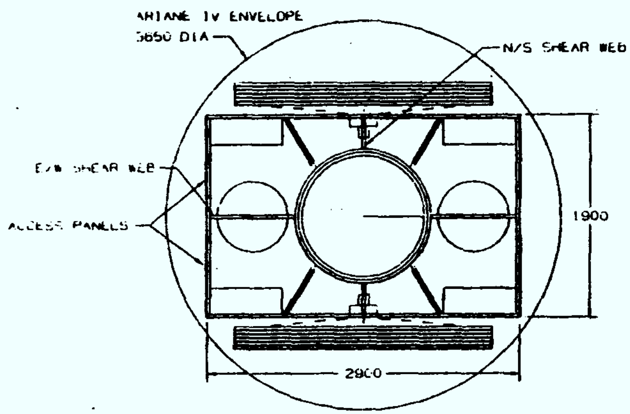
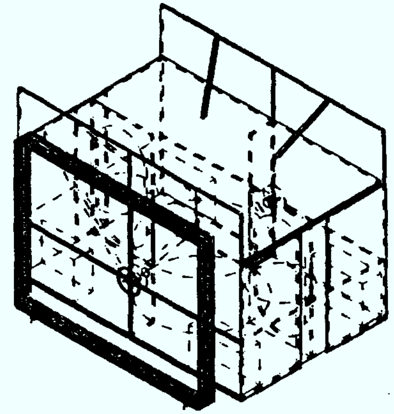
PLATFORM GEOMETRY : RECONFIGURED OLYMPUS

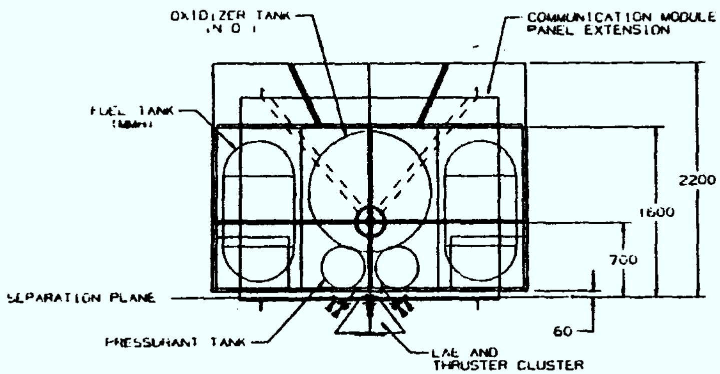


EARTH-FACING VIEW

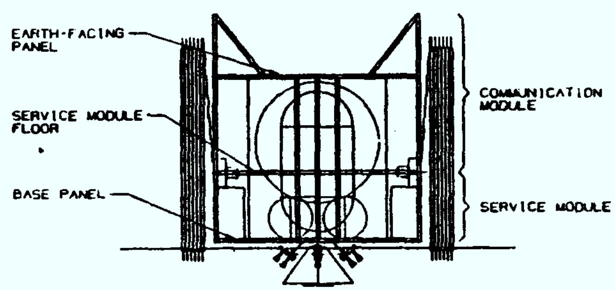


ISOMETRIC VIEW
(PROPULSION MODULE REMOVED)

G10



NORTH/SOUTH VIEW



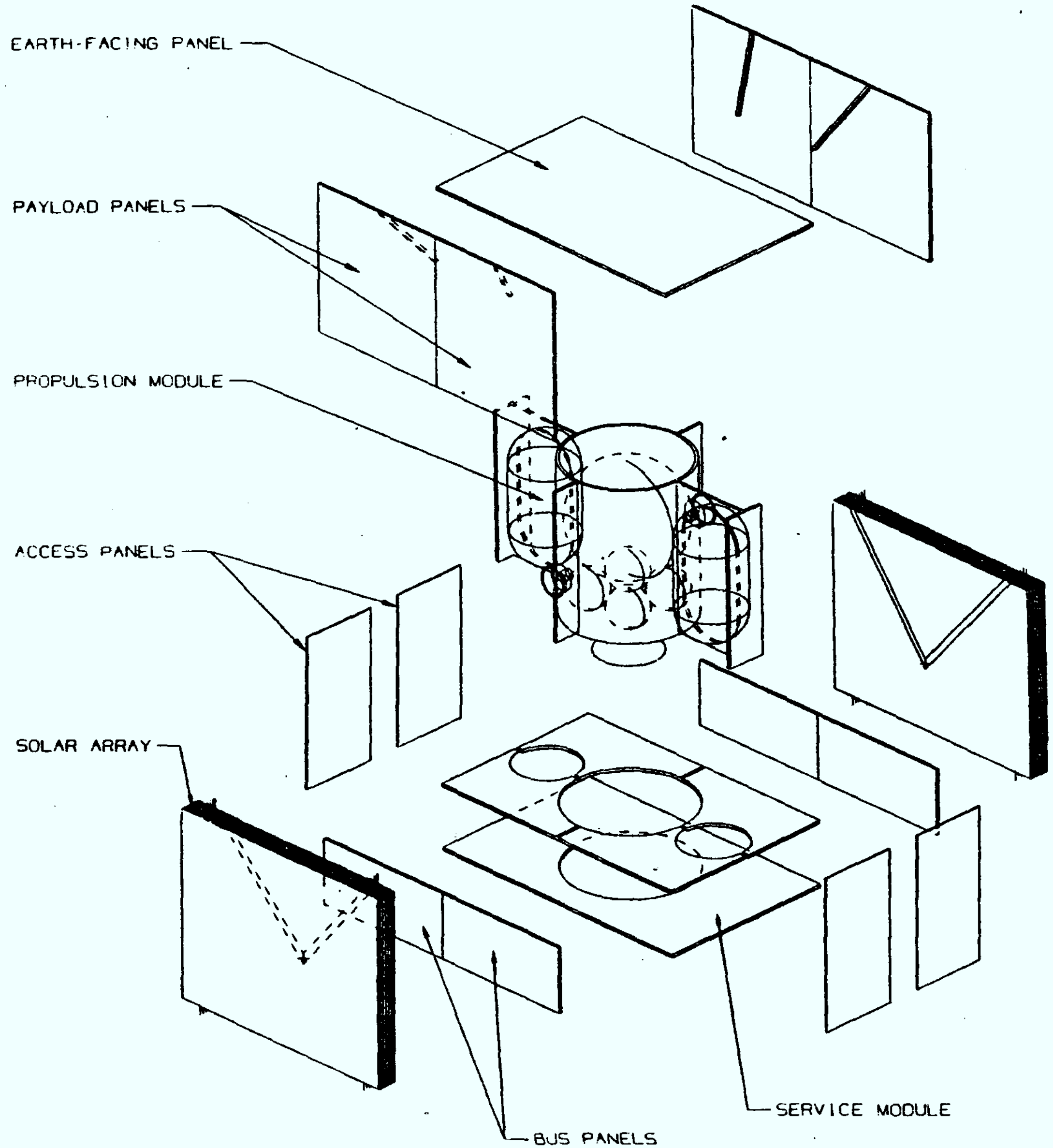
EAST/WEST VIEW

OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

RECONFIGURED OLYMPUS

PLATFORM GEOMETRY :



DESIGN MODIFICATIONS

- PROPULSION MODULE:
  - THRUST TUBE BECOMES A CONTINUOUS MEMBER EXTENDING FROM SEPARATION RING TO EARTH FACING PANEL.
  - FOUR RADIAL SHEAR WALLS RUNNING THE FULL LENGTH OF THE SPACECRAFT SUPPORT THE THRUST TUBE.
  - TWO HYDRAZINE TANKS (1.32 X 0.66 M EACH) EMBEDDED IN E/W SHEAR WALLS
  - PM DESIGNED AS CENTRAL STRUCTURE AND PRIMARY LOAD PATH
- COMMUNICATION MODULE:
  - ONE OR TWO PAYLOAD PANELS PER NORTH/SOUTH FACE
  - VARIABLE EXTENSION ABOVE NADIR PANEL TO ALLOW GROWTH
  - EAST/WEST ACCESS PANELS

5/12

DESIGN MODIFICATIONS (CONT'D)

o SERVICE MODULE

- FOUR NORTH/SOUTH BUS PANELS
- SM FLOOR RETAINED FOR AOCS EQUIPMENT
- LOWER PANEL TO ENCLOSE SM, MOUNT UNITS.
- BATTERIES MOUNTED DIRECTLY TO BUS PANELS, NOT ISOLATED FROM SPACECRAFT INTERIOR (REDUCES RADIATOR REQUIREMENTS AND OPERATIONAL CONSTRAINTS) ; ALLOWANCE FOR UP TO FOUR NIH2 BATTERIES
- STRUTWORK REMOVED ; SM SUPPORTS BUS EQUIPMENT ONLY

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o SOLAR ARRAY

- CURRENT RMSD RIGID PANEL ARRAY HAS BEEN ASSUMED (2.4 X 1.9 M PANELS)
- STOWS "SIDEWAYS" ON SPACECRAFT; ARRAYS OF UP TO 6 PANELS PER WING POSSIBLE
- SOLAR ARRAY DRIVE MOUNTED DIRECTLY TO CENTRAL STRUCTURE
- EIGHT TIEDOWN LOCATIONS ARE PROVIDED; TIEDOWNS ARE MADE DIRECTLY TO SHEAR PANELS.

SUMMARY OF NEW DESIGN

o MORE COMPACT

- MORE EFFICIENT STRUCTURE, THERMAL CONTROL, ELECTRICAL AND RF DISTRIBUTION
- EASIER TO HANDLE AND TEST
- LEAVES MUCH MORE VOLUME AVAILABLE FOR STOWING ANTENNAS WITHIN LAUNCHER ENVELOPE (ESPECIALLY ABOVE EARTH FACING PANEL)

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- EASIER TO ASSEMBLY AND ANALYZE
- FEWER INTERFACES
- LOWER MASS

o ALLOWS FOR GROWTH

- EXTENDIBLE RADIATOR PANELS
- SIMILAR DESIGN APPROACH COULD BE USED FOR MUCH LARGER STRUCTURES

o RETAINS EXISTING OLYMPUS TECHNOLOGY

- STRUCTURE CONCEPT IS DERIVATIVE
- MINIMUM MODIFICATIONS TO VIRTUALLY ALL OTHER SYSTEMS.

OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

RECONFIGURED OLYMPUS

STRUCTURAL MASS ESTIMATES

THRUST TUBE & SEPARATION RING	35	
E/W SHEAR WALLS	10	
N/S SHEAR WALLS	5	
N/S PAYLOAD PANELS	30	
N/S BUS PANELS	15	
UPPER FLOOR (CM)	12	
MIDDLE FLOOR (SM)	10	
LOWER FLOOR (SM)	8	
CPS MOTOR FLOOR	8	
ACCESS PANELS	6	
MISCELLANEOUS STRUCTURAL EQUIPMENT	20	
TOTAL	159	KG
PREVIOUS OLYMPUS STRUCTURE MASS	243	KG
SAVINGS	84	KG

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OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

ANIK E ON RECONFIGURED OLYMPUS - MASS BUDGET

SUBSYSTEM	MASS (KG)	
STRUCTURE	159	- SPAR SASD STUDY
ARRAY (3500 WATT)	124	- SPAR RMSD STUDY
TT & C	40	
POWER (FULL ECLIPSE)	175	
ACS	84	- BAE STUDY
CPS	125	
THERMAL	45	
HARNESS	50	
TOTAL PLATFORM	802	
PAYLOAD	337	
TOTAL DRY	1139	
ALLOWABLE DRY	1186	(ARIANE 4 - 12 + 2 YEAR STORAGE)
MARGIN	47	

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GSE COMPATIBILITY

- ALL MAJOR OLYMPUS GSE WILL BE APPLICABLE TO THE MODIFIED OLYMPUS WITHOUT MODIFICATION OR CONSTRAINTS EXCEPT FOR THE SPACECRAFT CONTAINER AND THE SPACECRAFT HANDLING TROLLEY.

- THE HANDLING TROLLEY CAN BE USED WHEN THE SPACECRAFT IS ROTATED WITH THE Z-AXIS VERTICAL.

- SEVERAL MINOR ADAPTORS, BRACKETS, SLINGS WHICH ARE SPECIFIC TO OLYMPUS MODULE DIMENSIONS WOULD HAVE TO BE MODIFIED OR REDESIGNED. THIS INCLUDES SUCH ITEMS AS:

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  - N/S PANEL ADAPTOR
  - S/C HORIZONTAL SLING
  - N/S PANEL SLING
  - ANTENNA FLOOR ADAPTOR
  - EIM RELATED GSE
  - SOLAR ARRAY GSE

CONCLUSIONS

- WITH STRUCTURAL AND SOLAR ARRAY MODIFICATIONS AS DESCRIBED IN THIS STUDY, THE OLYMPUS PLATFORM HAS THE CAPABILITY OF COMPETING DIRECTLY AGAINST ALL EXISTING NORTH AMERICAN AND EUROPEAN PLATFORMS.

  SUCH MODIFICATIONS CAN BE CARRIED OUT IN A WAY TO RETAIN THE ESSENTIAL DESIGN CHARACTERISTICS OF THE OLYMPUS PLATFORM AND PRESERVE COMPATIBILITY WITH EXISTING GSE.

- COMPETITION WITH NORTH AMERICAN SPACECRAFT MAY REQUIRE THE ADOPTION OF A PROTOFLIGHT TESTING APPROACH, WHICH CAN ALLEVIATE THE NEED FOR A NEW QUALIFICATION MODEL STRUCTURE.

- IF THE ABOVE APPROACH IS ADOPTED, THE COST AND SCHEDULE IMPACT OF THE STRUCTURAL MODIFICATION CAN BE MINIMIZED

RECOMMENDATION

- THE WORK PERFORMED UNDER THE DOC STUDY HAS SHOWN THAT SIGNIFICANT MASS REDUCTIONS AND A MORE COMPETITIVE DESIGN IS POSSIBLE. TO MORE FULLY DEFINE THE MODIFICATIONS NEEDED, SPAR SUGGESTS THAT A PHASE A CONCEPTUAL STUDY BE INITIATED AS EARLY AS POSSIBLE AND BE COMPLETED BY MID 1987

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SECTION H: VUGRAPH PACKAGE DATED

NOVEMBER 27, 1987,

WHICH WAS PRESENTED TO DOC AT DOC HQ, OTTAWA

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SPAR BRIEFING TO DOC

ON OLYMPUS RECONFIGURATION STUDY

DOC HEADQUARTERS

NOVEMBER 27, 1986

BRIEFING TO DOC ON OLYMPUS RECONFIGURATION
OTTAWA, ONTARIO
NOVEMBER 27, 1986

SASD
SPAR

CONTENTS OF PRESENTATION	PRESENTER
1. SUMMARY OF RESULTS OF RECONFIGURATION STUDY	
MATERIAL PREPARED BY OLYMPUS PMC	A. KIDD
2. MATTERS ARISING OUT OF PRESENTATION TO OLYMPUS PARTICIPANTS	C. MORGAN
2.1 VIEWS AND CONCLUSIONS OF INDUSTRIAL CONSORTIA	
- EXTRACTS FROM PMC PRESENTATION	
2.2 ESA PROPOSED STUDY	
- CONTENT	
- SPAR RESPONSE	
- BENEFITS TO SPAR/CANADA	
2.3 WORK DISTRIBUTION ON RECONFIGURED OLYMPUS	
3. THE NEXT STEPS	
4. CONCLUSION	

STUDY GOALS AND DRIVERS

- THE BASIC PURPOSE OF THIS STUDY IS TO EXAMINE METHODS OF INCREASING THE COMPETITIVENESS OF THE OLYMPUS SPACECRAFT FOR PAYLOADS IN THE MEDIUM RANGE (MASS: 300 - 400 KG; POWER 2500 - 3500W).

  THIS IS THE RANGE MOST PAYLOADS ARE EXPECTED TO BE WITHIN THE NEXT DECADE. (REF: BAE STUDY)

- THE NEED FOR SUCH AN EFFORT CAN BE SEEN WHEN THE OLYMPUS L-1 MISSION IS COMPARED TO THE ANIK E ON THE RCA 5000.

		OLYMPUS L-1	NORTH AMERICAN PLATFORM (ANIK E)
PAYLOAD:	MASS	335 KG	337 KG
	POWER	2850 WATTS	2100 WATTS
	ECLIPSE POWER	1110 WATTS	2100 WATTS
LAUNCH VEHICLE		DEDICATED ARIANE III (2420 KG GTO)	DUAL LAUNCH ARIANE IV (2500 KG GTO)
MISSION LIFE		5 YEARS	10 YEARS + 2 YEARS IN ORBIT STORAGE

THE NORTH AMERICAN PLATFORM CLEARLY OFFERS GREATER ECLIPSE POWER AND LIFE WHILE USING ALMOST IDENTICAL LAUNCH RESOURCES.

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GOALS AND DRIVERS (CONTINUED)

- WHEN THE OLYMPUS SPACECRAFT IS COMPARED TO NORTH AMERICAN SPACECRAFT OF THE 300 - 400 KG/2500 - 3500 WATT RANGE, TWO SUBSYSTEMS STAND OUT AS VERY HEAVY IN COMPARISON. THESE ARE STRUCTURE AND POWER. THE OLYMPUS STRUCTURE DELTA IS ABOUT 100 KG AND THE OLYMPUS SOLAR ARRAY ALONE HAS A 90 KG DELTA. THE REMAINING OLYMPUS POWER SUBSYSTEM IS ABOUT 60 KG HEAVIER THAN AN EQUIVALENT NORTH AMERICAN SYSTEM

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OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

GOALS AND DRIVERS (CONTINUED)

- THREE TARGET PAYLOADS USED IN STUDY:

  ANIK E
  AUSSAT II
  NORTH AMERICAN MSAT

- ALL OF THE ABOVE PAYLOADS CAN BE EXPECTED TO REQUIRE A DUAL LAUNCH ON AN ARIANE IV. THIS LEADS TO THE FOLLOWING BUS REQUIREMENT.

  - LAUNCH CAPACITY OF ARIANE IV DUAL LAUNCH IS 2500 KG INTO GTO.

  - AVERAGE PAYLOAD REQUIREMENTS ARE 400 KG AND 2500 WATTS.

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- THE ABOVE DATA, COMBINED WITH ASSUMPTION OF LIQUID APOGEE ENGINE, BIPROP CPS, AND 10 YEAR LIFE, LEAVE A MAXIMUM ALLOWANCE OF 796 KG FOR THE PLATFORM WITH A 44 KG MARGIN. THE SPACECRAFT POWER MUST BE SUPPLIED USING THIS MASS.

OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

POWER SUBSYSTEM

- STUDIES MADE DURING THE RADARSAT PROGRAM AT SPAR HAVE SHOWN THAT SUBSTANTIAL MASS SAVINGS (ABOUT 30 KG) WOULD BE POSSIBLE BY ALLOWING THE PAYLOAD UNITS THE OPTION AT UNREGULATED POWER, WHILE CONTINUING TO SUPPLY 50 VDC REGULATED POWER TO THE PLATFORM UNITS

  THE ISSUE OF WHETHER THE BATTERIES NEED TO HAVE DEDICATED RADIATOR PANELS SHOULD ALSO BE ADDRESSED

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OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

LAUNCH VEHICLE COMPATIBILITY

- IN LIGHT OF THE STS ACCIDENT AND ARIANCESPACE'S DEVELOPMENT LEAD, IT WOULD APPEAR THAT ARIANE IV CAN BE EXPECTED TO SET THE STANDARD FOR LAUNCH VEHICLE ENVELOPES.
- USING THE FULL ARIANE IV DIAMETER (3.65M) RESULTS IN A MORE COMPACT PLATFORM DESIGN WHICH ALLOWS MORE HEADROOM TO STOW THE LARGE REFLECTORS REQUIRED FOR MANY PAYLOADS.
- SOME EXISTING ELV'S (SUCH AS THE TITAN IV) REQUIRE USE OF AN UPPER STAGE. A COMPACT DESIGN GREATLY EASES STOWAGE CONSTRAINTS IN SUCH CASES.

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OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

STRUCTURE/CONFIGURATION

THE CURRENT OLYMPUS ARCHITECTURE MAKES THE SPACECRAFT HEAVIER FOR SMALLER PAYLOADS:

1. SIZED FOR ARIANE III LAUNCH
   - RESULTS IN A LONGER, THINNER, AND HEAVIER STRUCTURE

2. DIMENSIONED FOR MAXIMUM PAYLOAD
   - RESULTS IN HEAVIER THERMAL, ELECTRICAL DISTRIBUTION
   - SOME STRUCTURE (NOT ALL) SIZED FOR MAXIMUM P/L

3. DIFFICULT LOAD PATHS
   - COMMUNICATION/SERVICE MODULE TRANSITIONS
   - MUCH OF THE BENDING LOAD IS IMPARTED TO PAYLOAD PANELS AND SOLAR ARRAYS.

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INITIAL CONCLUSIONS:

- TWO POSSIBLE WEIGHT REDUCTION SCENARIOS CAN BE ENVISAGED
  - SUBSYSTEM WEIGHT SAVING CAMPAIGN BY USE OF LIGHTWEIGHT MATERIALS, DESIGN OPTIMIZATION, ETC.
  - RECONFIGURATION OF STRUCTURE SUBSYSTEM
- IT IS UNLIKELY THAT SUBSYSTEM WEIGHT SAVING CAMPAIGN COULD PROVIDE THE DESIRED 180 KG REDUCTION, THEREFORE, THE SPAR STUDY FOCUSED IN ON THE RECONFIGURATION SCENARIO.

APPROACH FOR THE RECONFIGURATION STUDY

- DESIGN FOR ARIANE IV DUAL LAUNCH, UPPER POSITION
- DESIGN THE STRUCTURE TO ACCOMMODATE THE SMALLER PAYLOADS, AND A MORE NARROW RANGE OF SIZES.
- SPACECRAFT GROWTH IS ACHIEVED BY COMMONALITY OF DESIGN APPROACH AND STRUCTURAL TECHNIQUES.

OLYMPUS P.M.C.

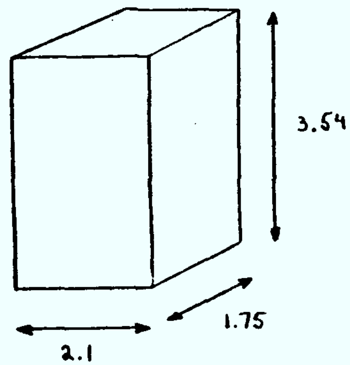
PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

#11

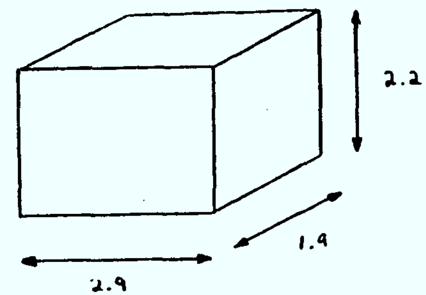
OLYMPUS RECONFIGURATION

- DIMENSIONS:

OLYMPUS -



PROPOSED -



- SPACECRAFT HAS BEEN "TRUNCATED" AT WHAT WAS PREVIOUSLY THE SOLAR ARRAY DRIVE LEVEL, 1.6M. ABOVE THE SEPARATION PLANE

- X/Y DIMENSIONS INCREASED TO FILL THE ARIANE IV ENVELOPE

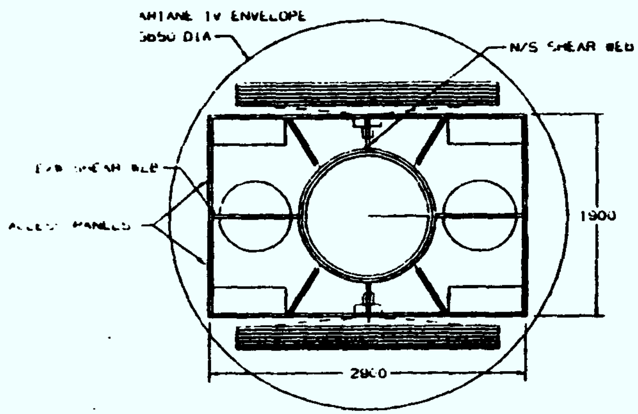
- UPPER FUEL TANK REMOVED ; HYDRAZINE CARRIED IN TWO CYLINDRICAL TANKS OF EQUAL VOLUME LOCATED ADJACENT TO THE THRUST TUBE.

OLYMPUS P.M.C.

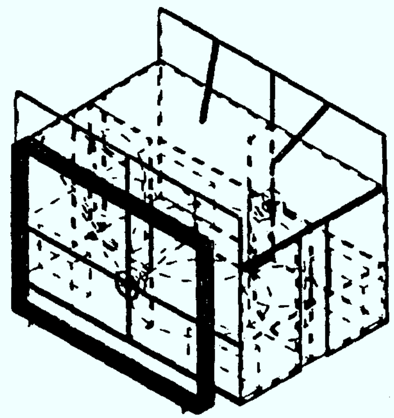
PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

PLATFORM GEOMETRY : RECONFIGURED OLYMPUS

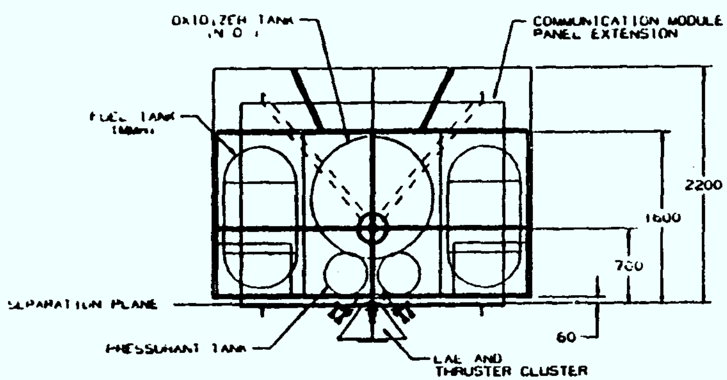
H/2



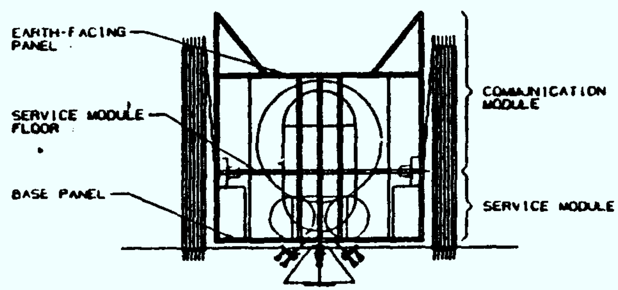
EARTH-FACING VIEW



ISOMETRIC VIEW
(PROPULSION MODULE REMOVED)



NORTH/SOUTH VIEW



EAST/WEST VIEW

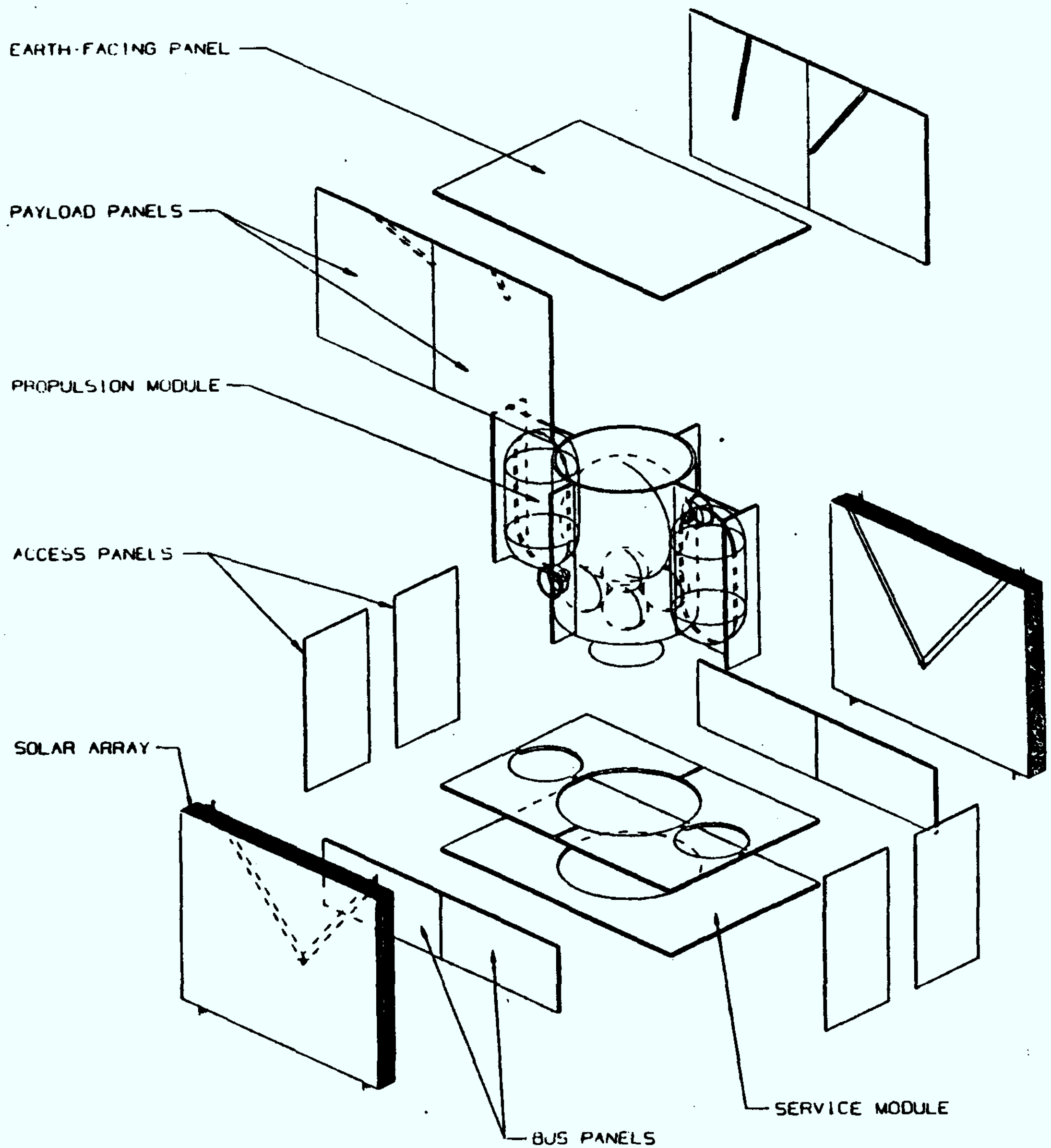
OLYMPUS P.M.C.

PRESENTATION TO THE
OLYMPUS PARTICIPANTS
OCTOBER 13, 1986

RECONFIGURED OLYMPUS

PLATFORM GEOMETRY :

H/3



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- COMMUNICATION MODULE:
  - ONE OR TWO PAYLOAD PANELS PER NORTH/SOUTH FACE
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DESIGN MODIFICATIONS (CONT'D)

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H15

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OLYMPUS P.M.C.

PRESENTATION TO THE
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OCTOBER 13, 1986

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OLYMPUS P.M.C.

PRESENTATION TO THE
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OCTOBER 13, 1986

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3. OLYMPUS MARKETING CONSIDERATIONS

3.3 OLYMPUS MARKET NICHE

- OLYMPUS can be competitive at the high end of its range (7.5 kW/500 kg) where its efficiency is optimised

- At the lower end of its range (2.5 - 3.5 kW/300 - 400 kg) OLYMPUS could be competitive only if its cost effectiveness can be improved

- OLYMPUS F2 is competitive because the design and build costs for the platform have been funded in the current programme

- OLYMPUS should be supported by the participating countries for national programmes and by ESA for technology advancement programmes.

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3. OLYMPUS MARKETING CONSIDERATIONS

3.4 MARKET STRATEGY

- As much publicity as possible should be derived from the F1 programme during the final integration, launch and in-orbit performance. The marketing "tools" such as price and schedule data, advertising and system application studies should be developed

- Near term targets (up to early 1990's) should be actively pursued especially:

  - The use of F2 for Italian Direct Broadcast
  - EUROPESAT
  - AUSSAT II (maximum configuration)

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- To make OLYMPUS competitive at 2.5 - 3.5 kW/300 - 400 kg the design evolution process must be vigorously pursued and weight reduction work continued. The weight reduced reconfigured OLYMPUS could be sold as direct competition to RCA, Ford, HAC and AEROSPATIALE/MBB in the 2.5 - 3.5 kW/300 -400 kg class and a plan must be developed to ensure that its build time can meet potential customer needs. Other identified targets must be actively pursued using the reduced/reconfigured OLYMPUS platform whilst continually assessing the competition's performance.

- The potential of using OLYMPUS in a "CONDOMINIUM" approach for providing regional direct broadcast into Europe must be explored.

4.5 OLYMPUS DESIGN EVOLUTION

Whilst OLYMPUS holds a unique position at the top end of the range, it is prudent to consider the necessary steps needed to improve the competitive position at the low end of the range and thus increase the market potential. The main effort is directed towards mass saving, and this fact is illustrated graphically in Section 4.1. Analysis shows the priorities to be the solar array, structure and attitude control subsystem.

Studies are in hand to assess these aspects in both Europe and Canada. The studies are:

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- Rigid solar array - ESA Study (BAe, Fokker, SPAR)
- Attitude control electronics subsystem - ESA Study (BAe)
- Conceptual reconfiguration - DOC Canada Study (SPAR)

Because of the emergence of significant targets at the low end of the OLYMPUS range it is recommended that these activities should be continued.

Fokker AERITALIA BRITISH AEROSPACE SELENIA SPAZIO SPAR

Olympus P.M.C

Presentation to
the Olympus Participants
13th October, 1986

4. Report by PMC

4.8 CONCLUSIONS

- Marketing activities can now really begin with primary focus on near-term targets
  - F2 RAI
  - Europesat
  - Aussat II (Maximum configuration)
- Design evolution being planned to expand and improve market nic to improve longer term target capture
- Address global market opportunities
- Encourage Europe to think Olympus

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ESA PROPOSED STUDY

SASD SPAR

STUDY SUMMARY

0 THE PROPOSED ESA STUDY CONSISTS OF 3 PARTS

TASK 1 PERFORMANCE COMPARISON WITH U.S. SPACECRAFT

TASK 2 REQUIREMENTS COMPARISON WITH U.S. SPACECRAFT

TASK 3 AREAS OF POTENTIAL IMPROVEMENT

0 ESA PROPOSED THAT SPAR AEROSPACE LEAD THE STUDY AND BAE WOULD PROVIDE NEEDED INPUTS

0 ESA EXPECTED THE STUDY TO CONSUME ABOUT 20 MAN MONTHS OF EFFORT

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STUDY SUMMARY

O ESA EXPECTS A STUDY DURATION OF ABOUT 4 MONTHS. THE SUGGESTED OUTLINE OF THE STUDY WAS AS FOLLOWS:

WEEK 0	-	KICK OFF MEETING AT SPAR
WEEK 8	-	PROGRESS MEETING AT SPAR TO REVIEW RESULTS OF TASKS 1 AND 2
WEEK 12	-	PROGRESS MEETING AT SPAR TO REVIEW RESULTS OF TASK 3
WEEK 15	-	PRESENTATION OF RESULTS OF STUDY AND REVIEW OF DRAFT FINAL REPORT AT ESTEC
WEEK 18	-	DELIVERY OF FINAL REPORT

#24

SPAR RESPONSE TO ESA

SPAR LARGELY AGREED TO THE ESA PROPOSED SOW, SUBJECT TO THE FOLLOWING COMMENTS

- O SINCE SPAR MUST RESPECT THE PROPRIETARY NATURE OF INFORMATION GIVEN BY OTHER SPACECRAFT MANUFACTURERS, NO DETAILED SUBSYSTEM CAN BE GIVEN TO ESA. INSTEAD, TYPICAL VALUES WILL BE PRESENTED. THE STUDY SHOULD BE LIMITED TO NO LESS THAN 3 TYPES OF SPACECRAFT.

- O SPAR FEELS TASK 3 (AREAS OF POTENTIAL IMPROVEMENT) SHOULD BE EXTENDED AND THE OTHER TASKS SHORTENED SINCE A START HAS ALREADY BEEN MADE IN THIS AREA.

- O SPAR WOULD LIKE TO ADD TO TASK 3 THE GENERATION OF A SIMPLE STRUCTURAL MODEL TO VERIFY MASS SAVING ESTIMATES.

- O A CHANGE IN THE MEETING SCHEDULE SHOULD BE MADE TO REFLECT THE ABOVE COMMENTS

#427

SPAR RESPONSE TO ESA

SPAR PROPOSED A 20 - 23 MAN MONTH EFFORT, DIVIDED AS FOLLOWS:

SPAR	15 - 16 MAN MONTHS
BAE	3 - 5 MAN MONTHS
FOKKER	2 MAN MONTHS

BAE AND FOKKER HAVE INDICATED A WILLINGNESS TO SUPPORT SPAR AT THE ABOVE LEVELS.

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BENEFITS TO SPAR/CANADA

THE PRINCIPAL BENEFITS ARISING FROM SUCH A STUDY WOULD BE:

0 THE DEVELOPMENT OF A COMPETITIVE SPACECRAFT PLATFORM IN WHICH SPAR, AND CANADA WOULD HAVE A SHARE.

0 THE FURTHER DEVELOPMENT OF SPACECRAFT LEVEL STRUCTURAL ANALYSIS SKILLS.

0 THE FURTHER ESTABLISHMENT OF SPAR AND CANADA'S ROLE AS A REPRESENTATIVE OF NORTH AMERICAN SPACECRAFT TECHNOLOGY WITH ESA.

H29

SASD SPAR

WORK DISTRIBUTION ON RECONFIGURED OLYMPUS

- DISTRIBUTION OF WORK ON OLYMPUS DEPENDS TO A LARGE EXTENT ON INDUSTRIAL AGREEMENT AND WHO PRIMES THE SPACECRAFT CONTRACT IN QUESTION. IT WOULD BE PREMATURE TO ATTEMPT TO FORECAST SUCH A DISTRIBUTION AT THIS TIME.

- HOWEVER, DISTRIBUTIONS INVOLVING CANADIAN INDUSTRY SHOULD BE POSSIBLE IN THE FOLLOWING AREAS.

1) <u>PAYLOAD</u>

THIS IS MISSION SPECIFIC. HOWEVER, SPAR WOULD EXPECT TO WORK WITH LEAD CONSORTIUM MEMBER IN IDENTIFYING PARTS TO BE BUILT IN CANADA.

2) <u>SPACECRAFT LEVEL I & T</u>

IT WOULD BE LIKELY FOR SPACECRAFT LEVEL I & T TO BE PERFORMED IN CANADA. ALSO STRUCTURAL TESTING OF THE MODIFIED STRUCTURE COULD TAKE PLACE AT DFL.

3) <u>SOLAR ARRAY</u>

CURRENT STUDIES OF RECONFIGURED SOLAR ARRAYS INDICATE THAT FOKKER MAY HAVE A TECHNICAL AND COST ADVANTAGE IN RIGID PANEL ARRAY DESIGN BECAUSE OF PREVIOUS EXPERIENCE IN THIS AREA. THIS MAY LEAD TO FOKKER RECEIVING ARRAY WORK IN THE RECONFIGURED OLYMPUS.

H30

WORK DISTRIBUTION ON RECONFIGURED OLYMPUS (CONT'D)

4) STRUCTURE

SPAR WOULD PROPOSE THAT STRUCTURE/THERMAL ANALYSIS, BUILD AND TEST BE SHARED IN ORDER TO MAINTAIN CANADA'S POSITION IN THE CONSORTIUM.

5) POWER SUBSYSTEM

PREVIOUS STUDIES HAVE IDENTIFIED SCOPE FOR IMPROVEMENT IN THE OLYMPUS POWER SUBSYSTEM (IN ADDITION TO SOLAR ARRAYS). CANADIAN INDUSTRY HAS STRONG TECHNICAL CAPABILITIES IN THIS AREA. SPAR WOULD BE PREPARED TO NEGOTIATE WITH BAE TO GIVE CANADIAN INDUSTRY A SHARE IN THIS AREA.

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SASD
SPAR

NEXT STEPS

O OBTAIN DOC SUPPORT FOR SPAR IN WINNING ESA STUDY

O AS A RESULT OF PERFORMING STUDY, SPAR WOULD BEGIN TO ESTABLISH A ROLE WITHIN ESA AS A FULL AND ACTIVE TEAM MEMBER OF THE OLYMPUS PROGRAM, WITH RECOGNIZED EXPERTISE IN STRUCTURAL/THERMAL/POWER AREAS, IN ADDITION TO THE PAYLOAD, I & T AND ARRAY TECHNOLOGIES ALREADY DEMONSTRATED.

H32

CONCLUSION

0 OLYMPUS MODIFICATION OFFERS REAL PROMISE OF GENERATING A COMPETITIVE PLATFORM IN THE CLASS OF SPACECRAFT WHICH WILL HAVE THE GREATEST NUMBER OF EXPECTED SALES IN THE NEXT DECADE.

0 THE PROGRAM OFFERS A REASONABLE METHOD FOR CANADA TO GET A SHARE OF A PLATFORM.

0 ESA STUDY SHOULD BE SUPPORTED

H33

SECTION J: LETTER REPORT ADDRESSING THE

DOC S.O.W. ANNEX A REV 1 ITEMS

SECTION J: LETTER REPORT ADDRESSING THE DOC S.O.W.

ANNEX A REV 1 ITEMS

ANNEX A REV 1 ITEM #	COMMENT ON WORK PERFORMED
1	This is DOC Statement - no Spar action
2.	The current payload support capability of Eurostar is below the Olympus L-1 capacity. Eurostar brochure (EUR/6/86) provided to Mr. G. Booth gives data on Eurostar which comfirms the above. BAe have stated that they see Eurostar complementing rather than competing with Olympus. Cost differentials are not known as each prime requirement is costed separately. The Olympus PMC are addressing cost modelling during 1987.
3.	This was not addressed by JPB as the reconfiguration study itself did not address this topic before the October 13, 1986 presentation. Spar provided a Canadian perspective in Section H of this report.
4.	See paragraph 2 of Item 2 above.
5.	The heritage of a downsized Olympus which follows a launch of L1 will be the same as for U.S. manufactures who stretch their designs i.e. RCA 4000 to 5000 for Anik E. Downsizing modifies significantly only the structure and fuel tanks. Non-recurring costs associated with a downsizing could be amortized over several programs.
6.	All work concerning the Rigid Solar Array inputs to the study were performed by Spar RMSD.

7. Presentation material generated for the Olympus participants meeting October 13, 1986 was supplied to DOC for comment prior to being sent to BAe for incorporation into the presentation handout. (See Section G).

8. The October 13, 1986 presentation handout was provided to all participants (G. Booth). No official minutes were taken at the meeting although the points raised are to be included in the PMC minutes and will be issued by PMC after the #21 meeting scheduled for January, 1987.

9. Copies of pertinent material, tel/cons and review meetings were held as agreed

10. All material supplied.