The coating requirements for the Command Module heat shield are shown in Figure 18. Also noted are the expected properties following the boost contamination. Figure 19 summarizes the effect of the contamination. The reduction in absolute minimum temperature means that either tighter constraints must be placed upon spacecraft attitude or the temperature requirement relaxed. The specification minimum heat shield temperature of -150°F is based on microscopic surface cracking of the ablator and gaps in the heat shield joints resulting from thermal strains The actual occurence of cracking is unlikely in the thinest sections of the heat shield which are the coldest. The thermal strain analysis is not considered of sufficient accuracy to predict that gaps will not occur at -150°F and will occur at -170°F. The heat shield thermal deformations will be verified in the MSC Space Environment Simulation Laboratory (SESL). An increase in predicted entry temperature requires that the heat shield be cold-soaked back to +150°F prior to entry for a worst case entry trajectory. Cabin environment temperature extremes are changed by less than 5°F as the result of coating degradation.

Although other coatings are used on the spacecraft, their requirements are sufficiently loose so that no problem results from the boost contamination.

We reviewed the available window contamination data previously. It is expected that jettisonable covers will be necessary to assure the visibility required for docking and scientific observation. The data from the AS-202 mission should confirm this. Such covers are in design at the present time.

MSC is also designing covers which can be used for protection of the Service Module radiators if desired. Effectivity of these covers will be determined upon conclusion of the present studies.

4.0 CONCLUSIONS

Conclusions resulting from these studies are as follows:

1. Execution of the lunar mission requires addition of ECS cooling water and tankage or boost protective covers for the ECS radiator coating.

2. EPS radiator performance is acceptable with predicted coating degradation.

3. Command Module heat shield and cabin temperatures slightly exceed specification limits for predicted Command Module coating degradation. These off-limits conditions are acceptable.

4. Predicted degradation was obtained by extrapolation of limited data; therefore, some uncertainty exists. They are "best engineering judgment" predictions. Flight data from AS-204 and -205 is required for confirmation.

5. The spacecraft windows will be degraded. The extent and acceptability will be investigated further and verified by flight data.

6. No spacecraft changes will be implemented pending results of boost cover design studies and further flight data.

NASA-S-66-5219 JUN

THERMAL ATTITUDE CRITERIA

EARTH PARKING ORBIT

- X-AXIS PARALLEL TO VELOCITY VECTOR ± 20°
- 1.5 TO 4.5 HOURS DURATION
- S C ROLLED 180° 1 HOUR AFTER START OF THIS PHASE
- CONSIDER WORST CASE ROLL WITHIN ABOVE LIMITS

TRANSLUNAR INJECTION THRU TRANSPOSITION

- 0.5 TO 2.0 HOURS DURATION
- CONSIDER ATTITUDE RANDOM (i e, WORST CASE)

Figure 1

NASA-S-66-5221 JUN

THERMAL ATTITUDE CRITERIA (CONT)

TRANSLUNAR-TRANSEARTH COAST

- X-AXIS NORMAL TO INCIDENT SUN ± 20°
- ROLL AT 1.0-2.5 REV/HOUR
- 60 TO 110 HOURS DURATION
- ARBITRARY (WORST CASE) ATTITUDE PERIODS OF 3 HOURS MAX AT ANY TIME
- ROLL MODE STABILIZED TEMPERATURES TO INITIATE ALL ARBITRARY ATTITUDE PERIODS

Figure 1 (cont)

NASA-S-66-5220 JUN

THERMAL ATTITUDE CRITERIA (CONT)

LUNAR ORBIT

- X-AXIS APEX DOWN ± 20° OF LOCAL VERTICAL
- ARBITRARY ATTITUDE FOR LM ASCENT & DESCENT
- ARBITRARY ATTITUDE FOR PERIODS OF 3 ORBITS WITH TOTAL NOT TO EXCEED 8 ORBITS
- ASSUME STABILIZATION TO START ARBITRARY PERIOD

LUNAR LANDING

- PRIOR KNOWLEDGE OF DAY OR NIGHT LANDING
- LANDING ORIENTATION CONSTRAINED IN TILT ANGLE & SEPARATION DISTANCE

Figure 1 (cont)

NASA-S-66-5227 JUN

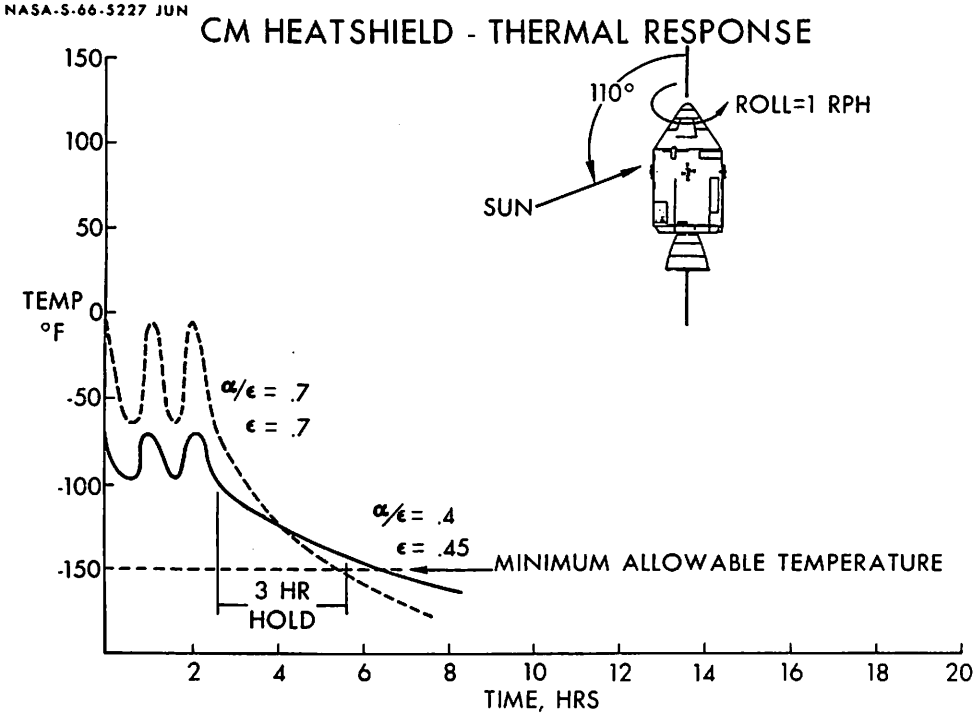


Figure 2

NASA-S-66-6126 JUN

MOST SENSITIVE COMPONENTS

CSM

- CM HEAT SHIELD
- CM RCS ENGINE
- SM RCS ENGINE
- SPS ENGINE AND FEED LINES
- ECS COOLANT LOOP
- EPS COOLANT LOOP
- CREW COMPARTMENT
- THERMAL COATINGS
- EXPENDABLE STORAGE

Figure 3

NASA-S-66-6428 JUN

MOST SENSITIVE COMPONENTS (CONT)

- LM
  - LANDING GEAR
  - EXPENDABLE STORAGE
  - HIGH GAIN, LANDING AND RENDEZVOUS RADAR ANTENNA
  - A/S RCS ENGINE
  - DESENT ENGINE
  - ASCENT ENGINE
  - THERMAL COATINGS
  - COOLANT LOOP
  - CREW COMPARTMENT

Figure 3 (cont)

NASA-S-66-6472 JUN

THERMAL DESIGN

- DESIGN IS BASED UPON CONTROLLED TEMPERATURE TRANSIENTS
- STEADY-STATE DESIGN RESULTS IN WEIGHT AND COMPLEXITY PENALTIES
- LIMITED CAPABILITY FOR WORST CASE MISSION CONDITIONS
- THESE CONDITIONS ESTABLISH DESIGN ENVELOPE
- THIS CAPABILITY ALLOWS A WIDE RANGE OF MISSIONS TO BE EXECUTED
- PARAMETRIC DATA TO DEFINE BOUNDS OF CAPABILITY IS IN WORK

Figure 4

NASA-S-66-6040 MAY

SOURCE OF DEPOSITS

- LAUNCH VEHICLE RETRO ROCKETS
- TOWER JETTISON MOTOR OPERATION
- ABLATING CORK

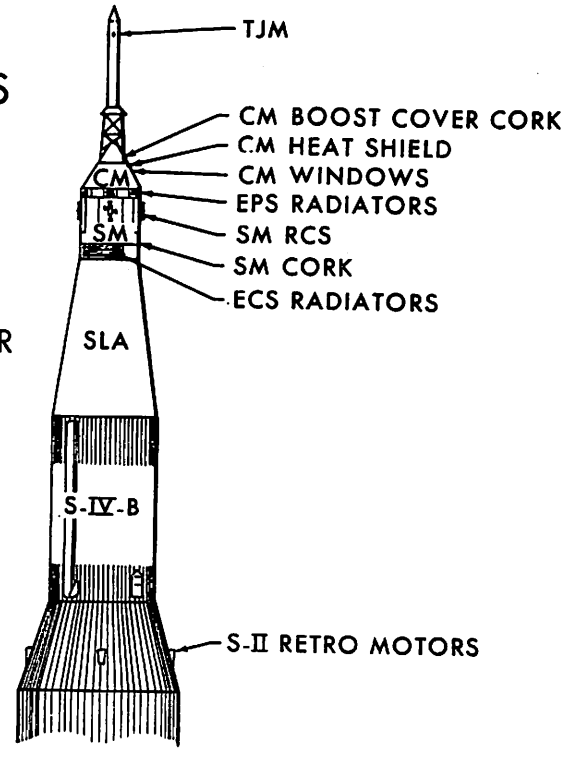


Figure 5

NASA-S-66-5035 JUNE 1

CONTAMINATION DATA

FLIGHT TEST DATA

(S-13 WHITE PAINT - USED ON S-IV)

SATURN 8,9,AND 10 (PEGASUS) INFLIGHT DATA - MSFC

	EXPOSED SMA EXTERNAL	SHROUDED CALORIMETERS
INITIAL α/ϵ	0.25	0.25
FINAL α/ϵ	0.5 TO 0.6	0.25

Figure 6

NASA-S-66-5216 JUN

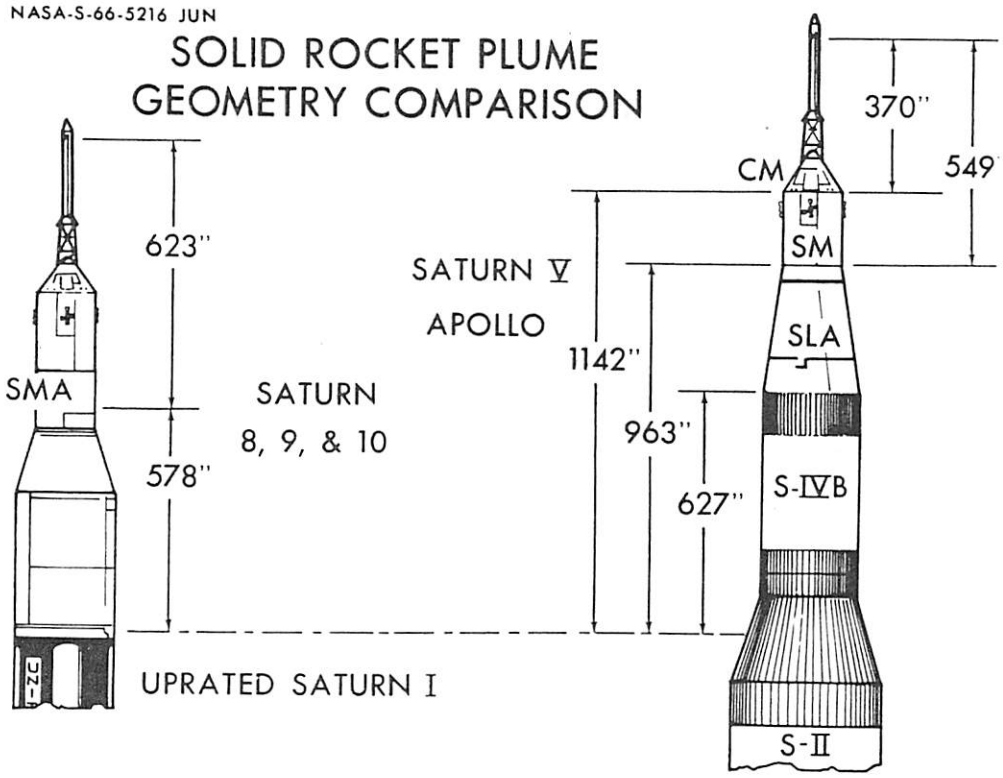


Figure 7

NASA-S-66-5224 JUN

CENTAUR/IB CONTAMINATION TEST AT AEDC

J-4 FACILITY

TEST ALTITUDE: 120,000 FEET

MOTOR: TX-143

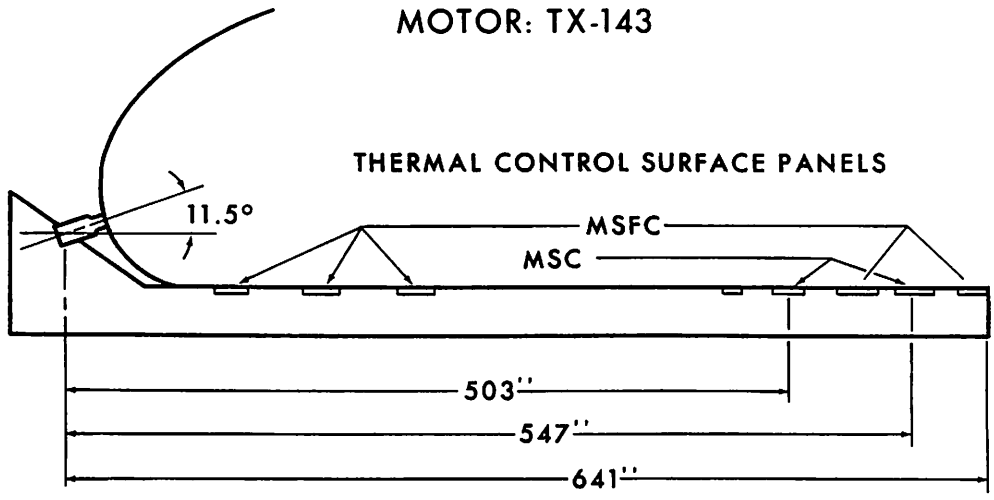


Figure 8

NASA-S-66-5036 JUNE 1

CONTAMINATION DATA (CONT)

GROUND TEST DATA

AEDC

APOLLO RADIATOR COATING	α/ϵ	α	ϵ
INITIAL (PRE-FIRE)	.196	.18	.92
FINAL (POST-FIRE)			
AVERAGE	.35	.33	.93
RANGE	.28-.45	.26-.42	.92-.94
APOLLO SM SKIN COATING	α/ϵ	α	ϵ
INITIAL (PRE-FIRE)	1.0	.25	.25
FINAL (POST-FIRE)			
AVERAGE	1.14	.50	.43
RANGE	.96-1.43	.38-.63	.28-.66

Figure 9

NASA S-66-5406 MAY 31

TJM AND S-II RETRO PARTICLE IMPINGEMENT

	TOTAL INCIDENT FLUX LB_m / FT^2	COATING PROPERTIES INITIAL α	ϵ	FINAL α	ϵ
AEDC TEST 528 INCHES FROM NOZZLE	1.64×10^{-4}	.18	.90	.42	.90
AEDC TEST 569 INCHES FROM NOZZLE	1.32×10^{-4}	.18	.90	.35	.90
SA 8, 9, 10 - SMA	2.02×10^{-3}	.22	.90	.45-.54	.90
APOLLO - CM	8.3×10^{-3}	.16	.40		
APOLLO - SM	2.9×10^{-3}	.18	.90	PREDICTED .50	.90

Figure 10

NASA-S-66-5033 JUNE 1

EMISSION SPECTROGRAPHIC ANALYSIS

	SPACECRAFT 009	SPACECRAFT 002	AEDC TESTS
ALUMINUM	MAJOR**	LIGHT MINOR	*
ANTIMONY	TRACE	NONE	NONE
BARIUM	TRACE	NONE	NONE
BISMUTH	NONE	TRACE	NONE
BORON	LIGHT MINOR	LIGHT MINOR	LIGHT MINOR
CADMIUM	BASE	NONE	BASE
CALCIUM	HEAVY MINOR	BASE	TRACE
CHROMIUM	LIGHT MAJOR	TRACE	TRACE
COPPER	MINOR	BASE	LIGHT MINOR
IRON	MAJOR	MAJOR	LIGHT MAJOR
LEAD	HEAVY MINOR	HEAVY MAJOR	LIGHT MAJOR
MAGNESIUM	HEAVY MINOR	BASE	BASE

Figure 11

NASA-S-66-5034 JUNE 1

EMISSION SPECTROGRAPHIC ANALYSIS (CONT)

	SPACECRAFT 009	SPACECRAFT 002	AEDC TESTS
MANGANESE	LIGHT MINOR	TRACE	TRACE
MOLYBDENUM	TRACE	NONE	NONE
NICKEL	TRACE	TRACE	TRACE
POTASSIUM	NONE	NONE	*
SILICON	HEAVY MAJOR	MINOR	*
SILVER	TRACE	NONE	TRACE
SODIUM	HEAVY MAJOR	MAJOR	NONE
TIN	BASE	BASE	BASE
TITANIUM	LIGHT MAJOR	LIGHT MAJOR	NONE
VANADIUM	BASE	NONE	NONE
ZINC	LIGHT MAJOR	HEAVY MINOR	*
ZIRCONIUM	TRACE	NONE	NONE

Figure 11 (cont)

NASA-S-66-5007 JUN

SOLAR ABSORPTIVITY VS CONTAMINATION FOR APOLLO T/C COATING

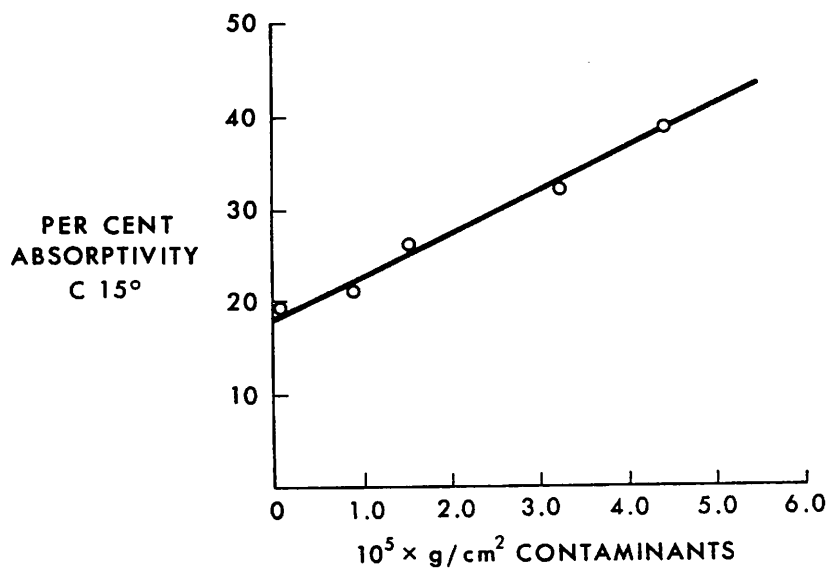


Figure 12

NASA-S-66-5223 JUN

C/M WINDOWS

AFRM 009 TRANSMITTANCE, %

	TOTAL	DIRECT	DIFFUSE
CLEAN	90 - 97	90 - 97	—
DEGRADED	68	29	39

- COMMENTS
  - NO DETERMINATION OF RELATIVE EFFECT OF BOOST, ENTRY, AND RECOVERY
  - CAMERA WILL BE FLOWN ON AS 202
  - MORE QUANTITATIVE ANALYSIS IS IN PROGRESS

Figure 13

NASA-S-66-5032 JUNE 1

PRESENT THERMAL COATINGS

	α		ϵ	
	DESIGN	DEGRADED	DESIGN	DEGRADED
ECS RADIATOR	0.2	.5	0.92	.92
EPS RADIATOR	0.2	.5	0.92	.92
CM HEAT SHIELD	0.16	.49	0.4	.7

Figure 14

NASA-S-66-6820 JUN

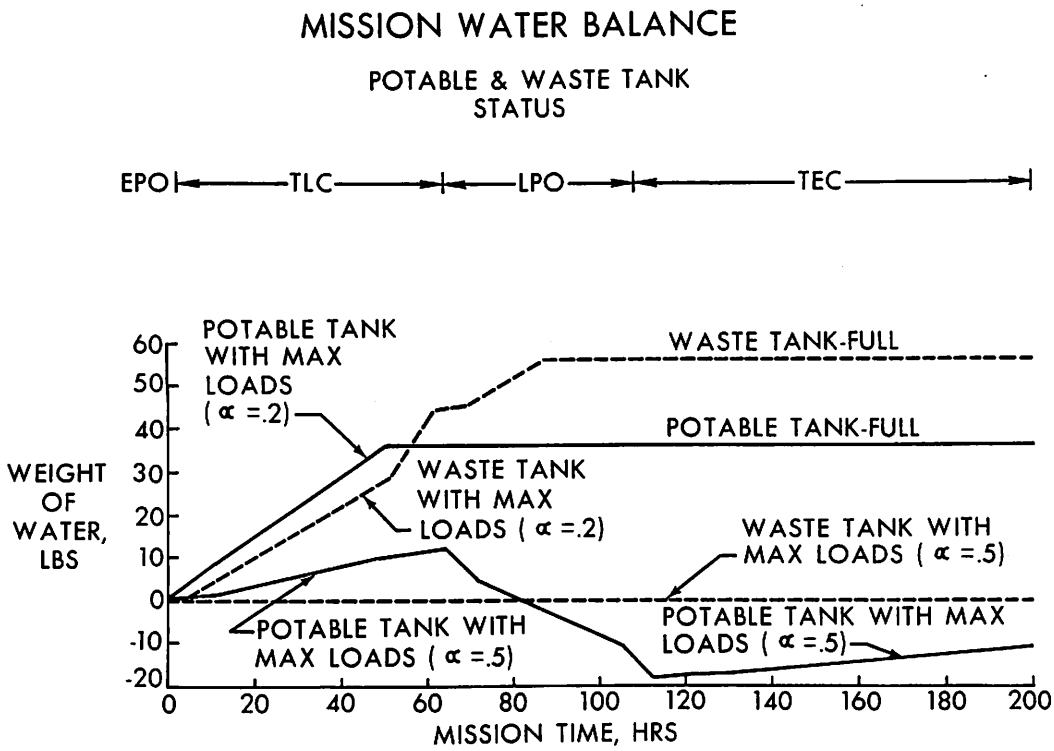


Figure 15

NASA-S-66-5030 JUNE 1

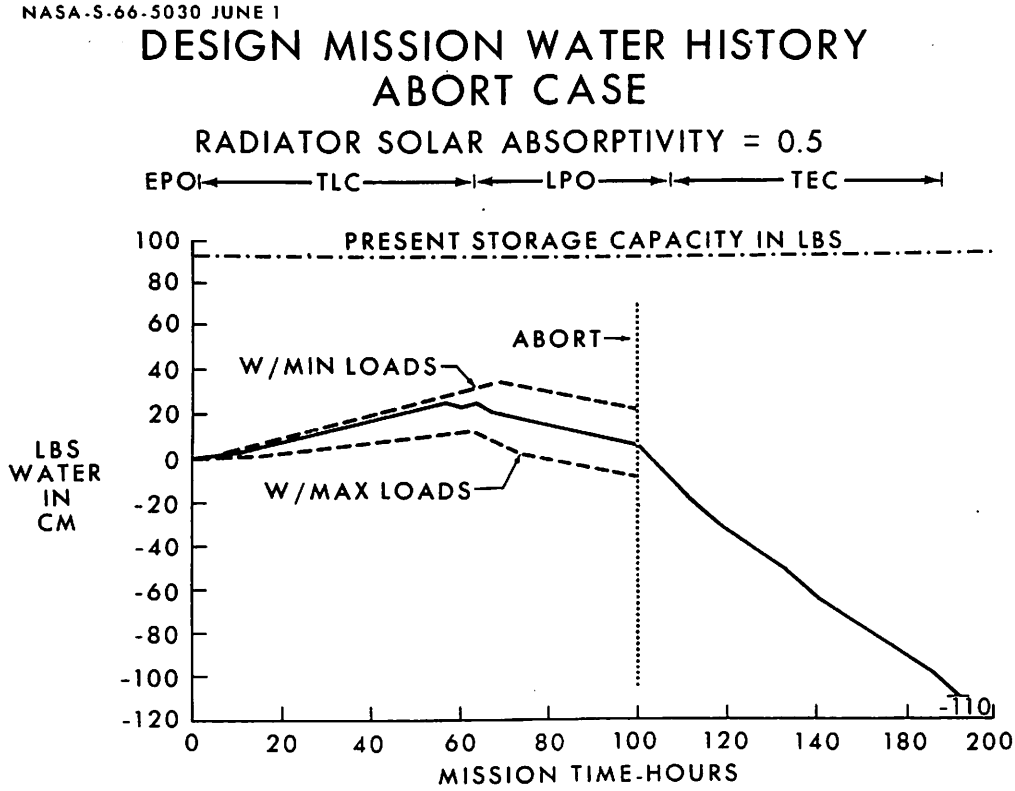


Figure 16

NASA-S-66-6809 JUN

BLOCK II EPS RADIATOR CAPACITY FOR TWO FUEL CELLS IN LUNAR ORBIT

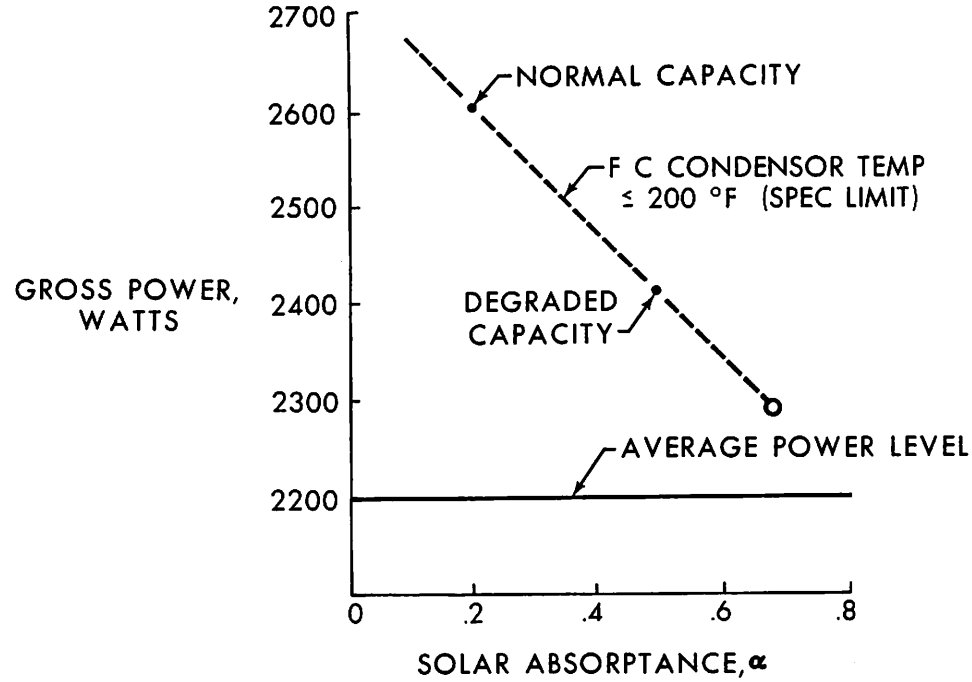


Figure 17

NASA-S-66-6450 JUN

CM COATING PERFORMANCE*

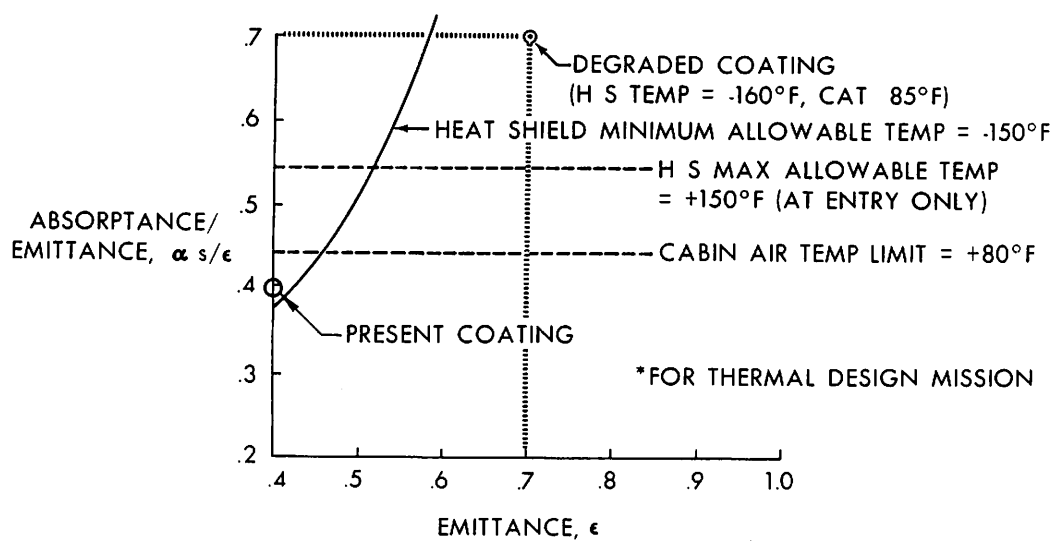


Figure 18

NASA-S-66-6455 JUN

RESULTING TEMPERATURE CONDITIONS - BLOCK II

HEAT SHIELD - MINIMUM ABLATOR TEMP CRITERIA

$\alpha_{s/\epsilon}$ =0.4 ϵ =0.4	-150
$\alpha_{s/\epsilon}$ =0.7 ϵ =0.65	-170°F

HEAT SHIELD - MAXIMUM ABLATOR TEMP (PRE-ENTRY)

$\alpha_{s/\epsilon}$ =0.4 (BLOCK II TCC)	+108°F
$\alpha_{s/\epsilon}$ =0.54	+150°F
$\alpha_{s/\epsilon}$ =0.7	+190°F

ECS

$\alpha_{s/\epsilon}$ =0.4	CAT ≤ 80°F
$\alpha_{s/\epsilon}$ =0.7	CAT ≤ 85°F

Figure 19

SERVICE MODULE
REACTION CONTROL SYSTEM
PROPELLANT MANAGEMENT

BY

OWEN E. MAYNARD

SERVICE MODULE REACTION CONTROL SYSTEM PROPELLANT MANAGEMENT

This discussion of the Service Module RCS propellant management philosophy was not presented during the symposium due to lack of time. It is included in this report to illustrate how the maneuver requirements are being planned to provide maximum propellant reserves to accommodate contingencies.

The Service Module RCS system is used for spacecraft attitude control and for small velocity changes, such as providing main engine ullage.

Figure 1 shows the key points of the SM-RCS propellant usage philosophy. An austere baseline was developed in which only those maneuvers absolutely required were allowed. In addition, the maneuvers allowed are performed with minimum practical rates with maximum utilization of roll maneuvers instead of pitch or yaw to take advantage of the lower inertia. Planning to fly the mission in this way will assure maximum propellants for both expected and unexpected contingencies.

Expected contingencies, that is, contingencies for which RCS propellants have been specifically budgeted are loss of one quad, LM rescue and failure of the MSFN navigation loop. If no contingencies occur prior to the LM rejoining the CSM, then the propellant margin can be used for non-essential maneuvers to further enhance the mission accomplishments.

Figures 2 and 3 show the key features of the austere baseline.

Maneuver requirements during earth orbit are provided by the S-IVB stage reaction control system; hence, these requirements are not included on these figures. Navigation sightings can be made in earth orbit using the S-IVB RCS should this prove to be desirable. The SM RCS is first used for transposition and docking, as described in the session on the general mission description.

MSFN is the prime source of navigation data and only two mid-course corrections are expected during the translunar and transearth phases: one near each end of the phase. The majority of the transit time is spent in a thermal roll mode in which the spacecraft is rolled about its longitudinal axis which is maintained within $\pm 20^{\circ}$ of normal to the vehicle-sun line. In lunar orbit, MSFN is again the prime source of navigation data; however, some sightings will be taken on the landing area for altitude refinement and on a

few other sites for confidence. Other maneuvers required during the lunar orbit phase are as shown on the figures. Since each of these maneuvers is described in the general mission description, they are not described in this paper.

Figure 4 shows the austere maneuver plan used for IMU alignment to illustrate how SM RCS propellants can be conserved if care is taken to preplan the maneuvers for maximum efficiency.

Figures 5 and 6 show the additional maneuvers required to accomplish onboard navigation should the MSFN navigation loop fail.

Figure 7 describes the three LM rescue contingency situations considered. The first two are associated with an on-time launch, and the third one with an anytime launch.

Characteristics of these situations will be discussed in another paper, but the point to be made here is that the LM rescue represents 180 to 300 pounds of the total 790 pounds available, which is 22% to 37% of the total even if the LM performs the docking. The anytime launch situation in which the CSM must perform the docking involves about 45% of the total. Therefore, it is vital to understand the detail requirements leading to such a contingency. These are currently under investigation.

Figure 8 presents the factors and calculations related to an RCS quad failure. The consequences of a quad failure are shown under "consequences". Propellants available after a quad failure are determined based on the equations shown under "calculations". (The second one infers good management to balance quads at all times.)

The items under "checks" indicate that translation demands after a quad failure are significant, and has led to investigation of no ullage starts for the SPS with the LM rescue contingency.

Figure 9 shows the non-scheduled activities still under review.

Growth factors, of course, would be very significant. Additional safety factor considerations would only be identified as experience grows. Unevaluated factors are considered to be small.

In the case of the RCS activities, a thorough understanding of the maneuver requirements are required before they are

admitted into the budget, and, of course, we must understand by simulation and flight test that the system does perform as estimated. A detailed understanding of situations which lead to anytime launch and subsequent LM rescue and finally the ability of the SPS to perform with no ullage starts at low propellant levels must be understood.

This philosophy results in a 36 lb. reserve for the worst case anytime launch (with LM performing docking) with one RCS quad out and with having to do onboard navigation from there on. For the more probable LM rescue contingency associated with on-time launch, the reserve would increase to about 50 pounds or 12% of two good quads.

NASA-S-66-6429 JUN

SM-RCS PROPELLANT USAGE
PHILOSOPHY

- AUSTERE BASELINE
- PLANNED CONTINGENCY RESERVE
  - ONE QUAD OUT
  - LM RESCUE
  - MSFN OUT
- EXERCISE FRUGAL MANAGEMENT OF PROPELLANT PRIOR TO LM REJOINING CSM

FIGURE 1

NASA-S-66-6831 JUN

BASIS FOR AUSTERE PROFILE BASELINE TRANSLUNAR AND TRANSEARTH

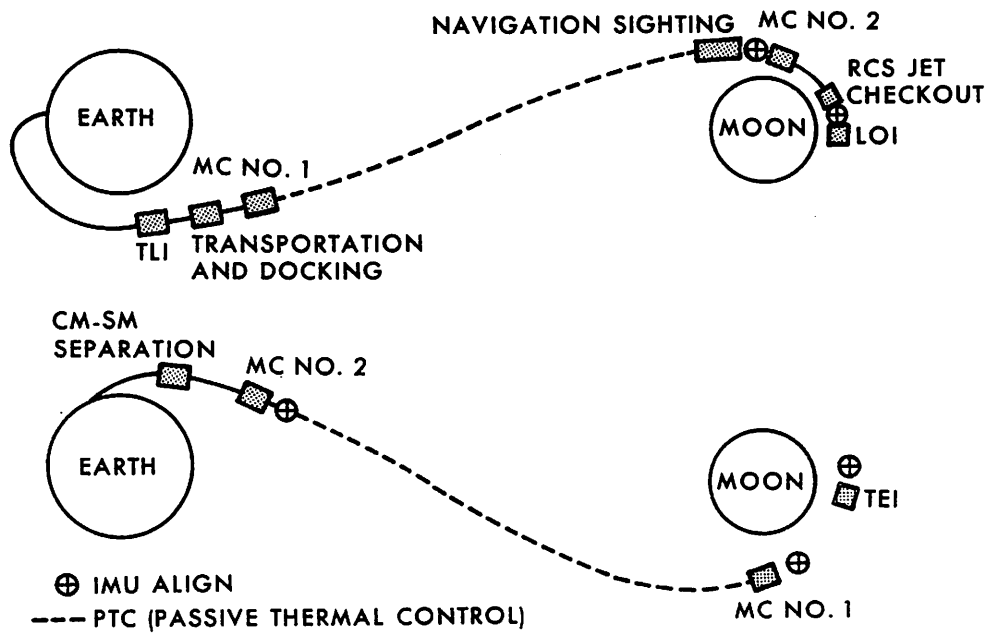


FIGURE 2

NASA-S-66-6431 JUN

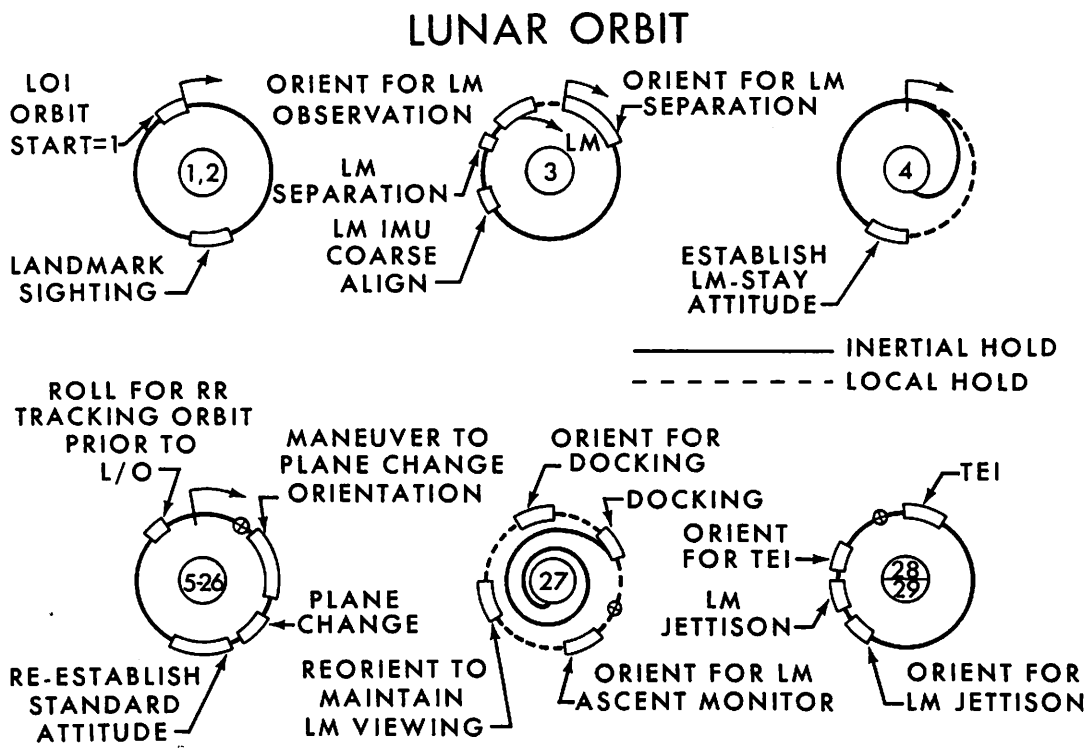


FIGURE 3

NASA-S-66-6454 JUN

EXAMPLES OF EFFICIENT MANEUVERS FOR AUSTERE PROFILE BASELINE

MANUAL ROLL MANEUVER ONLY
REQUIRED FOR SCT ACQUISITION
OF 2 STARS WITH LM OFF

MANUAL PITCH-ROLL ONLY
REQUIRED FOR SCT ACQUISITION
OF 2 STARS WITH LM ON

} ONE G & N ROLL MANEUVER
ONLY REQUIRED FOR SXT
ACQUISITION OF 2 STARS

TWO JET ULLAGE

FIGURE 4

NASA-S-66-6830 JUN

MSFN OUT

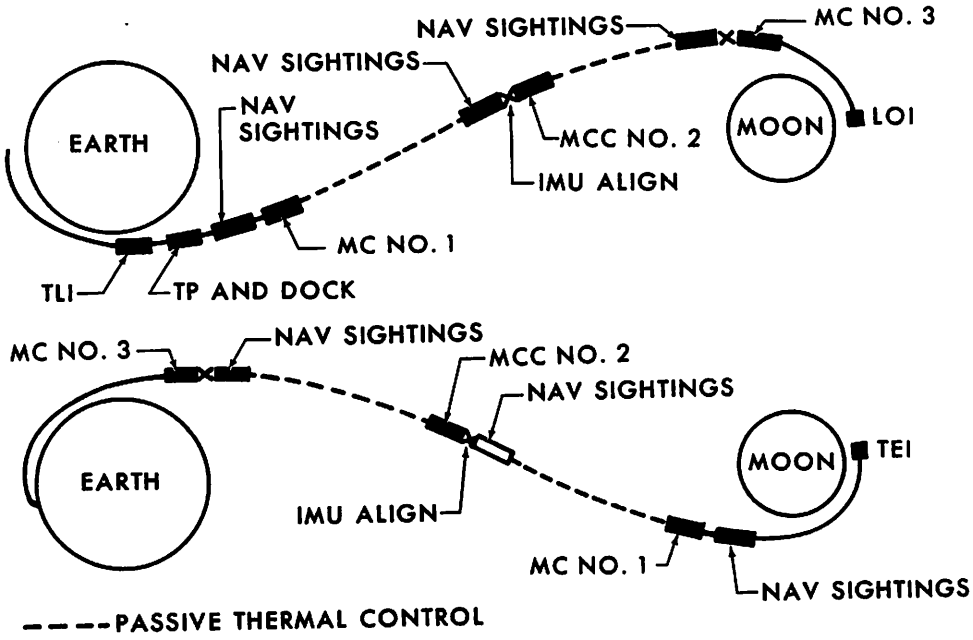


FIGURE 5

NASA-S-66-6832 JUN

MSFN OUT-LUNAR ORBIT

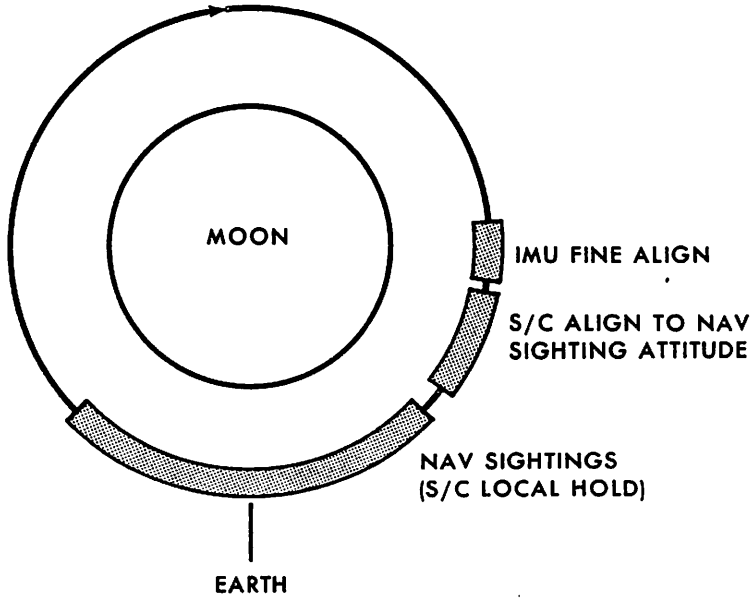
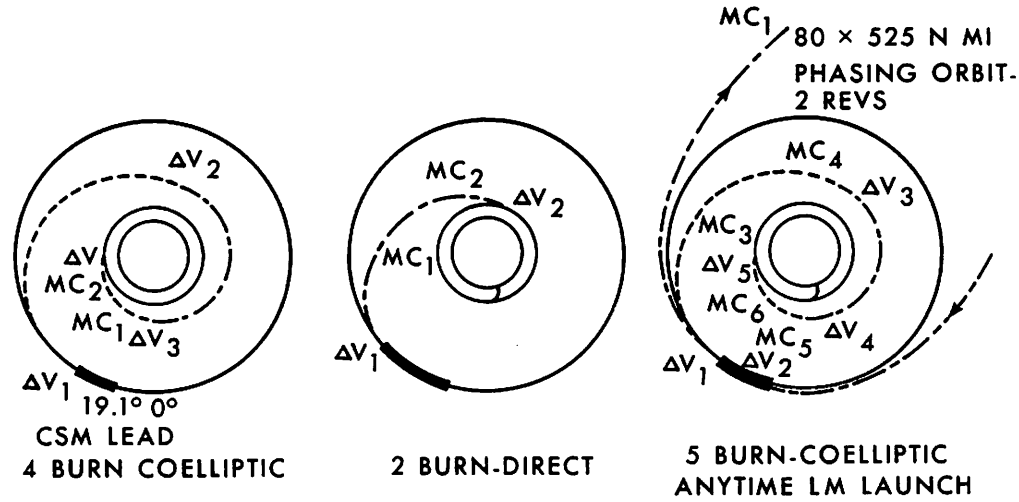


FIGURE 6

NASA-S-66-6829 JUN

LM RESCUE CASES



CSM RCS PROPELLANT FOR LM RESCUE

FULL CSM RESCUE	289 LB	272 LB	395 LB
LM-ACTIVE DOCKING	194	178	301

FIGURE 7

NASA-S-66-6488 JUN

TECHNIQUE FOR EVALUATING QUAD-OUT PERFORMANCE

- CONSEQUENCES
  - LOSS OF REMAINING PROPELLANT IN FAILED QUAD
  - REQUIRES ADDITIONAL PROPELLANT FOR QUAD MANAGEMENT OF REMAINING QUADS
  - TRANSLATION DEMAND MUST COME FROM REMAINING QUAD PAIR
  - ALL SUBSEQUENT ROTATIONAL MANEUVERS MUST BE MANUAL
- CALCULATIONS
  - POSTULATE EQUAL QUAD USAGE PRIOR TO FAILURE
  - PROPELLANT LOSS=PROPELLANT/QUAD=¼ (TOTAL AVAILABLE-AMOUNT USED)
  - PROPELLANT TOTAL RESERVE =3 × PROPELLANT/QUAD
  - PROPELLANT TRANSLATION RESERVE =2 × PROPELLANT/QUAD
- CHECKS
  - TRANSLATION DEMAND POST FAILURE VS TRANSLATION RESERVE
  - TOTAL DEMAND POST FAILURE VS TOTAL RESERVE

FIGURE 8

NASA-S-66-6442 JUN

FLEXIBILITY ACTIVITY/RESERVE

- GROWTH FACTORS
  - LATER MISSIONS
- SAFETY FACTORS
  - UNDEFINED MANUAL MANEUVERS
  - UNPREDICTABLE OPERATIONAL DISPERSIONS
  - CALCULATION APPROXIMATIONS
  - INPUT DATA EXTRAPOLATION ERRORS
  - MISSION UNCERTAINTIES
- UNEVALUATED FACTORS
  - FUEL SLOSHING
  - DISTURBANCE TORQUES
  - BODY BENDING
  - ORIENT AGAINST SOLAR FLARES
  - REPEAT NAV SIGHTINGS
  - TANK TEMPERATURE EFFECTS
  - CROSS COUPLING

FIGURE 9