NAME:____ **CLOSED BOOK SECTION**

MECH 372 Space Systems Design and Engineering II
Module 1 Exam

Winter 2017

For both sections of this exam, write the final answer in the space provided. If no space is provided and there is a numeric answer, then BOX YOUR FINAL ANSWER.

Part A - Closed Book (40 Points)

1. (7) Match each of the missions listed with the single most compelling justification for using a spacecraft in order to achieve the stated mission. Each justification may be used once, more than once, or not at all; but for any one mission, you may only list one justification (or it will be marked wrong).

MISSIONS	JUSTIFICATIONS
G_____ Sputnik	A. Global view of Earth
A_____ NOAA Weather Satellite	B. Above Earth's atmosphere
D_____ Mars Pathfinder Rover	C. Space environmental characteristics
E_____ Asteroid mining spacecraft	D. In situ characterization and exploration
A_____ GeoEye Earth Observation Spacecraft	E. Exploitation of resources
B_____ Webb Space Telescope	F. Space Tourism
C_____ GeneSat-1 biological microgravity mission	G. Pride and distinction

2. (2) State two specific components that would be examples of payload components for a communication satellite.

Could be amplifier, receiver, transmitter, antenna, etc.

3. (2) Imagine a satellite mission that divides its mission architecture into a space segment, a launch segment and a ground segment. List at two (and only two) things that would be typical elements of the ground segment.

Need at least two of the following: communication or tracking station, mission control center, operations crew, engineering staff, ground segment communication links…

4. (1) State Kirchhoff's Voltage Law (this should be a written statement that states the general law – not a specific example with a diagram and/or an equation).

The sum of voltage drops across all elements in any closed current path within a circuit is zero
Or
The sum of the currents entering into any node within a circuit is zero (e.g., what goes in must come out)

5. (1) Consider a simple RC circuit used as a timer, with an output voltage that goes from 0V at t=0 to a steady state voltage of 10V. What is the output voltage when t=RC sec?

~63% of range, which would be ~6.3V

6. (2) In this course, we have considered four primary sources of thermal energy within the space thermal environment in order to perform thermal balance computations (these did not include particle heating). List these four ***sources***.

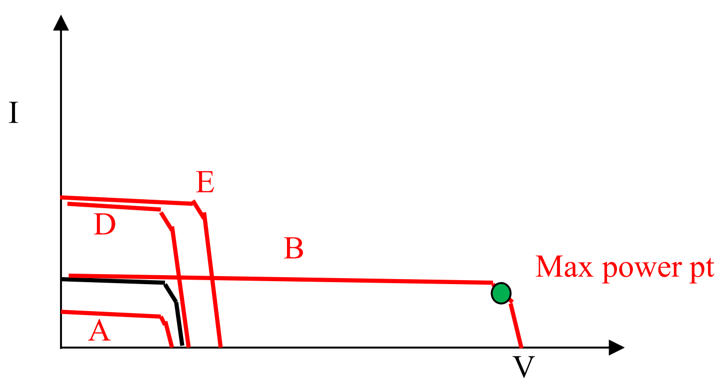
Direct solar, albedo, earth IR, internal dissipation

7. (1) What is a "cold bias" thermal design strategy?

Given uncertainty in thermal design, and to protect the risk of a thermal design that leaves a component too hot, the cold bias strategy intentionally specifies a thermal design that leaves a component too cold with the knowledge that a simple heater can be used to slightly elevate the component's temperature.

8. (5) The I-V curve below indicates the performance of a single photovoltaic cell for a specific temperature and illumination level.

A. Sketch the I-V curve for the same cell operating at ½ the original illumination level, and label this curve "A"
B. Sketch the I-V curve for a string of 4 cells operating in series at the original illumination level, and label this curve "B"
C. Mark and label the approximate maximum power operating point on the curve from part B.
D. Sketch the I-V curve for two cells in parallel, assuming the original illumination and temperature, and label this curve "D"
E. Sketch the curve of the two cells in parallel from part D given the original illumination but at a temperature ~100 deg C colder than that original condition, and label this curve "E".



9. (5) For each part of the question, circle the most appropriate response:

a.	Which passive component stores energy in a magnetic field?	Capacitor	Inductor
b.	Which surface color typically has a higher α/Ɛ ratio?	White surface	Black surface
c.	Which thermal testing type uses more stressful levels?	Qualification	Acceptance
d.	What battery type is designed to be rechargeable?	Primary	Secondary
e.	Which Depth of Discharge % will degrade a battery faster?	30%	70%

10. (14) True or False: Circle the appropriate solution for each individual question.

T F a. A “mission architecture” or “concept of operations” diagram is typically a drawing/chart showing all the main elements of each segment of the mission and how they interact.

T F b. The pn junction is one of the core innovations at the heart of semiconductor devices. A pn junction results in a small E field being established within the device.

T F c. When expressed in standard binary notation, the number 25 may be written as 11001.

T F d. An “OR” digital logic function/gate gives a TRUE/HIGH output if any of its inputs are TRUE/HIGH.

T F e. Whether a component is being used at a particular point in time or not, the thermal subsystem should be designed such that it can maintain the temperature of that component within its “operating temperature range.”

T F f. From a thermal perspective, a “selective surface” is one that has distinctly different values for the infrared vs visible wavelength values of absorptivity (or emissivity).

T F g. Most spacecraft use both active heating and active cooling to maintain the temperature of components.

T F h. The symbol ε is used by spacecraft thermal engineers to express the value of both absorptivity and emissivity in the visual (solar) frequency range.

T F i. In general, joint conductance can be improved by using a soft interstitial material and by using a high contact pressure.

T F j. The Beta angle of a noon-midnight orbit is 0°.

T F k. RTGs use nuclear power strategy such as fission or fusion to produce power.

T F l. It takes about 2 hours to fully charge a battery at a “2C” rate.

T F m. Solar cell efficiency depends on the materials used to make the pn layers of the cell.

T F n. In the context of a satellite’s power budget, a satellite should never have an operating mode that requires more power than can be produced by the satellite’s solar arrays.

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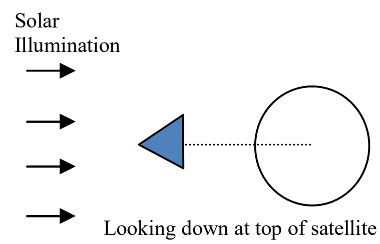
Part B – Open Book (40 points)

11. (15) Consider a triangular prism-shaped satellite that is currently at a location directly between the Sun and Earth. The orbit altitude is 600 km. The triangular top and bottom are perfectly insulated and are equilateral triangles. The sides of the satellite each have an area of 1 m^2. One of these sides points directly at the Earth. The diagram shows this scenario; the triangular "top" of the satellite can be seen in the diagram. The Earth radius is 6378 km. The solar flux is 1358 W/m^2, and the average albedo factor is 0.3. Even though the satellite is in orbit, we will assume that we can analyze the steady state thermal balance at this point in the orbit (the thermal mass of the satellite is very small). PLACE ANSWERS IN THE PROVIDED BOX.



To begin, assume that the spacecraft's surface has values of α=0.4, and ε=0.8.

a) How many Watts are absorbed by the spacecraft due to direct solar heating?
Use the projected area!!! Sun proj area is 1 sq m
Given that the projected area wrt the sun is 1 sq m:
$Q_s=\alpha J_s A_s=0.4*1358*1=543W$

b) How many Watts are absorbed by the spacecraft due to albedo?
The beta is 0 degrees, and from Slide 29 we see F ~ 1
This would mean $Ja = a*J_s *1\ W = 407\ W$
Albedo proj area is 1 sq m
So, $Q_{alb} = 0.4*407*1 = 163\ W$

c) How many Watts are absorbed by the spacecraft due to planetary heating?
Given the geometry, one face is directly facing Earth 1 sq m
Slide 30 gives the approximation $J=(237)*[R/(R+h)]^2=198W$
Other sides see very small part of Earth – F~.15 or so, I'm OK if you
Ignored these contributions – either way works; here, I ignore it
So, $Q_p=\varepsilon J_p A=158W$

d) If the satellite is dissipating 500W of internal energy, what is its steady state temperature at this point in the orbit (making the assumption that the satellite has a very small thermal mass).
$Q_s+Q_{alb}+Q_p+500W = 1364\ W = \varepsilon\sigma T^4 A_{surf}$ where $A_{surf}=3\ m^2$
T ~ 316 °K

e) Now, assume no internal power dissipation and that the desired satellite steady state temperature at this point in the orbit is 10° C. What new α/ε would you select for the surface of the satellite?
$(\alpha/\varepsilon)\ [(J_sA_s+J_{alb}A_{alb})/\sigma A_{surf}] \quad + [J_pA_p / \sigma A_{surf}] = T^4$; $(\alpha/\varepsilon) = 0.506$

Q_{Sun} = ____________

Q_{alb}=____________

Q_{plan}=____________

T_{ss}=____________

(α/ε) =____________

NOTE – when grading parts d and e, I used your numbers for a,b,c and computed it via matlab to check

NOTE – when grading 12, I was looking for the threshold e values for both cases, and then some statement of the range using these values; also, there needed to be a solution or acknowledgement that alpha wasn't directly an issue (although there is of course the limitation of e + a + t = 1)

12. (6) A payload component must be kept at a temperature between 0ºC to 50ºC at all times. The payload dissipates 10W when in standby mode and 100W when fully on and functional. The payload is coupled perfectly to a louvered panel that radiates to deep space; all other sides of the payload and the back side of the louver are perfectly insulated. The size of the louvered radiator is 0.5 m x 0.5 m, and it can be commanded to change between either of two surfaces, each with a different α/Ɛ ratio. Compute what the range of allowable α and Ɛ values must be for each louvered surface (one used for the hot case, and one used for the cold case). **Box your answers.**

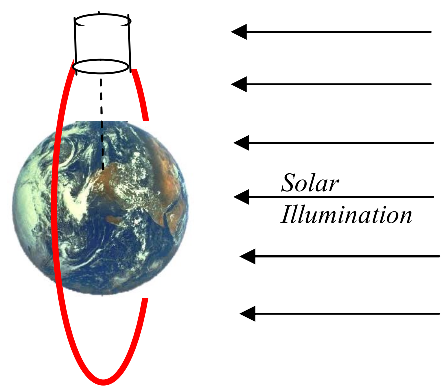
Hot case (T=50C): Qint=100W=e*sigma*T^4*A=e*sigma*$(323)^4$*0.25; so e=0.648 or more
Cold case (T=0C): Qint=10W=e*sigma*T^4*A=e*sigma*$(273)^4$*0.25; so e=0.127 or less
Both cases independent of alpha given no visible input/output

13. (4) A battery pack is assembled using cells that have a voltage of 2.5V and capacity of 1.5 A-hr. The pack consists of strings of 8 cells in series, and 3 of these strings wired in parallel. What is the nominal overall voltage of the battery pack? If the battery pack is operated with a maximum Depth of Discharge of 67%, how much continuous current can be sourced during a 30 minute eclipse?

Pack Voltage = _______________ Continuous Current = ______________________

Pack V = 2.5V * 8 cells = 20V
Parallel configuration provides max capacity of 3 * 1.5 A-hrs = 4.5 A-hrs
75% DoD would provide .67 * 4.5 A-hrs = 3 A-hrs
This can be expended during 30 min, leading to a continuous current of 2 * 3 = 6 Amps

14. (15) Consider a spinning cylindrical satellite in a dusk-dawn (β=90°) low Earth orbit. The spin axis is always aligned with the center of the Earth such that one of the circular ends of the cylinder is always pointed toward the Earth; the other end of the satellite is always pointed away from the Earth. The cylinder rotates about the spin axis such that points on the curved surface rotate in and out of the sun's illumination (the circular ends do not see the sun. The satellite is 1 m in diameter, and you will be determining its height. Solar cells are body mounted and evenly distributed over the exterior of the curved surface of the satellite; there are no solar cells on the circular ends of the satellite. The cells are 27% efficient GaAs with V_{cell}=0.5V and I_{cell}=190mA, and the fill factor is 90%. You may assume that there are no losses due to temperature deviation, shadowing, or regulation efficiency. The satellite's mission life is 5 years, and yearly degradation is 3%. Solar illumination is 1358 W/m^2. Note that this solar array design problem DOES NOT completely follow the example in lecture, which is for a planar array – adapt accordingly. **BOX YOUR ANSWERS.**



a) Given the cell efficiency and lifetime degradation, what is the end of life power density (W/m^2) for a cell that is pointed perfectly at the sun?
P'eol=1358*0.27*$(1-.03)^5$ = 315 W/m^2

b) Given the geometry and size of the satellite (its diameter and height) and the fill factor, how much power does the body mounted solar panel produce at the satellite's end of life, as a function of the satellite's height, "H"?
A_exposed_to_sun = H m^2
P_eol = P'eol * A_exposed_to_sun * fill factor = 284*H W

A major issue in this project was realizing that the panel is body mounted on a cylinder such that solar illumination never lights up all cells – the rect projection of the cylinder is the amount illuminated vs the surf area of the cylinder is the amount you need to populate – there is a factor of pi between the two. You had to account for this in the problem – and to some extent I was a little flexible in where you did this if you did it differently than me.

c) The required end of life solar array power is 500 W. Given this, what is the minimum height of the satellite that allows enough area for the body mounted array?

500 W = 284*H, so, H=1.76 m

d) How many cells total are required for the entire solar array area?

The ratio of total area to the projected/exposed area is pi*d*H / d*H = pi

So the total power producing capability of the array (if completely illuminated) is pi*500=1570 W

So, #cells = Array power / cell power = 1570/(.5*.19) = 16,526

e) If the satellite power bus operates at 28V, approximately how many cells should be placed in series for a given string?

$N_{series} = V_{SA} / V_{cell} = 28/0.5 \sim 56$ cells

f) How would you lay out a string on the external surface of the satellite? Would you arrange the cells in a given string so that they run 'vertically' up/down the side of the satellite (in a line parallel to the satellite's axis of rotation), or would you arrange the cells of a given string so that they wrap around the circumference of the satellite? Why does it make a difference?

You don't the cells to be evenly illuminated, so run them vertically.